HIGHLIGHTS

REVISION NO. 33 Jun 01/10

Pages which have been revised are outlined below, together with the Highlights of the Revision

CH/SE/SU C PAGES	REASON FOR CHANGE	EFFECTIVITY
CHAPTER 22		
L.E.P. 1- 9	Revised to Reflect this revision indicating new,revised, and/or deleted pages	
T. OF C. 6	Revised to reflect this revision	
22-10-00 523- 524	Minor additions and amplification **MODIFIED TEXT IN ACCORDANCE WITH MPD TASK 221320-02-1.**	
22-27-00 510	Minor additions and amplification **UPDATED TO BE IN ACCORDANCE WITH MPD TASK** 222000-03-1	
22-37-00 02 5	Minor additions and amplification **MODIFIED ILLUSTRATION.**	

Printed in France

CHAPTER 22

AUTO FLIGHT

LIST OF EFFECTIVE PAGES

N, R or D indicates pages which are New, Revised or Deleted respectively
Remove and insert the affected pages and complete the Record of Revisions and the Record of Temporary Revisions as necessary

CH/SE/SU	C	PAGE	DATE
RECORD OF TEMP. REVISION			
L.E.P.	R	1- 9	Jun01/10
T. of C.		1	Jun01/09
T. of C.		2	Jun01/07
T. of C.		3	Jun01/07
T. of C.		4	Jun01/07
T. of C.		5	Jun01/06
T. of C.	R	6	Jun01/10
T. of C.		7	Jun01/06
T. of C.		8	Jun01/08
T. of C.		9	Jun01/06
T. of C.		10	Jun01/04
T. of C.		11	Jun01/04
T. of C.		12	Jun01/04
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22-35-12		401	Jun01/04
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22-35-12		403	Jun01/04
22-35-12		404	Jun01/04
22-35-12		405	Jun01/04
22-36-00	01	1	Jun01/04
22-36-00	01	2	Jun01/04
22-36-00	01	3	Jun01/04
22-36-00	01	4	Jun01/04
22-36-00	01	5	Jun01/04
22-36-00	01	6	Jun01/04
22-36-00	01	7	Jun01/04
22-36-00	01	8	Jun01/04
22-36-00	01	9	Jun01/04
22-36-00	02	1	Jun01/04
22-36-00	02	2	Jun01/04
22-36-00	02	3	Jun01/04
22-36-00	02	4	Jun01/04
22-36-00	02	5	Jun01/04
22-36-00	02	6	Jun01/04
22-36-00	02	7	Jun01/04
22-36-00	02	8	Jun01/04
22-36-34	01	401	Jun01/04
22-36-34	01	402	Jun01/04
22-36-34	01	403	Jun01/04
22-36-34	02	401	Jun01/04
22-36-34	02	402	Jun01/04
22-36-34	02	403	Jun01/04
22-37-00	01	1	Jun01/04
22-37-00	01	2	Jun01/04
22-37-00	01	3	Jun01/04
22-37-00	01	4	Jun01/04
22-37-00	01	5	Jun01/04
22-37-00	01	6	Jun01/04
22-37-00	01	7	Jun01/04
22-37-00	01	8	Jun01/04
22-37-00	01	9	Jun01/04
22-37-00	01	10	Jun01/04
22-37-00	01	11	Jun01/04
22-37-00	01	12	Jun01/04
22-37-00	01	13	Jun01/04
22-37-00	01	14	Jun01/04
22-37-00	01	15	Jun01/04
22-37-00	01	16	Jun01/04
22-37-00	01	17	Jun01/04
22-37-00	01	18	Jun01/04
22-37-00	01	19	Jun01/04
22-37-00	01	20	Jun01/04
22-37-00	01	21	Jun01/04
22-37-00	01	22	Jun01/04
22-37-00	01	23	Jun01/04
22-37-00	01	24	Jun01/04
22-37-00	01	25	Jun01/04
22-37-00	01	26	Jun01/04
22-37-00	01	27	Jun01/04
22-37-00	01	28	Jun01/04
22-37-00	01	29	Jun01/04
22-37-00	01	30	Jun01/04
22-37-00	01	31	Jun01/04
22-37-00	01	32	Jun01/04
22-37-00	01	33	Jun01/04
22-37-00	01	34	Jun01/04
22-37-00	01	35	Jun01/04
22-37-00	01	36	Jun01/04
22-37-00	02	1	Jun01/04
22-37-00	02	2	Jun01/04
22-37-00	02	3	Jun01/04
22-37-00	02	4	Jun01/04
22-37-00	02 R	5	Jun01/10
22-37-00	02	6	Jun01/04
22-37-00	02	7	Jun01/04
22-37-00	02	8	Jun01/04
22-37-00	02	9	Jun01/04
22-37-00	02	10	Jun01/04
22-37-00	02	11	Jun01/04
22-37-00	02	12	Jun01/04
22-37-00	02	13	Jun01/04
22-37-00	02	14	Jun01/04
22-37-00	02	15	Jun01/04
22-37-00	02	16	Jun01/04
22-37-00	02	17	Jun01/04
22-37-00	02	18	Jun01/04
22-37-00	02	19	Jun01/04
22-37-00	02	20	Jun01/04
22-37-00	02	21	Jun01/04
22-37-00	02	22	Jun01/04
22-37-00	02	23	Jun01/04
22-37-00	02	24	Jun01/04
22-37-00	02	25	Jun01/04
22-37-00	02	26	Jun01/04
22-37-00	02	27	Jun01/04
22-37-00	02	28	Jun01/04
22-37-00	02	29	Jun01/04
22-37-00	02	30	Jun01/04
22-37-00	02	31	Jun01/04
22-37-00	02	32	Jun01/04
22-37-00	02	33	Jun01/04
22-37-00	02	34	Jun01/04
22-37-00	02	35	Jun01/04
22-37-00	02	36	Jun01/04
22-37-00	02	37	Jun01/04
22-37-00	02	38	Jun01/04
22-37-00	02	39	Jun01/04
22-40-00		1	Jun01/05
22-40-00		2	Jun01/05
22-40-00		3	Jun01/05
22-40-00		4	Jun01/05
22-40-00		5	Jun01/98
22-40-00		6	Jun01/04
22-40-00		7	Jun01/04
22-40-00		8	Jun01/04
22-40-00		9	Jun01/04
22-40-00		10	Jun01/04
22-40-00		11	Jun01/04
22-40-00		12	Jun01/04
22-40-00		13	Jun01/04
22-40-00		14	Jun01/04
22-40-00		15	Jun01/04
22-40-00		16	Jun01/04
22-40-00		17	Jun01/04
22-40-00		18	Jun01/04
22-40-00		19	Jun01/04
22-40-00		20	Jun01/04
22-40-00		501	Dec01/97
22-40-00		502	Mar01/03
22-40-00		503	Jun01/04
22-40-00		504	Jun01/04
22-42-00		1	Jun01/05
22-42-00		2	Dec01/90
22-42-00		3	Jun01/05
22-42-00		4	Dec01/90
22-42-00		5	Dec01/90
22-42-00		6	Dec01/94

CH/SE/SU	C	PAGE	DATE	CH/SE/SU	C	PAGE	DATE	CH/SE/SU	C	PAGE	DATE
22-42-00		7	Dec01/94	22-43-00		17	Dec01/94				
22-42-00		8	Dec01/94	22-43-00		18	Dec01/94				
22-42-00		9	Jun01/05	22-43-00		19	Dec01/94				
22-42-00		10	Dec01/94								
22-42-00		11	Dec01/94								
22-42-00		12	Dec01/94								
22-42-00		13	Jun01/08								
22-42-00		14	Jun01/08								
22-42-00		15	Jun01/08								
22-42-00		16	Jun01/08								
22-42-00		17	Jun01/08								
22-42-00		18	Jun01/08								
22-42-00		19	Jun01/08								
22-42-00		20	Jun01/08								
22-42-00		21	Dec01/94								
22-42-00		22	Dec01/94								
22-42-00		23	Dec01/94								
22-42-00		24	Dec01/94								
22-42-00		25	Dec01/94								
22-42-00		26	Dec01/94								
22-42-00		27	Dec01/94								
22-42-00		28	Dec01/94								
22-42-00		29	Dec01/94								
22-42-00		30	Dec01/94								
22-42-00		31	Jun01/04								
22-42-00		32	Jun01/04								
22-42-00		33	Dec01/94								
22-42-00		34	Dec01/94								
22-42-00		35	Dec01/94								
22-42-00		36	Dec01/94								
22-42-00		37	Jun01/04								
22-42-00		38	Jun01/04								
22-42-21		401	Dec01/90								
22-42-21		402	Dec01/90								
22-43-00		1	Dec01/94								
22-43-00		2	Dec01/90								
22-43-00		3	Dec01/90								
22-43-00		4	Jun01/04								
22-43-00		5	Dec01/94								
22-43-00		6	Dec01/90								
22-43-00		7	Dec01/94								
22-43-00		8	Dec01/94								
22-43-00		9	Dec01/94								
22-43-00		10	Dec01/94								
22-43-00		11	Dec01/94								
22-43-00		12	Dec01/94								
22-43-00		13	Dec01/94								
22-43-00		14	Dec01/94								
22-43-00		15	Dec01/94								
22-43-00		16	Dec01/94								

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CHAPTER 22

AUTO FLIGHT

TABLE OF CONTENTS

	SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
	AUTO FLIGHT	22-00-00			
	Description and Operation			1	ALL
	Introduction			1	ALL
	System Description			1	ALL
	System Failure Behaviour			2	ALL
	Definitions				
	Description and Operation			3	ALL
	Introduction			3	ALL
	Integration into Aircraft			9	ALL
	Control and Indicating			27	ALL
	Connection with Flight Controls			36	ALL
	Connection with Engine Controls			43	001-100,
	Connection with Engine Controls			43	101-200,
	Computer Architecture			49	ALL
	Connections Between Component			53	ALL
	Associated with AFS				
	Connections between AFS			53	ALL
	Components				
	Connections between Peripherals			53	ALL
	and AFS				
	Maintenance			53	ALL
	Adjustment/Test			501	ALL
	Operational Test			501	ALL
	Power Supply Cutoff and			501	ALL
	Restoration Logic				
	AFS Light Test and Display			503	ALL
	AUTOPILOT	22-10-00			
	Description and Operation			1	ALL
	Introduction			1	ALL
	Component Location			7	ALL
	Functions Provided - Sub System			11	ALL
	Architecture				
	Adjustment/Test			501	ALL
	Operational Test			501	ALL
	FD Engage Logic			501	ALL
	AP Test in CWS Mode			503	ALL
	AP Test in CMD Mode,			504	ALL
	AP Instinctive Disconnect P/B				
	Test, AP OFF Warning Test				
R	Check of Pitch Autotrim			508	ALL
R	Circuits				
	AP Pitch Mechanical Override			511	ALL
	AP Roll Mechanical Override			514	ALL
	AP Yaw Mechanical Override			516	ALL
	Altitude Alert Test			518	ALL
	TO/GA Lever Test			519	ALL

Printed in France

SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
AP Disconnection through Torque Limiters			522	ALL
Deactivation/Reactivation			901	ALL
ELECTRICAL AND HYDRAULIC POWER SUPPLIES	22-12-00			
Description and Operation			1	ALL
Electrical Power Supply			1	ALL
Aircraft Electrical Supplies			1	ALL
Internal Power Supply Automatic Cut Off			3	ALL
Hydraulic Power Supply			5	ALL
FLIGHT CONTROL/ACTUATOR, FLAP AND SLAT, LANDING GEAR INTERFACE	22-13-00			
Description and Operation			1	ALL
Main Flight Controls			1	ALL
Torque Limiter Levers			1	ALL
Dynamometric Rods			2	ALL
Secondary Flight Controls			7	ALL
AUTOPILOT PITCH ACTUATOR	22-13-23			
Removal/Installation			401	ALL
Servo-Valve, Pitch-Actuator			413	ALL
Removal/Installation			413	ALL
Solenoid, Pitch Actuator			418	ALL
Removal/Installation			418	ALL
Filter, Pitch Actuator			423	ALL
Removal/Installation			423	ALL
Strainer, Pitch Actuator			427	ALL
Removal/Installation			427	ALL
PITCH TORQUE LIMITER LEVER 19CA	22-13-24			
Removal/Installation			401	ALL
AUTOPILOT ROLL ACTUATOR	22-13-25			
ROLL ACTUATOR			401	ALL
Removal/Installation			401	ALL
SERVO-VALVE			412	ALL
Removal/Installation			412	ALL
SOLENOIDS			417	ALL
Removal/Installation			417	ALL
FILTER			423	ALL
Removal/Installation			423	ALL
STRAINER			428	ALL
Removal/Installation			428	ALL
AUTOPILOT YAW ACTUATOR 22CA	22-13-27			
YAW ACTUATOR			401	ALL
Removal/Installation			401	ALL
SERVO-VALVE			412	ALL
Removal/Installation			412	ALL
SOLENOIDS			416	ALL
Removal/Instalation			416	ALL
FILTERS			421	ALL
Removal/Installation			421	ALL
STRAINERS			425	ALL

SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
Removal/Installation			425	ALL
YAW TORQUE LIMITER LEVER 23CA	22-13-28			
Removal/Installation			401	ALL
INTERFACE WITH PERIPHERAL SYSTEMS	22-14-00			
Description and Operation			1	ALL
Introduction			1	ALL
Signals Exchanged with Peripheral Systems			3	ALL
ADS (Air Data System)			3	ALL
IRS (Inertial Reference System)			4	ALL
Radio Navigation Equipment			8	ALL
Signals Exchanged with the FMC			9	ALL
Signals Exchanged with Other AFS Sub-Systems			10	ALL
Analog Signals			10	ALL
Discrete Signals			14	ALL
ARINC Signals			20	ALL
FLIGHT CONTROL UNIT - DISPLAYS, WARNINGS, LIGHTING	22-15-00			
Description and Operation			1	ALL
General			1	ALL
Flight Control Unit (FCU)			2	ALL
Description			2	ALL
FCU Operation			13	ALL
FCU Utilization			22	ALL
FCU Operational Logic			44	ALL
Individual Controls Located in Flight Compartment			47	ALL
AP Disconnect Pushbutton Switches			47	ALL
TO/GA Levers			47	ALL
FD/FPV Selector Switches			47	ALL
FD Bars Switching Pushbutton Switches			47	ALL
Flight Mode Annunciators			51	ALL
General			51	ALL
Longitudinal Modes			53	ALL
Lateral Modes			54	ALL
AP/FD Common Modes			55	ALL
AP/FD Engagement			55	ALL
Capability Indication			55	ALL
Operational Logic			56	ALL
FLIGHT CONTROL UNIT (303CA)	22-15-12			
Removal/Installation			401	ALL
AP DISCONNECT SWITCH (31CA, 32CA)	22-15-13			
Removal/Installation			401	ALL
ENGAGEMENT, BASIC MODES : FD, CWS, CMD	22-16-00			
Description and Operation			1	ALL
Flight Director			1	ALL
Engagement			1	ALL

SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
Disengagement			5	ALL
FD Bar Removal			7	ALL
FD Bar Switching			11	ALL
FCC-PFD Interface			11	ALL
FD Command Generation			11	ALL
Autopilot			16	ALL
Principles			16	ALL
AP Engagement in CWS Mode			25	ALL
AP Engagement in CMD Mode (Basic Modes)			36	ALL
AP OFF Warning			60	ALL
UPPER MODES	22-17-00			
Description and Operation			1	ALL
General			1	ALL
LONGITUDINAL UPPER MODES	22-17-10			
Description and Operation			1	ALL
Altitude Hold Mode (ALT)			1	ALL
Altitude Acquisition Mode (ALT ACQ)			9	ALL
Level Change Mode (LVL/CH)			20	ALL
PROFILE Mode (Coupling to FMC)			28	ALL
Altitude Alert			48	ALL
LATERAL UPPER MODES	22-17-20			
Description and Operation			1	ALL
Heading Selection Hold Mode (HDG SEL)			1	ALL
Navigation Mode (NAV)			7	ALL
VOR Mode			16	ALL
LOC Mode			25	ALL
COMMON UPPER MODES	22-17-30			
Description and Operation			1	ALL
Take off Mode (TO)			1	ALL
LAND Mode			13	ALL
GO AROUND Mode (GA)			44	ALL
FLIGHT CONTROL COMPUTER	22-18-00			
Description and Operation			1	ALL
Introduction			1	ALL
Digital Part Architecture			1	ALL
Architecture of the FCC from the Safety Aspect			20	ALL
FLIGHT CONTROL COMPUTER (301CA1, 301CA2)	22-18-34			
Removal/Installation			401	ALL
SPEED - ATTITUDE CORRECTION	22-20-00			
Description and Operation			1	ALL
Introduction			1	ALL
Component Location			2	ALL
Sub-System Functions			6	ALL
Yaw Damper			6	ALL
Pitch Trim			6	ALL
Flight Envelope Protection			6	ALL

SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
Maneuvering Speed Computation			7	ALL
Configuration Data Processing			7	ALL
ELECTRICAL AND HYDRAULIC POWER SUPPLY	22-22-00			
Description and Operation			1	ALL
Electrical Power Supply			1	ALL
General			1	ALL
Internal Power Supply Automatic Cutoff			5	ALL
Short Cutoff Protection			5	ALL
Hydraulic Power Supply			7	ALL
General			7	ALL
Yaw Damper Actuator Hydraulic Power Supply			7	ALL
FLIGHT CONTROL INTERFACE	22-23-00			
Description and Operation			1	ALL
Pitch Trim			1	ALL
Yaw Damper			5	ALL
Flap and Slat Data			12	ALL
Landing Gear Data			23	ALL
Spoilers/Speed Brakes and Control			29	ALL
Control Wheel Deflection Data				
YAW DAMPER ACTUATOR (3O4CC)	22-23-22			
Removal/Installation			401	ALL
Procedure			401	ALL
PITCH TRIM ACTUATOR 3CC1 (3CC2)	22-23-29			
Removal/Installation			401	ALL
SERVO VALVE	22-23-51			
Removal/Installation			401	ALL
Procedure			401	ALL
INTERFACE WITH PERIPHERAL SYSTEMS	22-24-00			
Description and Operation			1	ALL
General			1	ALL
ARINC Input Signals			1	ALL
ARINC Output Signals			8	ALL
Non Digital Input Signals			20	ALL
Non Digital Output Signals			27	ALL
Interconnection between FACs and Alpha Probes			30	ALL
Interconnection between FACs and Actuators			39	ALL
Interconnection between FACs and Stall Warning System			41	ALL
Interconnection Between FACs and EFIS			43	ALL
PITCH TRIM AND YAW DAMPER CONTROL UNIT - DISPLAYS, WARNINGS	22-25-00			
Description and Operation			1	ALL
Warnings			1	ALL

SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
Display			1	ALL
Trim Rotation Aural Warning			1	ALL
FAC/ATS ENGAGE UNIT (318CC)	22-25-12			
Removal/Installation			401	ALL
YAW DAMPER	22-26-00			
Description and Operation			1	ALL
General			1	ALL
Dutch Roll Damping			1	ALL
Turn Coordination			4	ALL
Engine Failure Automatic Compensation Function			4	ALL
Processing of the Three Function Signals			7	ALL
Power Loop			7	ALL
Yaw Damper Engage Logic			9	ALL
Adjustment/Test			501	ALL
Test of Mechanical Linkage between Yaw Damper Actuator and Rudder			501	ALL
Turn Coordination			503	ALL
FLIGHT AUGMENTATION COMPUTER (301CC1, 302CC2)	22-26-34			
Removal/Installation			401	ALL
ELECTRIC PITCH TRIM	22-27-00			
Description and Operation			1	ALL
General			1	ALL
Electric Trim Function			3	ALL
Theta Trim Function			4	ALL
Auto Trim Function			8	ALL
Static Stability Correctors			10	ALL
Incidence Trim Function			12	ALL
Pitch Trim Engage Logic			16	ALL
Monitoring of Parameters Used by the Pitch Trim System			27	ALL
Equalization Principle			32	ALL
Pitch Trim Operation - Change-Over Principle			32	ALL
Pitch Trim Power Loop			35	ALL
Adjustment/Test			501	ALL
Operational Test			501	ALL
TRIM Engage Logic			501	ALL
TRIM Disconnection Override			502	ALL
Override Test			503	ALL
Pitch Trim 1 Disengagement by AP			505	ALL
Pitch Torque Limiter Test				
Test of Pitch Trim 2 Operation in Single Channel Mode			507	ALL
Supply of FAC 1 in Ultimate Emergency			508	ALL
Detailed Inspection of			510	ALL

Printed in France

	SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
	the Autotrim Function				
	Pitch Trim Control Switch 7CC (8CC)	22-27-12			
	Removal/Installation			401	ALL
R	STANDBY ALPHA PROBE (27CC)	22-27-39			
R	Removal/Installation			401	ALL
	FLIGHT ENVELOPE PROTECTION	22-28-00			
	Description and Operation			1	ALL
	General			1	ALL
	Airspeed Information Display			1	ALL
	Alpha-Floor and Windshear Protection			11	ALL
	Windshear Detection and Warning			11	ALL
	Adjustment/Test			501	ALL
	Operational Test			501	ALL
	ALPHA FLOOR Function (Throttle Pusher)			501	ALL
	Speed Limits Computation			501	ALL
	FAC SPECIFIC MONITORING	22-29-00			
	Description and Operation			1	ALL
	FAC General Monitoring			1	ALL
	General			1	ALL
	Watchdog Description			1	ALL
	Real-Time Monitor Description			1	ALL
	FAC Self-Test			5	ALL
	General			5	ALL
	Activation Conditions			5	ALL
	Safety Test Principle			5	ALL
	FAC Maintenance			9	ALL
	General			9	ALL
	Fault Isolation Operation and Activation			9	ALL
	Precautions			9	ALL
	Adjustment/Test			501	ALL
	Operational Test			501	ALL
	Landing Gear Relay Test			501	ALL
	AUTOTHROTTLE	22-30-00 01			
	Description and Operation			1	001-100,
	Introduction			1	001-100,
	Component Location			2	001-100,
	Thrust Control Computer System Functions			8	001-100,
	Computation and Display of N1 Limit Value			8	001-100,
	Automatic Thrust Control			9	001-100,
	Function and Operation of Power Management Control			13	001-100,
	AUTOTHROTTLE	22-30-00 02			
	Description and Operation			1	101-200,
	Introduction			1	101-200,

SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
Component Location			4	101-200,
Thrust Control Computer System Functions			7	101-200,
Computation and Display of EPR Limit Value			7	101-200,
Automatic Thrust Control			8	101-200,
Function and Operation of Electronic Engine Control			8	101-200,
AUTOTHROTTLE	22-30-00	01		
Adjustment/Test			501	101-200,
Operational Test			501	101-200,
Alpha Floor Function			501	101-200,
EPR Limit Computation			504	101-200,
TAT Display			505	101-200,
ATS OFF Warning			507	101-200,
AUTOTHROTTLE	22-30-00	02		
Adjustment/Test			501	001-100,
Operational Test			501	001-100,
Alpha Floor Function			501	001-100,
Actuator Clutch and TO/GA Lever Operation			503	001-100,
N1 Limit Computation			505	001-100,
TAT Display			506	001-100,
ATS OFF Warning			508	001-100,
POWER SUPPLY	22-32-00			
Description and Operation			1	ALL
Aircraft Electrical Power Supplies			1	ALL
TCC Internal Power Supply Units and Power Cut Off Management			5	ALL
FLIGHT CONTROL/ENGINE INTERFACE	22-33-00	01		
Description and Operation			1	001-100,
Interface With Engine Controls			1	001-100,
Autothrottle Actuator 12CJ - Description			1	001-100,
Tee Gearbox 12CJ - Description			1	001-100,
LH and RH Coupling Units 3CJ1 and 3CJ2 - Description			7	001-100,
Mechanical Performance			7	001-100,
Dynamometric Rods - Description			10	001-100,
Throttle Position Detectors - Description			12	001-100,
Interface with Flight Controls			16	001-100,
Aircraft Configuration Data			19	001-100,
Flap and Slat Data			19	001-100,
Landing Gear Data			19	001-100,
Air Conditioning Data - Engine And Wing Anti-Icing			22	001-100,
FLIGHT CONTROL/ENGINE INTERFACE	22-33-00	02		
Description and Operation			1	101-200,
Interface With Engine Controls			1	101-200,

	SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
	Autothrottle Actuator 12CJ - Description			1	101-200,
	Tee Gearbox 12CJ - Description			1	101-200,
	LH and RH Coupling Units 3CJ1 and 3CJ2 - Description			5	101-200,
	Mechanical Performance			5	101-200,
	Dynamometric Rods - Description			8	101-200,
	Throttle Position Detectors - Description			9	101-200,
	Interface with Flight Controls			14	101-200,
	Aircraft Configuration Data			16	101-200,
	Flap and Slat Data			16	101-200,
	Landing Gear Data			16	101-200,
	Air Conditioning Data - Engine And Wing Anti-Icing			19	101-200,
	AUTOTHROTTLE ACTUATOR 12CJ	22-33-00			
	Deactivation/Reactivation			901	ALL
	Deactivation of the autothrottle actuator 12CJ			901	ALL
	Ref. MMEL 22-30-01 3 - Autothrottle Actuator (FAA only)			901	ALL
	Reactivation of the autothrottle actuator			903	ALL
	AUTOTHROTTLE CONTROL	22-33-12			
	Removal/Installation			401	ALL
R	THROTTLE POSITION DETECTORS 313CJ, 314CJ	22-33-16	01		
R	Removal/Installation			401	001-100,
	THROTTLE POSITION DETECTORS 313CJ, 314CJ	22-33-16	02		
	Removal/Installation			401	101-200,
	AUTOTHROTTLE ACTUATOR 12CJ	22-33-22			
	Removal/Installation			401	ALL
	AT COUPLING UNIT 3CJ1 (3CJ2)	22-33-28	01		
	Removal/Installation			401	001-100,
R	AT COUPLING UNIT 3CJ1 (3CJ2)	22-33-28	02		
R	Removal/Installation			401	101-200,
	INTERFACE WITH PERIPHERAL SYSTEMS	22-34-00	01		
	Description and Operation			1	001-100,
	Interface with Peripheral Systems			1	001-100,
	Air Dat Computer 1(2)			7	001-100,
	Inertial Reference Unit			8	001-100,
	Radio Altimeter Transceivers			8	001-100,
	Flight Management Computer 1(2)			9	001-100,
	Power Management Control			10	001-100,
	Signals Exchanged with AFS Sub-Systems and Components			11	001-100,
	Analog Signals			11	001-100,
	Discrete Signals			14	001-100,
	ARINC Signals			28	001-100,

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SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
INTERFACE WITH PERIPHERAL SYSTEMS	22-34-00	02		
Description and Operation			1	101-200,
Interface with Peripheral Systems			1	101-200,
Air Data Computer 1(2)			5	101-200,
Inertial Reference Unit			6	101-200,
Radio Altimeter Transceivers			7	101-200,
Flight Management Computer 1(2)			8	101-200,
Electronic Engine Controls			10	101-200,
Signals Exchanged with AFS			10	101-200,
Sub-Systems and Components				
Analog Signals			10	101-200,
Discrete Signals			16	101-200,
ARINC Signals			23	101-200,
THRUST RATING PANEL - DISPLAYS, WARNINGS	22-35-00	01		
Description and Operation			1	101-200,
Thrust Rating Panel			1	101-200,
EPR Limit Recopy on EPR Indicators			14	101-200,
EPR Limit Manual Setting				
TCCS Indicating and Warnings on			16	101-200,
EFIS Displays				
ATS OFF Warning			18	101-200,
THRUST RATING PANEL - DISPLAYS, WARNINGS	22-35-00	02		
Description and Operation			1	001-100,
Thrust Rating Panel			1	001-100,
N1 Limit Recopy on N1 Indicators			16	001-100,
N1 Limit Manual Setting				
TCCS Indicating and Warnings on			18	001-100,
EFIS Displays				
ATS OFF Warning			20	001-100,
THRUST RATING PANEL (303CJ)	22-35-12			
Removal/Installation			401	ALL
Pushbutton Switch Housing			404	ALL
THRUST RATING COMPUTING SYSTEM	22-36-00	01		
Description and Operation			1	101-200,
Thrust Control Computer (TCC) - Description			1	101-200,
EPR Limit Computation			1	101-200,
THRUST RATING COMPUTING SYSTEM	22-36-00	02		
Description and Operation			1	001-100,
Thrust Control Computer (TCC) - Description			1	001-100,
N1 Limit Computation			3	001-100,
General			3	001-100,
Rating Computation			4	001-100,
THRUST CONTROL COMPUTER 301CJ1	22-36-34	01		
Removal/Installation			401	101-200,
THRUST CONTROL COMPUTER 301CJ1 (301CJ2)	22-36-34	02		
Removal/Installation			401	001-100,

SUBJECT	CH/SE/SU	C	PAGE	EFFECTIVITY
AUTOTHROTTLE SYSTEM	22-37-00	01		
Description and Operation			1	001-100,
ATS Arming			1	001-100,
ATS Engagement			6	001-100,
AUTOTHROTTLE SYSTEM	22-37-00	02		
Description and Operation			1	101-200,
ATS Arming			1	101-200,
ATS Engagement			7	101-200,
SYSTEM MONITOR	22-40-00			
Description and Operation			1	ALL
General			1	ALL
Introduction			1	ALL
Fault Isolation and Detection			1	ALL
System Functions				
R Failure Indication			1	ALL
Trouble Shooting Procedure			1	ALL
Additional Recommendations			3	ALL
Component Location			4	ALL
Power Supply			7	ALL
Power Supply Automatic Restoration and Cut Off			7	ALL
Interface with Peripherals			7	ALL
ARINC Input Signals			10	ALL
ARINC Output Signals			19	ALL
Discrete Signals			20	ALL
Circuits Integrated in the Different Computers			20	ALL
Adjustment/Test			501	ALL
Operational Test			501	ALL
AFS Test			501	ALL
Land Test and Autoland Lights Test			502	ALL
FAULT ISOLATION PROCEDURE	22-42-00			
Fault Isolation Procedure Initiation			1	ALL
Fault Isolation Process			1	ALL
Use of the Maintenance Test Panel			6	ALL
Use of the DISPLAY Function			6	ALL
MAINTENANCE TEST PANEL (3CU)	22-42-21			
Removal/Installation			401	ALL
Equipment and Materials			401	ALL
Procedure			401	ALL
COMPLEX FAILURE ISOLATION SUPPORT	22-43-00			
Context Table Inside the MTP			1	ALL
MTP Context Description			1	ALL
MTP Context Reset			3	ALL
Context Table Inside the Computer			4	ALL
Context Description			6	ALL

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SUBJECT		CH/SE/SU	C	PAGE	EFFECTIVITY
	Details on the First Part of the Context			7	ALL
	Second and Third Words			8	ALL

1. Introduction

A. System Description

The auto flight system is made up of the following sub-systems :

(1)Flight control computer system (FCCS : 22-10). The FCCS fulfills several functions :

(a)Flight director (FD)

The pitch, yaw and roll guidance commands are computed and displayed to provide the crew with the means of following or acquiring flight paths or parameters.

(b)Autopilot in CMD mode (command).

The guidance principle is identical to the FD one but the hydraulic servo actuators are automatically controlled.

(c)Autopilot in CWS mode (control-wheel steering). The CWS mode is used in manual control although it can follow the commands delivered by the FD. The pitch and roll attitude is maintained until the control column is actuated. When the control column is actuated the AP pitch and roll actuators assist the pilot's movements.

(2)Flight augmentation computer system (FACS : 22-20).

The FACS fulfills the following functions :

(a)Pitch trim

The pitch trim function enables loads applied to the control column to be overridden (modification of the elevator stability position by trimmable horizontal stabilizer deflection).
Deflection is controlled either manually or automatically.

(b)Yaw damper

The yaw damper function serves to improve the lateral stability of the aircraft (Dutch roll). It is also used for engine failure compensation and turn coordination in manual flight.

(c)Miscellaneous functions

The FAC provides the following functions :
Computation and display of the safety speeds, flap/slat and landing gear relay monitoring as well as high angle-of-attack protection.

(3)Thrust control computer system (TCCS : 22-30)

The TCCS fulfills the following functions :

(a)Thrust computation

The maximum usable thrust is automatically computed then displayed.

(b)Autothrottle function

Thrust management is achieved by the TCC which operates the throttle control levers either to hold a speed or mach value or to command maximum or minimum thrust. The autothrottle modes are automatically selected by the FCC during AP/CMD or FD mode changes.

(4)System monitor (MTP : 22-40)

The MTP is in charge of the AFS maintenance by failure context recording. It is also used to perform ground tests.
These sub-systems include the computers, actuators, control units and associated peripherals.

B. System Failure Behaviour Definitions

The terms currently in use are defined below :

Fail-soft

An automatic flight system is fail-soft if, in the event of a failure, there is limited deviation of trim, flight path or attitude but the failure is not automatically detected.

An automatic flight system is fail-soft if, in the event of a failure, there is limited deviation of trim, flight path or attitude but the failure is not automatically detected.

Fail-passive

An automatic flight system is fail-passive if, in the event of a failure, there is no significant deviation of trim, flight path or attitude but the failure is automatically detected and the remaining part of the flight will be completed manually by the pilot.

Fail-operational

An automatic flight system is fail-operational if, in the event of a failure, a secondary system :

- either ensures automatic control of the aircraft
- or provides guidance commands to permit completion of the flight by the pilot.

2. Description and Operation

A. Introduction

(1) General AFS sub-system description

(a) Flight control computer system

The flight control computer system is designed in compliance with ARINC Characteristic 701.

It comprises two identical units which provide pitch, roll and yaw control signals to the autopilot actuators and to the primary flight display (PFD) command bars.

The yaw control is only activated when the slats are extended.

The AP and FD functions have common computation channels.

The auto flight system is certified for category III autoland with two FCCs.

The following table details the functions and available modes together with the sub-system failure behaviour for each possible configuration.

FUNCTIONS AND MODES	FAILURE BEHAVIOUR		
	FAIL-SOFT	FAIL-PASSIVE	FAIL-OPERATIONAL
FLIGHT DIRECTOR			
- Basic Modes V/S (Acquisition/Hold of vertical speed upon engagement then upon selection) HDG (Heading hold upon engagement)	X (outer loop for V/S)	X (inner loop and FD bar slaving)	X Since both FD's are always engaged, in the event of failure of one of these, the other continues to supply data to the associated PFD. Furthermore the PFD on failed side can always be manually switched to the operational FD.
- Upper Unmonitored Modes ALT and LVL/CH (Selected altitude Acq/Hold and level change) PROFILE (Vertical path Acq/Hold: connection with FMC) HDG SEL (Selected heading Acq/Hold) VOR (VOR radial Acq/Hold) LOC (LOC beam Acq/Hold) NAV (Lateral track Acq/Hold : connection with FMC)	X (outer loop)	ditto	
- Upper Monitored Modes TO (Takeoff) LAND (FD approach and landing) GO AROUND (Go around)		X	
AUTOPILOT			
- Basic Mode Same as FD		X (inner loop)	
- CWS		X	
- Upper Unmonitored Modes Same as FD	X (outer loop)		
- Upper Monitored Modes			X Faulty AP disconnects. Automatic change-over to operational AP occurs in dual AP.

NOTE : Yaw AP operated at slat extension is fail-passive.

(b) Flight augmentation computer system

The flight augmentation computer system provides some of the functions specified by ARINC Characteristic 701 and associated with aircraft handling qualities.

This sub-system comprises two identical units.

Each unit provides the following functions :

- yaw damping with engine failure automatic compensation (AP engaged in CMD),
- pitch trim,
- flight envelope protection, speed and angle-of-attack protection (particularly in the case of windshear),
- computation of maneuvering speeds (indicated by the PFD)
- specific monitoring.

The following table details the functions and sub-system characteristics.

FUNCTIONS AND MODES	FAILURE BEHAVIOUR		
	FAIL-SOFT	FAIL-PASSIVE	FAIL-OPERATIONAL
1 Yaw damping (YAW DAMPER) - dutch roll damping - turn coordination (flaps extended and autopilot not engaged) - engine failure automatic compensation (flaps extended and autopilot engaged)		X	Both YAW DAMPER sides are normally engaged together and are both active
2 Pitch Trim (PITCH TRIM) - electric trim (without AP) - automatic trim (with AP) - static stability augmentation (MACH TRIM, SPEED TRIM, INCIDENCE TRIM)		X	X Both PITCH TRIM sides are engaged together. Side 1 only is active, Side 2 serves as back-up in the event of disconnection of Side 1.
3 Flight Envelope Protection - computation of airspeed lower limit : V min with respect to aircraft configuration - computation of airspeed upper limit : V max - computation of minimum airspeed - high angle-of-attack and windshear protection	X		X Recovery of a valid item of information is possible through disengagement of faulty FAC
4 Computation of various flap and slat maneuvering speeds and computation of instantaneous acceleration	X		ditto
5 Configuration data processing			X

(c)Thrust control computer system

The Thrust control computer system is designed in compliance with ARINC Characteristic 703.

This system comprises two identical computers associated with one control panel (TRP : Thrust Rating Panel) and one autothrottle actuator.

It provides the following functions :

- computation and display of the engine management parameter correponding to a flight phase
- display of the target thrust
  - computed by the FMC according to the flight plan
  - computed by the TCC : - for flexible takeoff
    - in limit thrust hold mode (TARGET is displayed)
- automatic thrust control (autothrottle function).

The following table details the functions and sub-system characteristics.

FUNCTIONS AND MODES	FAILURE BEHAVIOUR		
	FAIL-SOFT	FAIL-PASSIVE	FAIL-OPERATIONAL
Computation of engine management N1/EPR parameter upper limit		X	Processed by computers
Automatic thrust control (Auto-throttle function) - airspeed acquisition/hold function - Mach acquisition/hold function - max. thrust acquisition/hold function (acquisition/hold of computed N1/EPR limit) for following operational phases : . takeoff . climb . go around . alpha-floor protection . maximum continuous (single engine operation)	X By means of stops (inhibition to prevent advance or retard from specific throttle lever positions) : - low limit position (contingent on aircraft configuration) - high limit position	X (computation)	ditto
- acquisition/hold function of the target thrust computed by the FMC			

(d) AFS monitoring and test system (maintenance test panel)

Each AFS computer includes integrated test circuits.
A maintenance test panel, located on the flight compartment lateral panel is connected to the AFS computers.
The panel displays directly usable symbols or messages, data regarding L.R.U. or interface failures.
Furthermore the panel serves to interrogate AFS sub-systems, either for the purpose of obtaining complementary data or so as to determine LAND mode (autoland) availability.
The section regarding AFS maintenance aids conforms to ARINC Characteristics 423 and 701.

B. Integration into Aircraft
The auto flight system is built around computers located in left electronics rack 80VU.
The actuators are directly connected to the flight controls.
All the control and display units are grouped in the flight compartment, on the glareshield, overhead panel and lateral panel.
Computer interconnections, connections between computers and control units or instruments, between computers and sensors (peripherals) are mostly accomplished using ARINC Characteristic 429 digital information transfer system buses.

**ON A/C 001-100,

(Ref. Fig. 001)

**ON A/C 101-200,

(Ref. Fig. 002)

**ON A/C ALL

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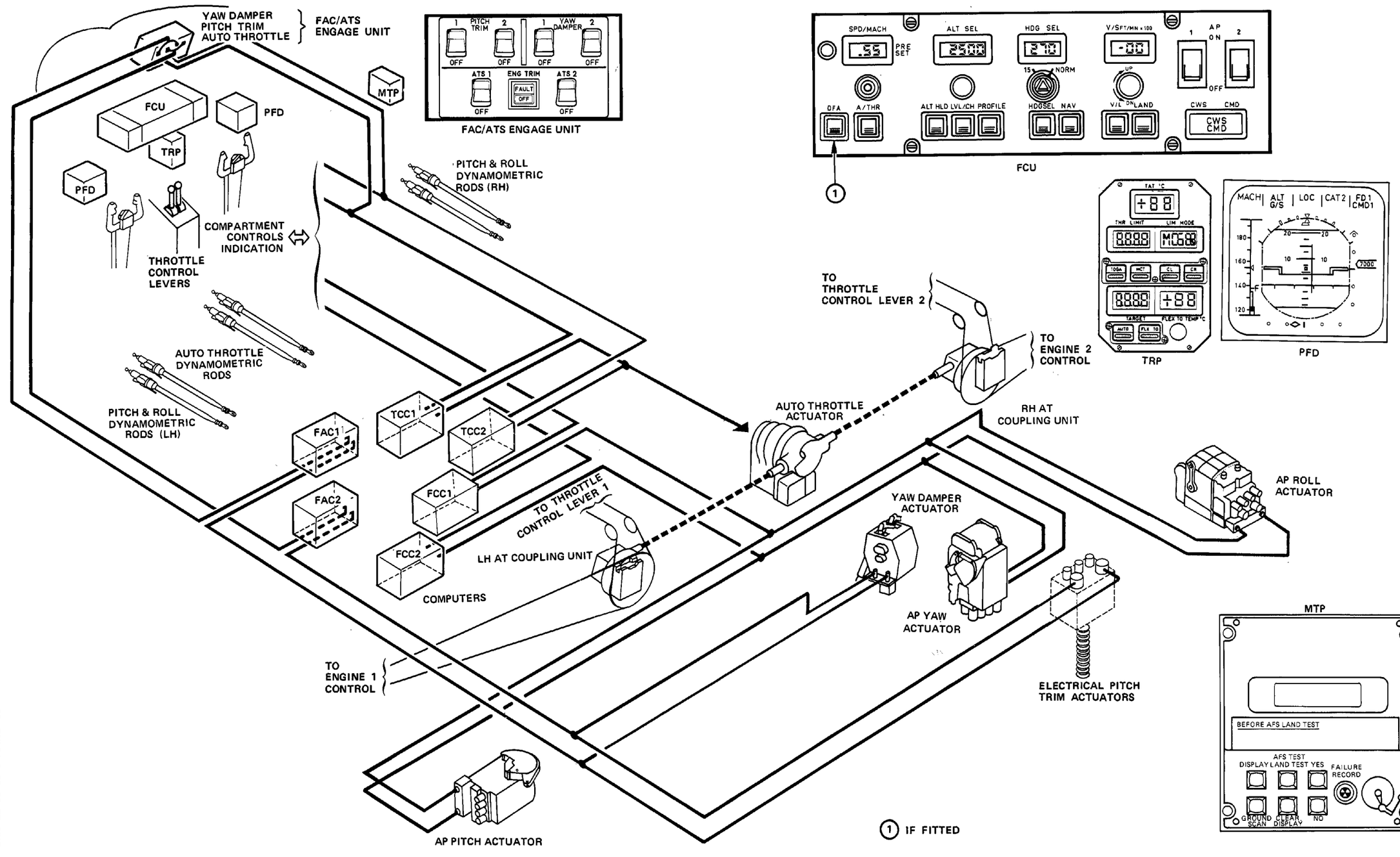
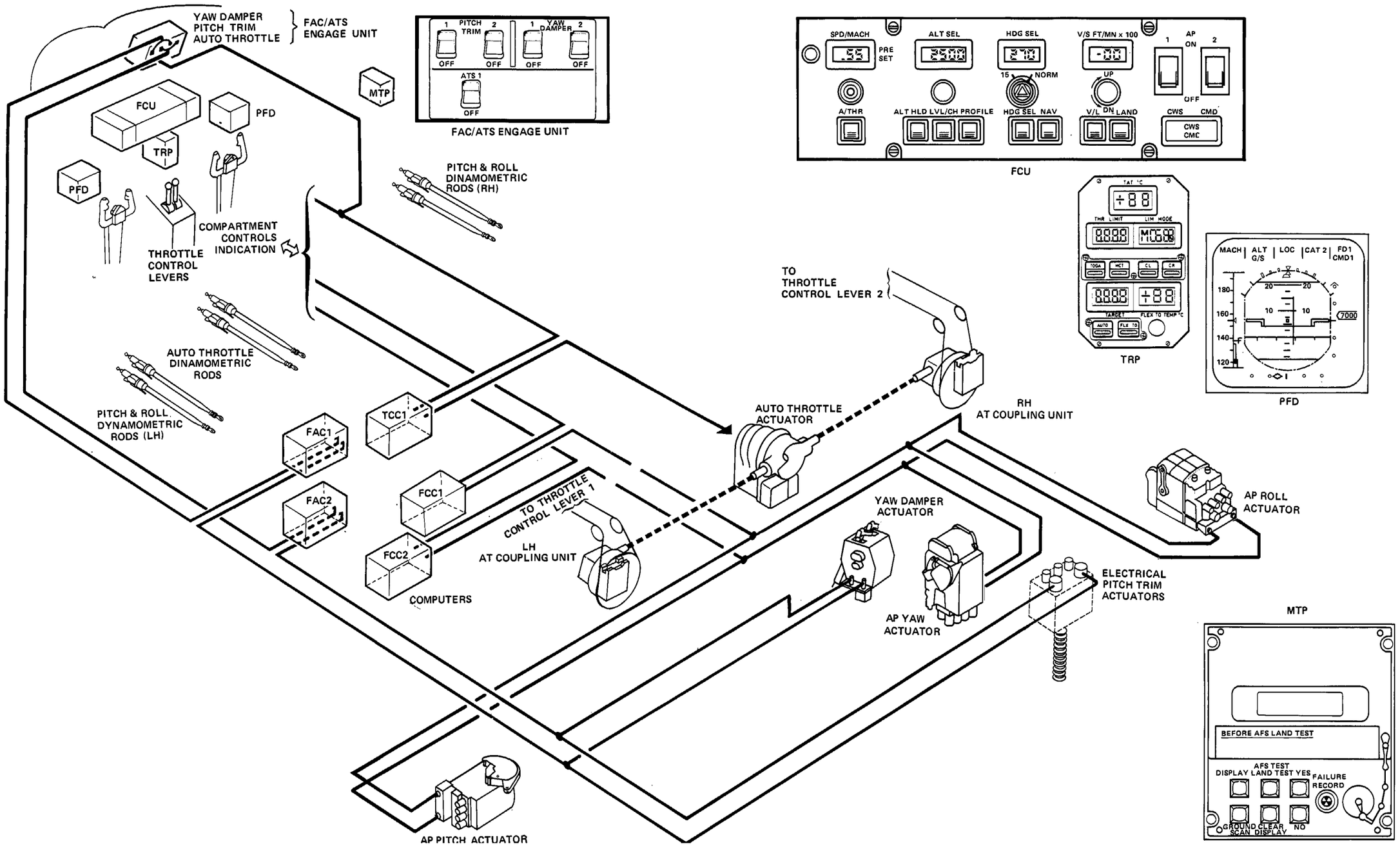


AFS - Component Layout
Figure 001

BM5 22 00 00 0 AAN2 - 00



AFS - Component Layout
Figure 002

BM5 22 00 00 0 FAM2 00

(1)Electrical power supply

Main supply of 28VDC is used to power the system so as to cope with switching and resulting transient interruptions.

Solely the flight augmentation computer system (FACS) and the thrust control computer system (TCCS) utilize 115VAC for powering the pitch trim actuators and the autothrottle actuator and power channel.

In addition the 115 V/400 Hz power supply is used to generate :

- the 5VAC used for the integral lighting
- the 4VDC and 8VDC required for the control unit display power supply.

The 26VAC power supply serves to excite the AFS sensors (synchros) and provides the computers with the reference required for demodulation of the synchro signals.

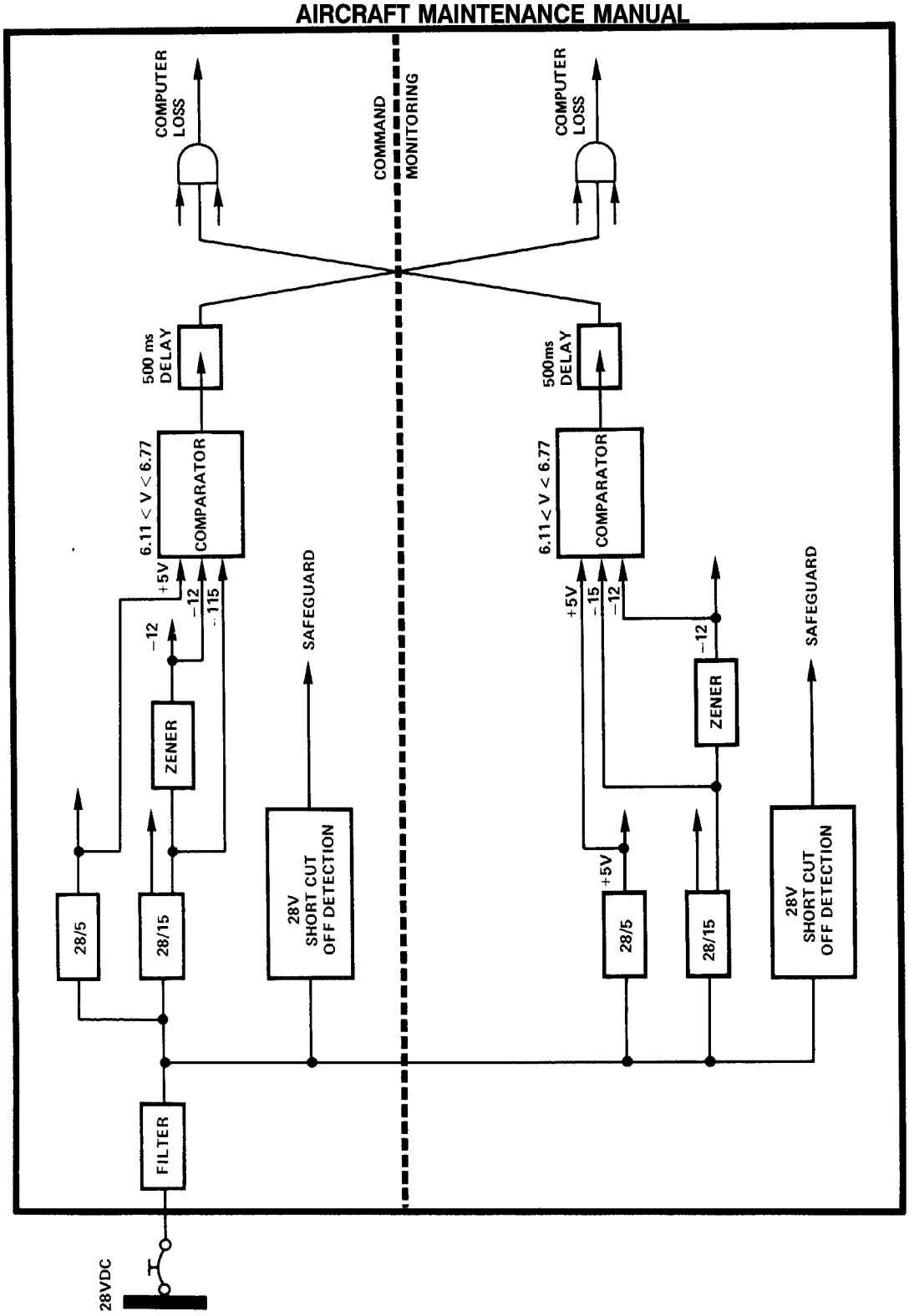
A 115VAC/5VAC step-down transformer provides the power used for placard (inscriptions) integral lighting.

The 8VDC and 4VDC power supplies are provided through a 3-phase transformer-rectifier. These power supplies power the TRP digital displays.

(a)Power supply blocks (Ref. Fig. 003)
Each computer is provided with two independent power supply blocks : one for the command channel, the other for the monitoring channel.
After filtering the 28 V supply is used for :
- generation of DC voltages required by the various electronic circuits and directly for :
- the engage lever power supply
- the clutches power supply.
The 28VDC through dc/dc converters generate the following voltages :
- +5 V for the logic circuits
- +15 V, -15 V for the analog circuits and engage logic
- -12 V from the -15 V for ARINC input or output circuits.
In addition, the power supply block generates, through independent converters :
- a +15 V ground reference for amplifier circuits
- +15 V and +5 V emergency voltages.
These voltages are used in the following circuits : RAM, failure detection and watchdog in order to safeguard the data in the event of main converter failures.
A short cutoff detection circuit is provided in order to safeguard the program in progress if necessary.
All the internal power supplies are monitored in the command as well as in the monitoring channels. To this end the -12 V, -15 V and +5 V power supplies are synthesized and the result is compared with a fixed algebraic value (threshold = 6.43 V).
In the event of disagreement (± 5 %), the computer is declared faulty. The result is crosschecked in the computer so that the heathly channel can detect the faulty one.

(b)Internal power supply automatic cut off and power supply restoration.
All the AFS computers and control units include internal +28VDC power supply automatic restoration and cut off circuits so as to optimize MTBF of all the components. In some cases, the FAC and the MTP ensure energization of the whole system to enable maintenance operations to be performed.
Moreover a safety test is initiated at first power rise in each energized component.

1 Automatic cut off
In normal operation, the internal +28VDC cut off occurs for each component when the three following conditions are met :
- aircraft on the ground
- engines shut down
- PITCH TRIM levers in OFF position.
If the computers and control units have been energized through one of the following tests :
- flight compartment lamp test
- MTP test (GROUND SCAN or AFS TEST/LAND TEST)
- safety test.
power supply is cut off at the end of these tests. In all the cases, AFS component cut off is delayed by 12.5 s. The MTP cut off is delayed by 20 s.
(Ref. Fig. 004)



BM5 22 00 00 0 AJM2 15 10
DM5

28V Monitoring Principle
Figure 003

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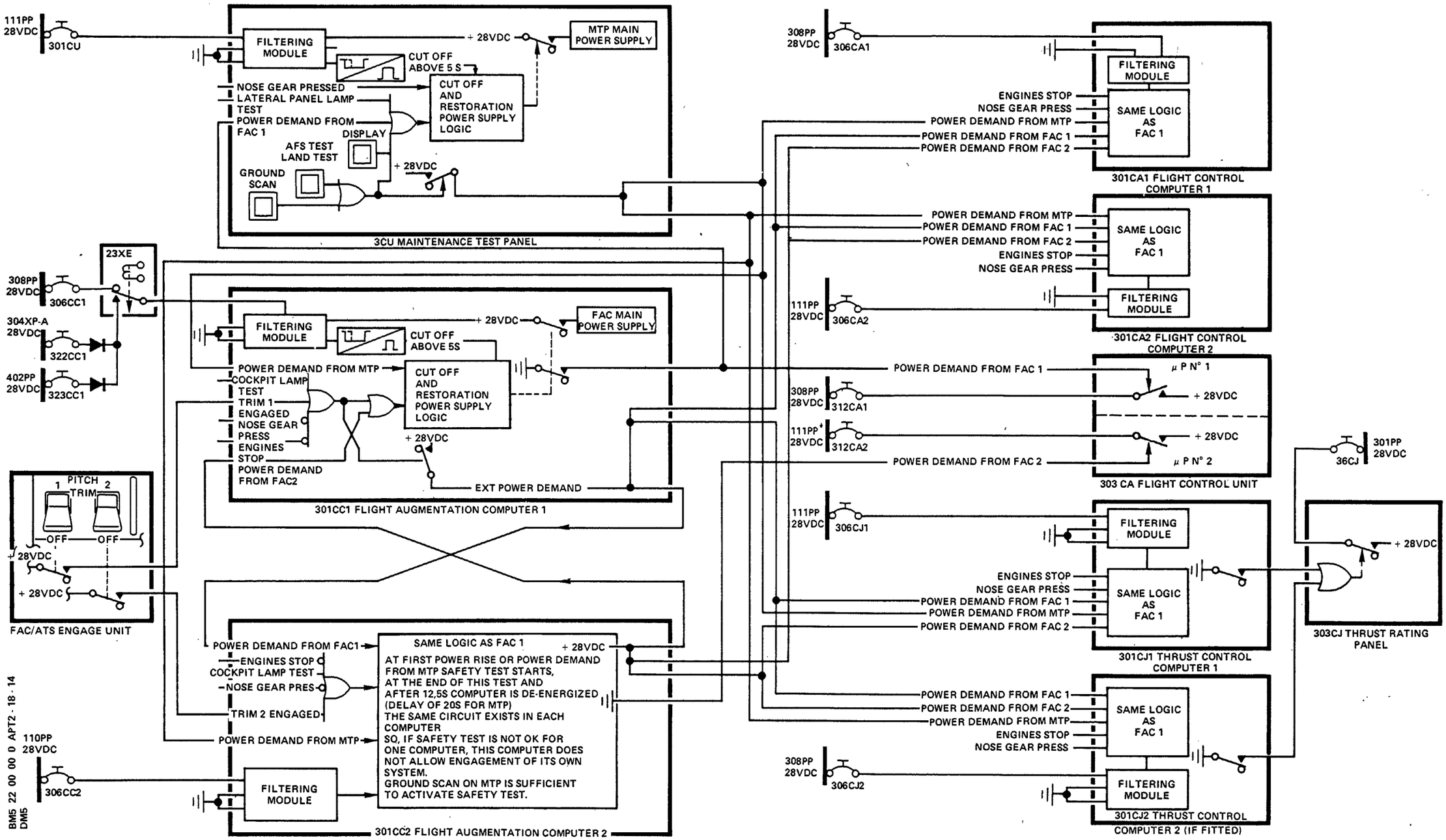


Power Supply Automatic Cut Off and Power Supply Restoration - Principle
Figure 004

2 Energization

At first power rise each computer starts its own safety test and remains energized throughout the test.

The computers (FAC, FCC, TCC) are energized independently if they receive one of the following items of information :

- aircraft in flight
- at least one engine running.

In that case, the MTP is energized with power by the TCC (TCC1 or TCC2 in dual TCC configurations).

The FCU is energized by the FAC1.

When the aircraft is on the ground with engines shut down, AFS energization is ensured by :

- the FAC in one of the following cases :
  . upon engagement of one PITCH TRIM lever
  . during flight compartment lamp test.
- the MTP when one of the following tests is performed :
  . GROUND SCAN
  . AFS TEST/LAND TEST.

It is to be noted that during GROUND SCAN tests, the MTP initiates its own test as well as other component safety tests.

The MTP is energized separately through the lamp test on the maintenance panel or through action on the DISPLAY pushbutton switch to enable fault message reading.

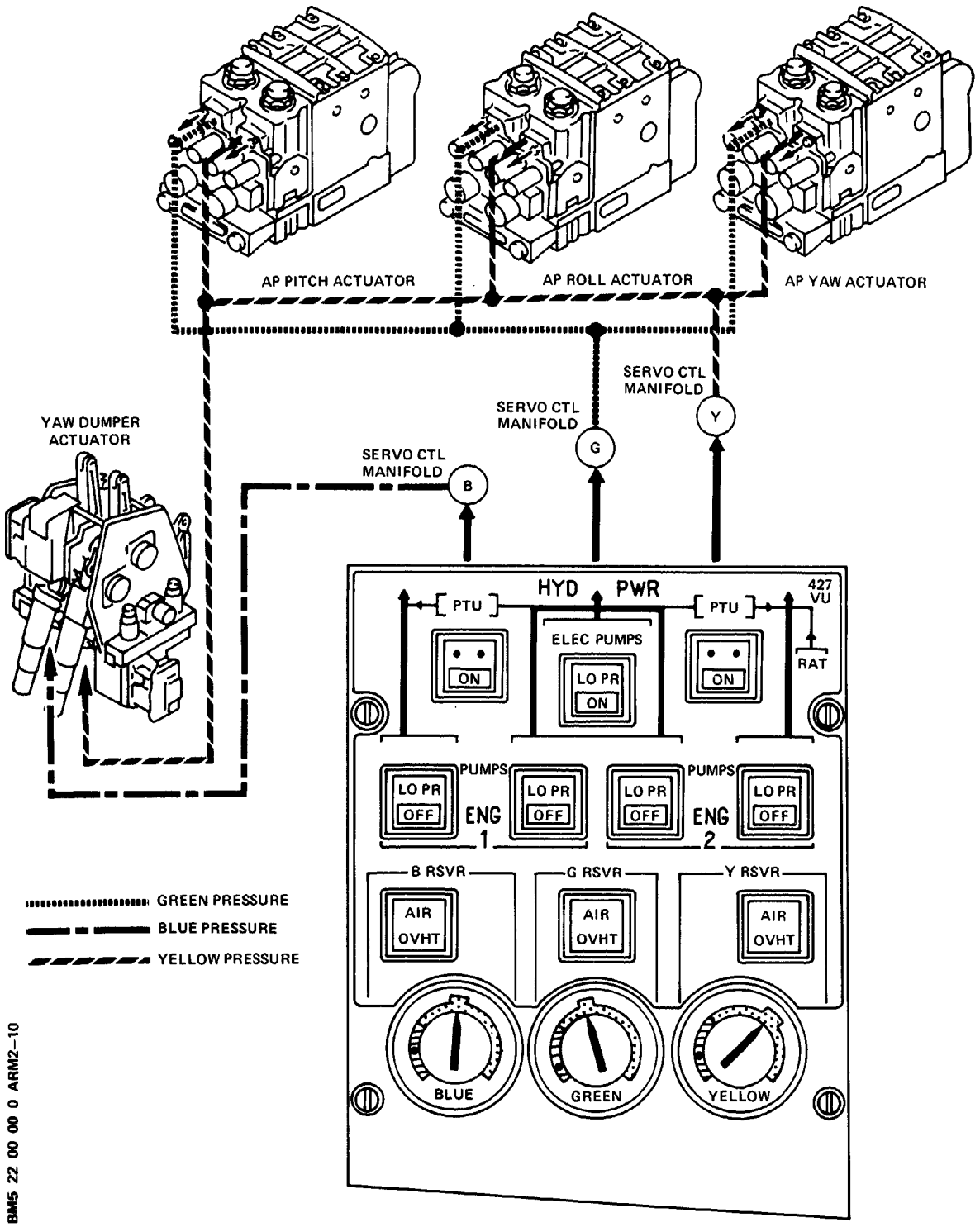
(2)Hydraulic power supply
The following figure shows the hydraulic power supplies of the various AFS actuators from the aircraft hydraulic system

(a)Blue hydraulic system
This system powers yaw damper actuator servomotor 1. The pressure switch which delivers the Blue system validity signal is installed directly on the servomotor.
The pressure switch provides a ground signal whenever the pressure setting of 100 bars (1450 psi) is exceeded.

(b)Green hydraulic system
This system powers autopilot pitch, roll and yaw actuator servomotors (side 1).
The pressure switch which delivers the Green system validity signal is Green reservoir.
The pressure switch provides a ground signal whenever the pressure setting of 100 bars (1450 psi) is exceeded.

(c)Yellow hydraulic system
This system powers :
- The autopilot pitch, roll and yaw actuator servomotors and yaw damper actuator servomotors (side 2).

It should be noted that the two servomotors associated with the yaw damper actuator are powered from separate hydraulic systems 1.e. Blue and yellow hydraulic systems.
In the event of engine failure, even in the case of a major failure (turbine disc bursting) the FACS function is maintained at all times through the remaining system.
The pressure switch which delivers the Yellow system validity signal is installed on the Yellow system HP manifold. a ground signal is applied whenever the pressure setting of 100 bars (1450 psi) is exceeded.
(Ref. Fig. 005)



AFS - Actuators Hydraulic Power Supply
Figure 005

(3)Landing gear data

The FAC is provided with landing gear discrete data which are monitored. After validation, the FACS transmits main landing gear data to the various AFS computers through the main data bus. In addition the AFS computers receive the nose landing gear data directly.

(a)Main landing gear data

Each FAC is connected to two relays (R and L gear), which are different for FAC1 or FAC2.

Information from these relays is transmitted by each FAC, in digital form, to the other AFS computers.

(b)Nose landing gear data

Information is transmitted to the AFS from two relays : one relay to which FAC1, FCC1, MTP and possibly TCC2 are connected in parallel; a second one which supplies FAC2, FCC2 and TCC1.

Since this information signal is used directly by the other computers, it is not re-transmitted.

(c)Relay power supply

There is a total segregation between AFS sub-systems 1 and 2 : relays associated with side 1 are fully independent from relays associated with side 2.

The FAC therefore receives 28VDC power for validating landing gear data and checking for their presence.

(d)Landing gear data monitoring

In each computer data are taken into account in the system engage logic :

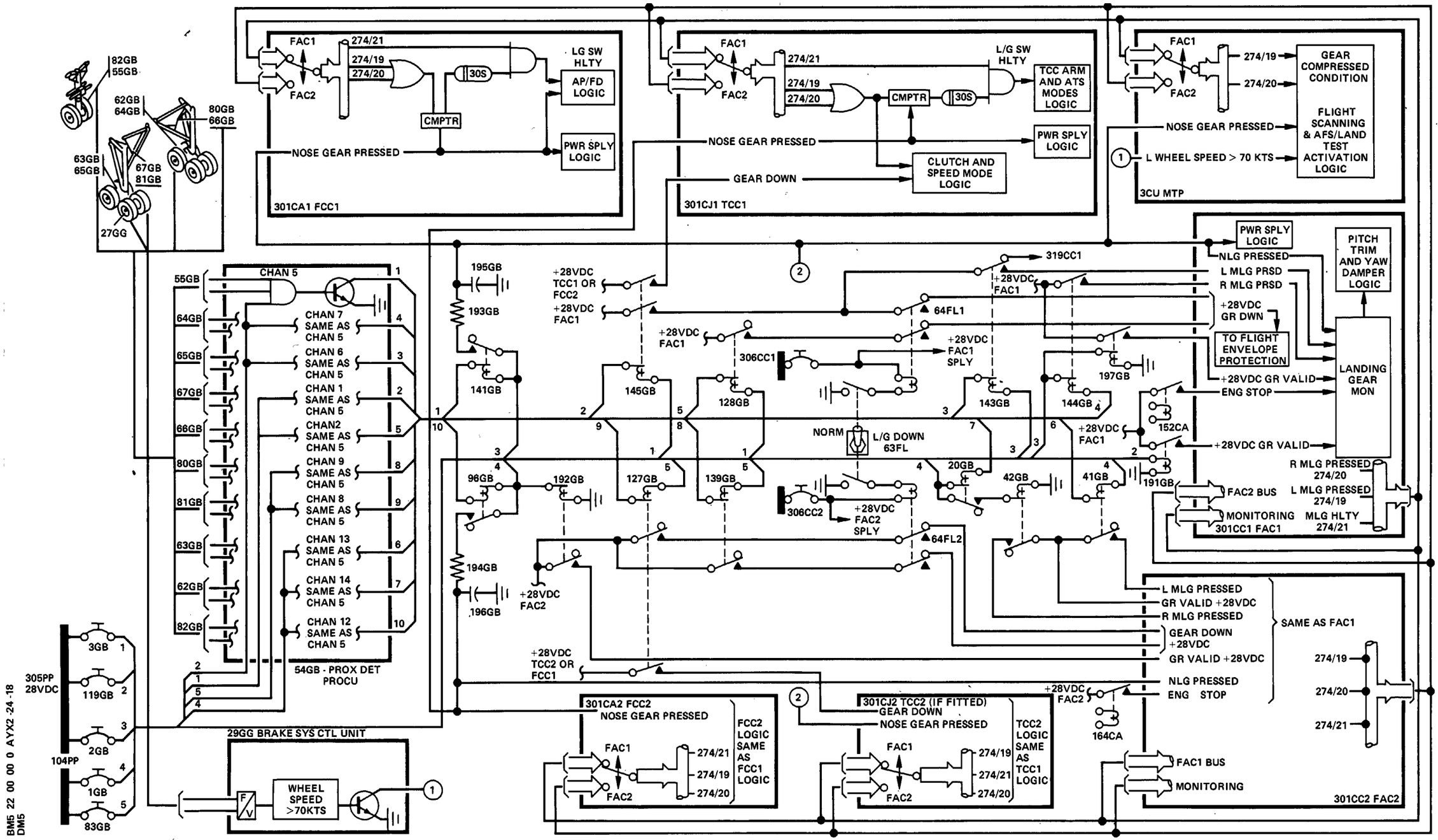
- main landing gear data validation
- conformity between L and R main landing gear
- conformity between main landing gear and nose gear.

NOTE : The other landing gear data are used in some computers and will be described in sub-chapter concerned.

(4)Wheel speed data

The MTP receives left wheel speed data from the brake system control unit which sends a ground signal, if the wheel speed is greater than 70 kts. Data is used for flight leg counting.

(Ref. Fig. 006)



AFS - Landing Gear Data Processing
Figure 006

EFFECTIVITY: ALL

R

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C. Control and Indicating

**ON A/C 101-200,

(Ref. Fig. 007)

**ON A/C 001-100,

(Ref. Fig. 008)

**ON A/C ALL

(Ref. Fig. 009)

(1)Flight Control Unit (FCU)
This unit transmits pilot(s) inputs and selections to the FCC and TCC in digital format (ARINC 429).
This unit includes :
. AP engage levers,
. mode pushbutton switches (except GO AROUND which is selected through the go around levers),
. selector knobs for flight parameters to be acquired and held (speed, Mach number, altitude, heading, vertical speed), and associated digital displays.

(2)EFIS control panels
The EFIS control panels are located on either side of the FCU and enable selection of displays on the CRT displays. Two complementary control panels (413VU and 412VU) located on the glareshield include a VOR/NAV/ILS switch connected with instrumentation and enables selection of the FD or flight path vector (FPV) bars.

(3)Thrust rating panel (TRP)
The TRP is located on the upper center instrument panel.
This unit transmits the flight phase selection made by the pilot (or the FMC in AUTO mode) to the TCC in digital format. In turn, the TRP displays the N1/EPR limit value corresponding to the operating TCC. It also displays :
- the N1/EPR target value determined by the operating FMC or TCC according to the ATS engaged modes.
- the flight phase given by the operating FMC (in AUTO mode).
- the total air temperature (TAT).
It also includes a selector knob with associated display for selection and display of the flexible takeoff fictitious temperature.

(4)ILS control box
The ILS control box is located on the center pedestal. It includes the ILS frequency and runway heading (QFU) selection knobs as well as associated digital displays.

(5)FAC/ATS engage unit
The FAC/ATS engage unit includes PITCH TRIM, YAW DAMPER and ATS arming levers.

(6)FMS control and display units (CDU)
The FMS control and display units are located on the center pedestal. They include an alpha-numeric keyboard and a CRT display. They are used

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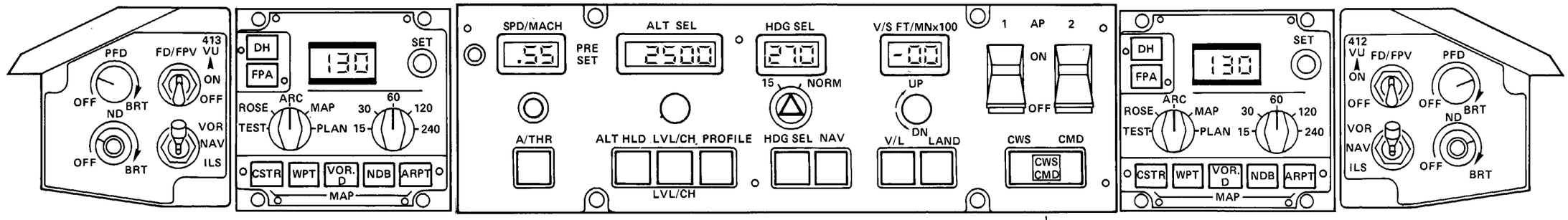

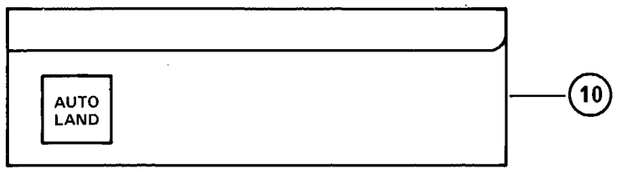

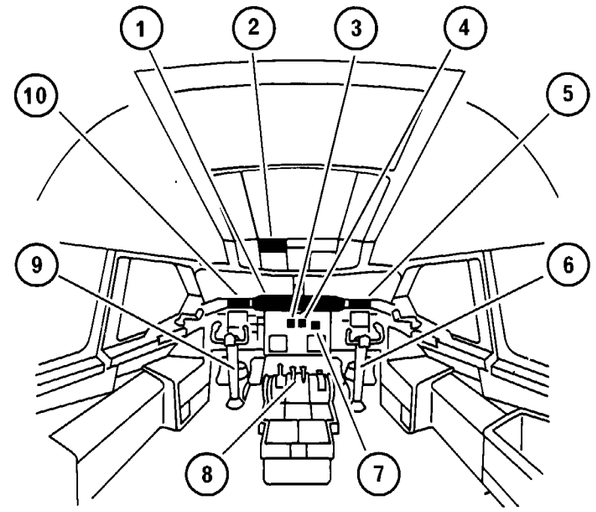

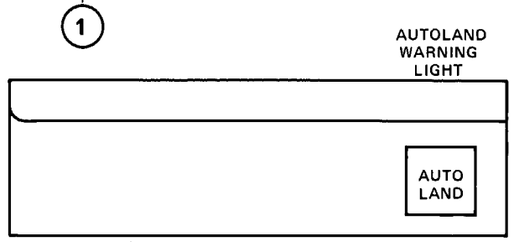

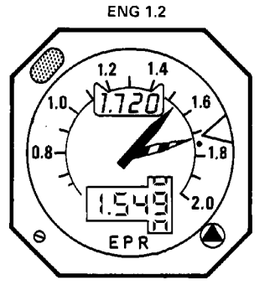

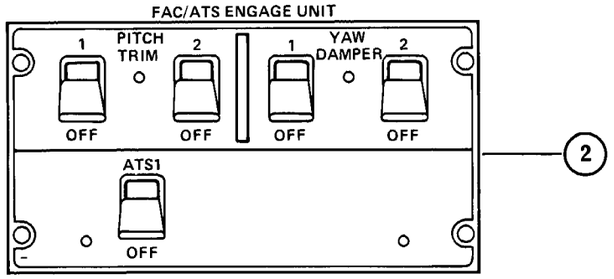

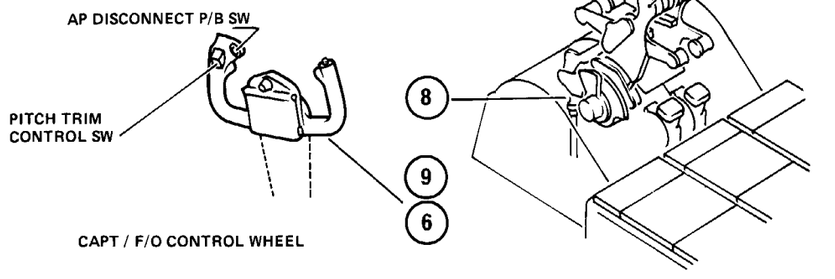

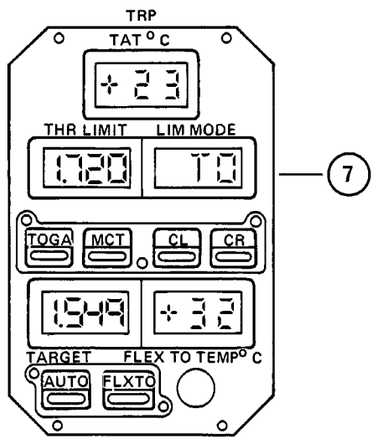

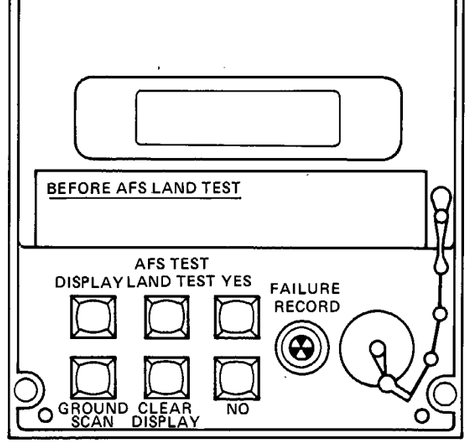

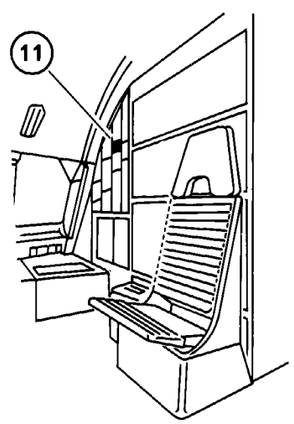

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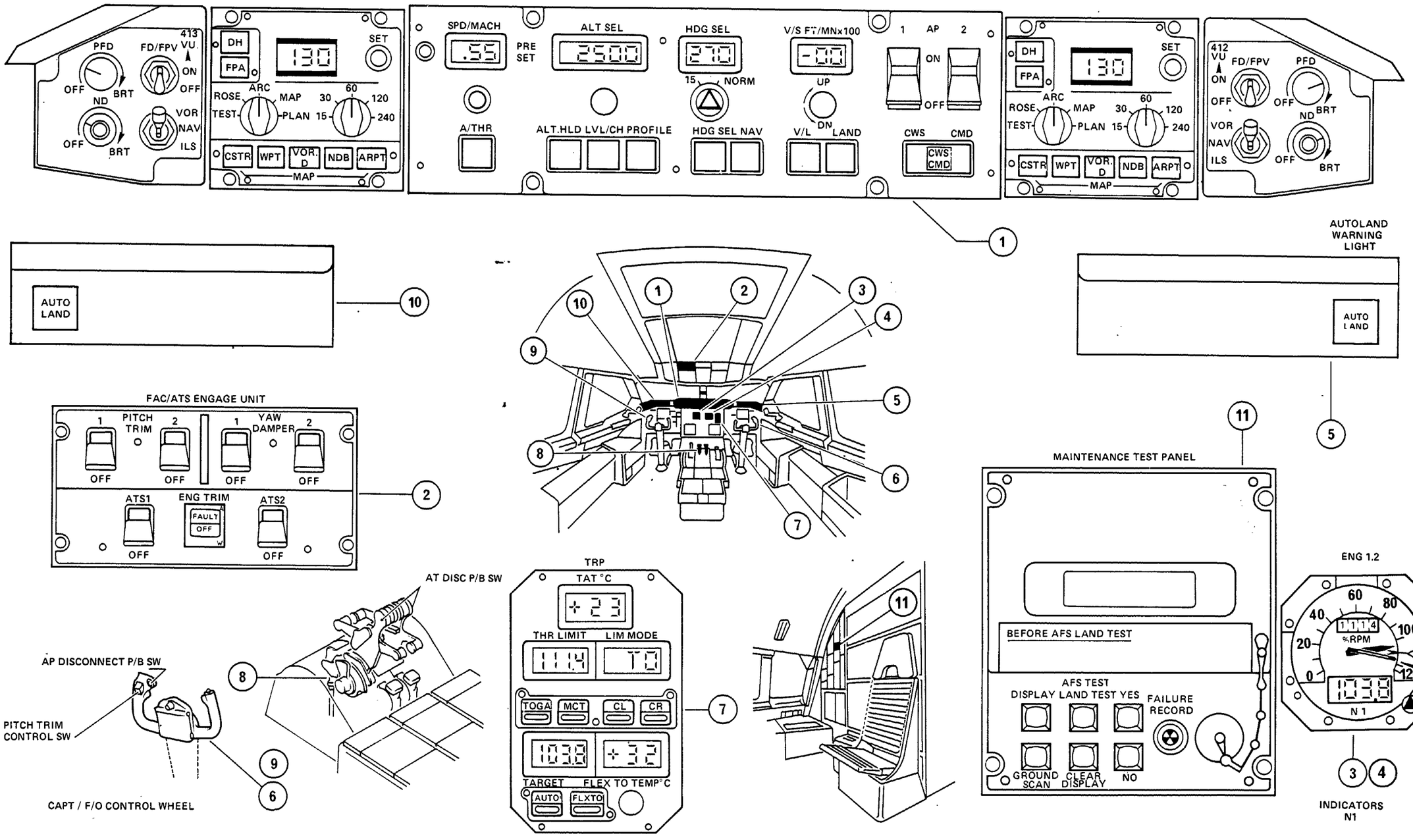
AFS Control and Indicating in Flight Compartment
(Sheet 1/2)
Figure 007



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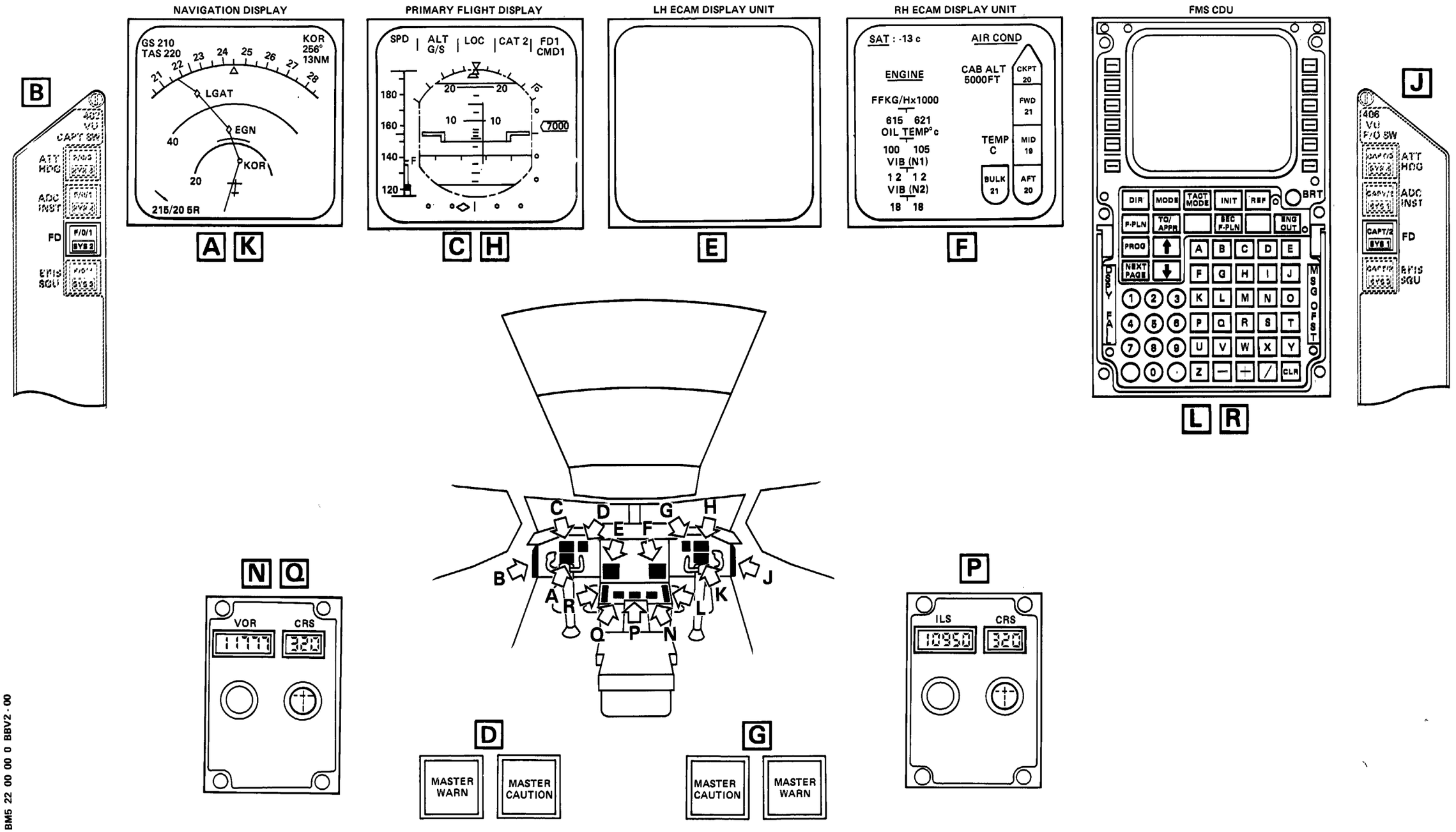
AFS Control and Indicating in Flight Compartment
(Sheet 1/2)
Figure 008



AFS Control and Indicating in Flight Compartment
(Sheet 2/2)
Figure 009

as interfaces between the pilots and the FMCs for introduction of the flight plan data, imposed constraints and selection of various functions, etc.

(7)Primary flight displays (PFD)

These are CRT displays which provide the following indications :

- attitude (pitch and roll) and heading information.
- AP/FD/autothrottle engagement status and modes; landing capabilities.
- airspeed given on a scale comprising :
  computed airspeed transmitted by the ADC and its derivative, airspeed selected through the FCU, maximum operating airspeed, minimum selectable airspeed, minimum airspeed and maneuvering airspeed (extension or retraction of surfaces indicated by a letter : F, S...).
- FD bars (roll, pitch and yaw) controlled by FCC1 and FCC2. Normally FD1 supplies PFD1 and FD2 supplies PFD2. Each pilot can select the commands generated by the opposite FCC on his PFD by means of the FD bars switching pushbutton switch (F/O/1 - SYS2 and CAPT/2 - SYS1).
  In addition each pilot can remove the FD bars and display the FPV by means of the FD/FPV selector switch located near the EFIS control panel.
  Finally a red FD1/2 indication comes on in the event of a flight director failure.

(8)Navigation displays (ND)

These are CRT displays. Displays such as rose, map modes... are selected through the EFIS control panel.
The selected heading index (heading set on the FCU) is represented by a triangle or by a digital indication if the selected heading is outside the scale values.

(9)Engine N1/EPR indicators

These are digital instruments with ARINC Specification 429 digital inputs.
Each indicator includes :

- a yellow triangular bug which repeats the N1/EPR limit value computed by the TCC (automatic mode) or the N1/EPR limit value selected and manually set by the pilot through the manual display knob (manual mode).

The N1/EPR limit value selected manually also appears in the upper digital display.

- a white pointer indicating the N1/EPR actual value which also appears in the lower digital display.
- a black and white striped pointer indicating the N1/EPR command value.

The last two indications are given by the EEC/PMC.

D. Connection with Flight Controls

The AFS is connected to the flight controls through a mechanical linkage and through actuators.

(1)Pitch axis

- the electrical pitch trim actuators operate in parallel with the trim control wheels, the control valves of the hydraulic motors driving the THS actuator screwjack which actuates the trimmable horizontal stabilizer. The electrical pitch trim actuators are located at the top of the THS actuator.
- The two AP pitch hydraulic servomotors are contained in a single casing with a single output shaft which is connected to the pitch control linkage through a torque limiter lever.
  These servomotors receive electrical signals from the FCCs.
  The servomotor servovalves transform these signals into hydraulic command signals which displace the output shaft and associated control linkage.
  Deflection of the control surface controlled by the AP is limited in amplitude and speed.
  In addition, override of the AP commands is always possible by the pilot through the normal piloting controls.
  The override is performed by causing the torque limiter lever to operate : the AP remains engaged, therefore when the pilot releases his load on the controls, the AP takes over control of the aircraft with new flight parameters applied.
  The principle outlined above also applies to the roll and yaw axes.

(2)Roll axis (Ref. Fig. 011)

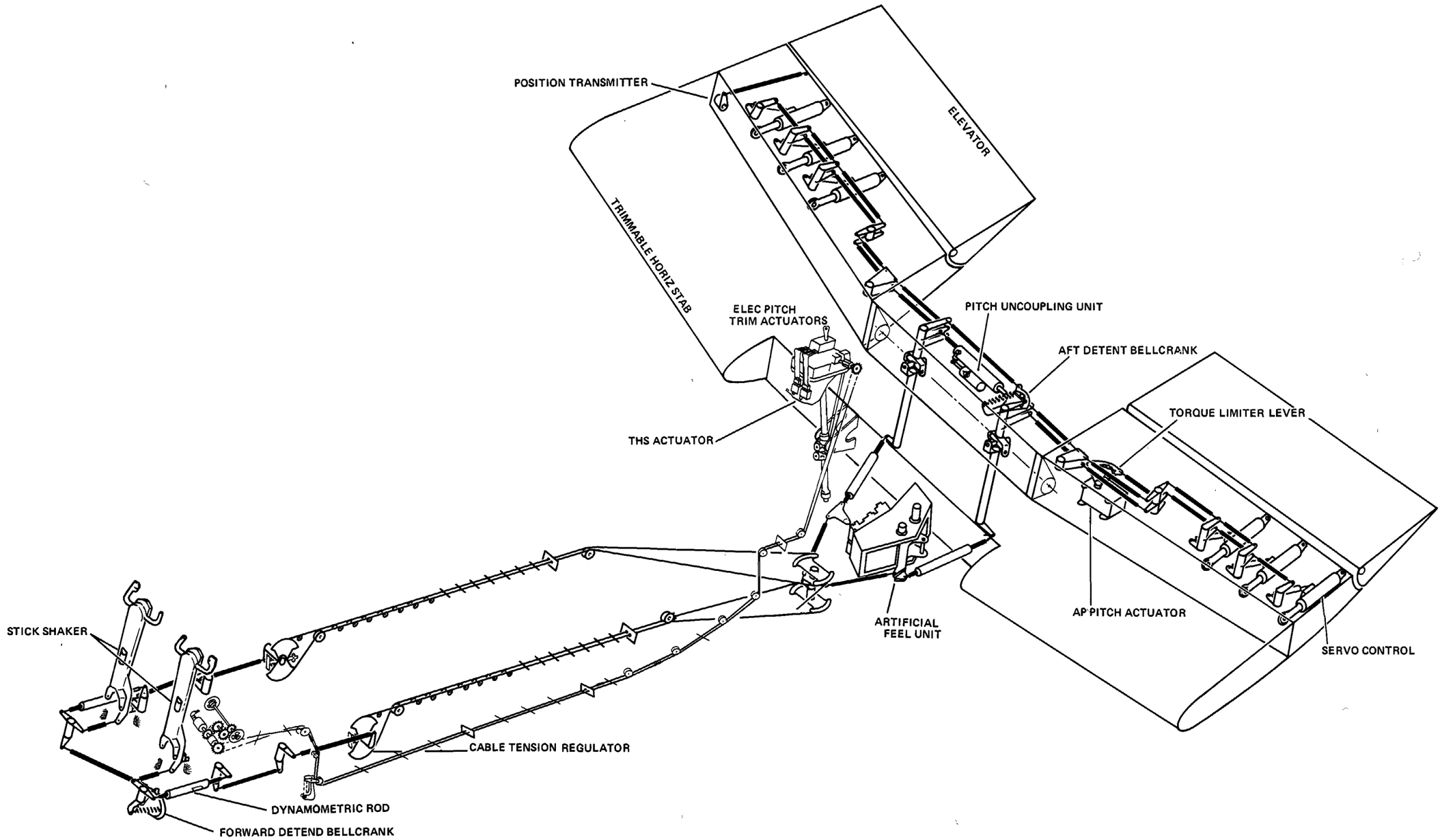
The AP roll actuator, which is identical to the AP pitch actuator, actuates the roll control linkage at the R wing differential and droop unit through a torque limiter lever.

(3)Yaw axis (Ref. Fig. 012)

The AP yaw actuator, which is identical to the pitch and roll axis units, actuates through the torque limiter lever, the differential mechanism which combines the following outputs to the rudder :

- yaw damper command,
- pilot input (through the pedals)
- or AP yaw command.

The AP yaw actuator directly actuates the rudder control linkage and subsequently the pedals.
The yaw damper actuator controls the rudder through the differential mechanism without displacing the pedals. The yaw damper actuator differs from the AP yaw actuator.
Moreover, the resultant output from the differential mechanism is modulated by the variable stop lever whose role is to reduce the rudder deflection when the C.A.S. increases.
(Ref. Fig. 010)



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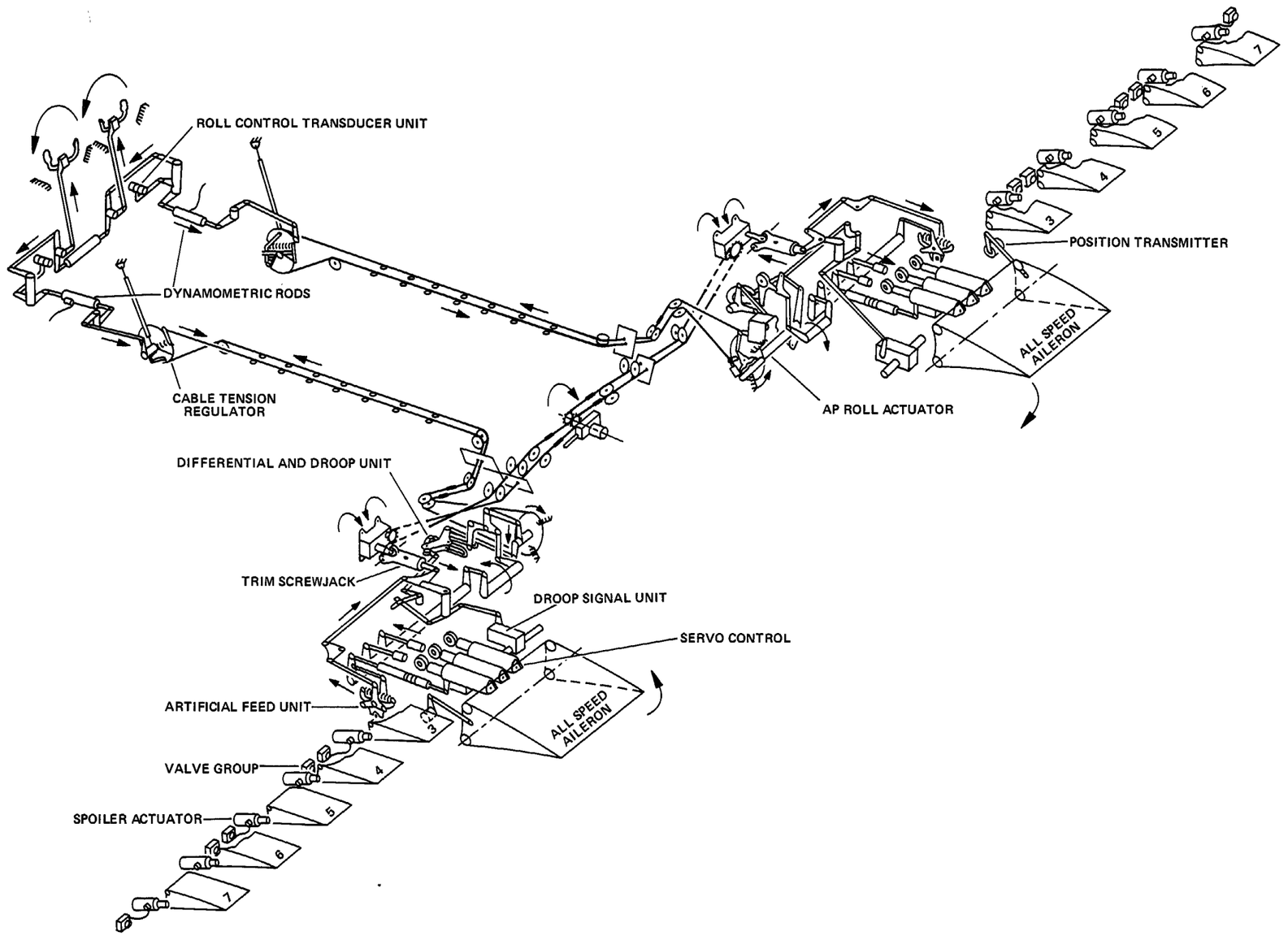
AFS - Pitch Control Channel Layout
Figure 010



AFS - Roll Control Channel Layout
Figure 011

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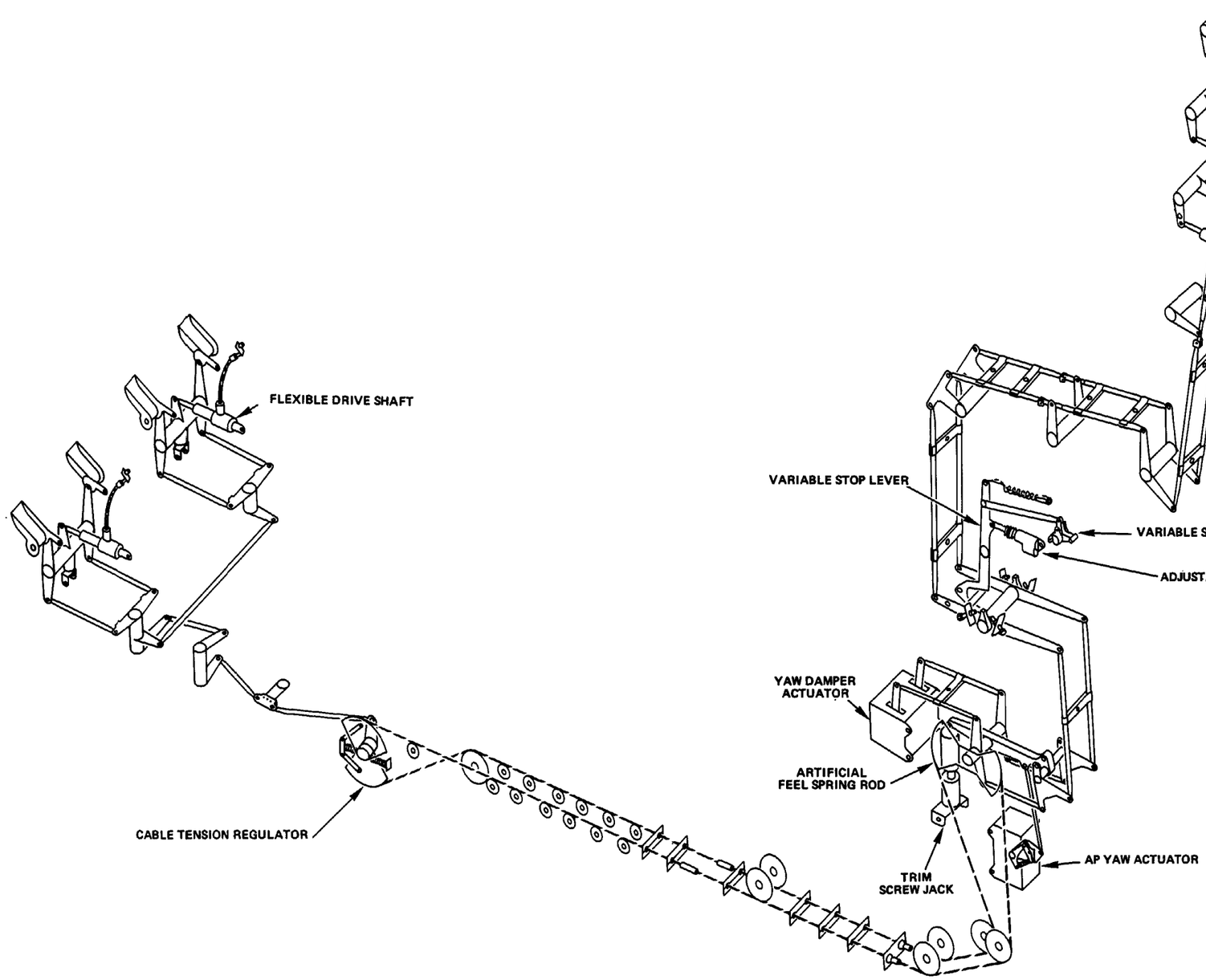

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AFS - Yaw Control Channel Layout
Figure 012

**ON A/C 001-100,

E. Connection with Engine Controls
The single electric motor for the autothrottle system actuates two engine control linkages through the two coupling units each including an electro-magnetic clutch and a safety friction clutch. The motor responds to inputs from one TCC.
Both throttles are operated simultaneously.
Nevertheless, at any given moment, each pilot can override the automatic operation separately for each engine, by simply applying a light load on the appropriate throttle control lever. Load sensing is performed by a dynamometric rod.
Throughout the period of load application, the associated coupling unit disconnects, thus enabling the pilot to adjust manually the throttle control lever.
As soon as the pilot releases the load, the autothrottle takes over control of the engine and maintains any possible difference between the two throttle settings.

**ON A/C 101-200,

E. Connection with Engine Controls
The single electric motor for the autothrottle system actuates two coupling units each including an electro-magnetic clutch and a safety friction clutch. The motor responds to inputs from one TCC. The throttle control lever positions are transmitted to the engine electronic control (EEC) through synchro resolver unit.
Both throttles are operated simultaneously.
Nevertheless, at any given moment, each pilot can override the automatic operation separately for each engine, by simply applying a light load on the appropriate throttle control lever. Load sensing is performed by a dynamometric rod.
Throughout the period of load application, the associated coupling unit disconnects, thus enabling the pilot to adjust manually the throttle control lever.
As soon as the pilot releases the load, the autothrottle takes over control of the engine and maintains any possible difference between the two throttle settings.

**ON A/C 001-100,

(Ref. Fig. 013)

**ON A/C 101-200,

(Ref. Fig. 014)

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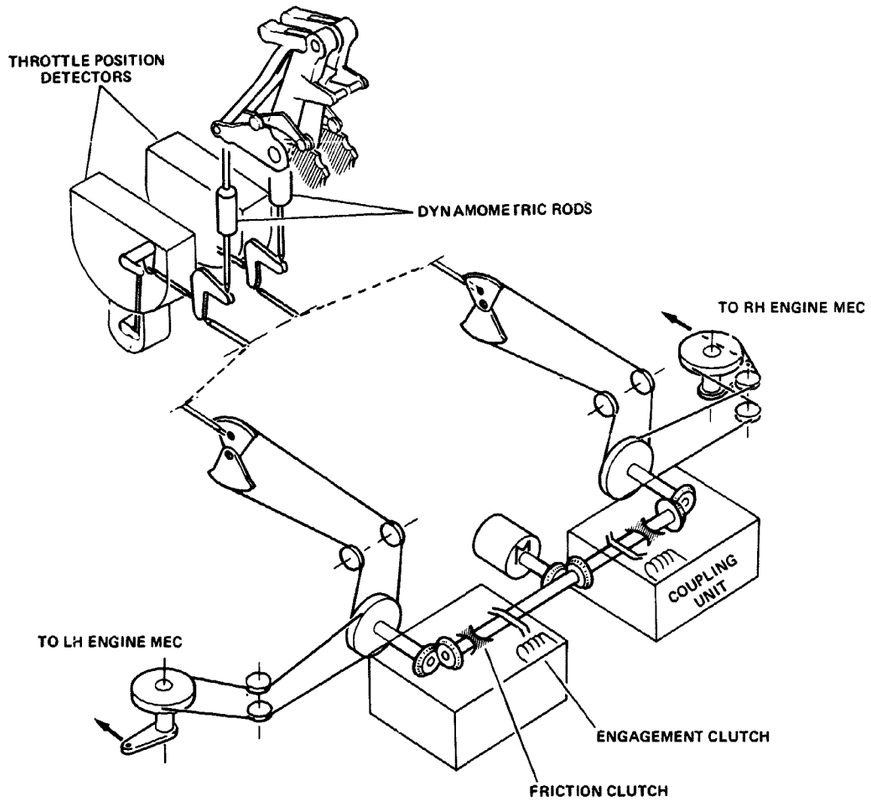

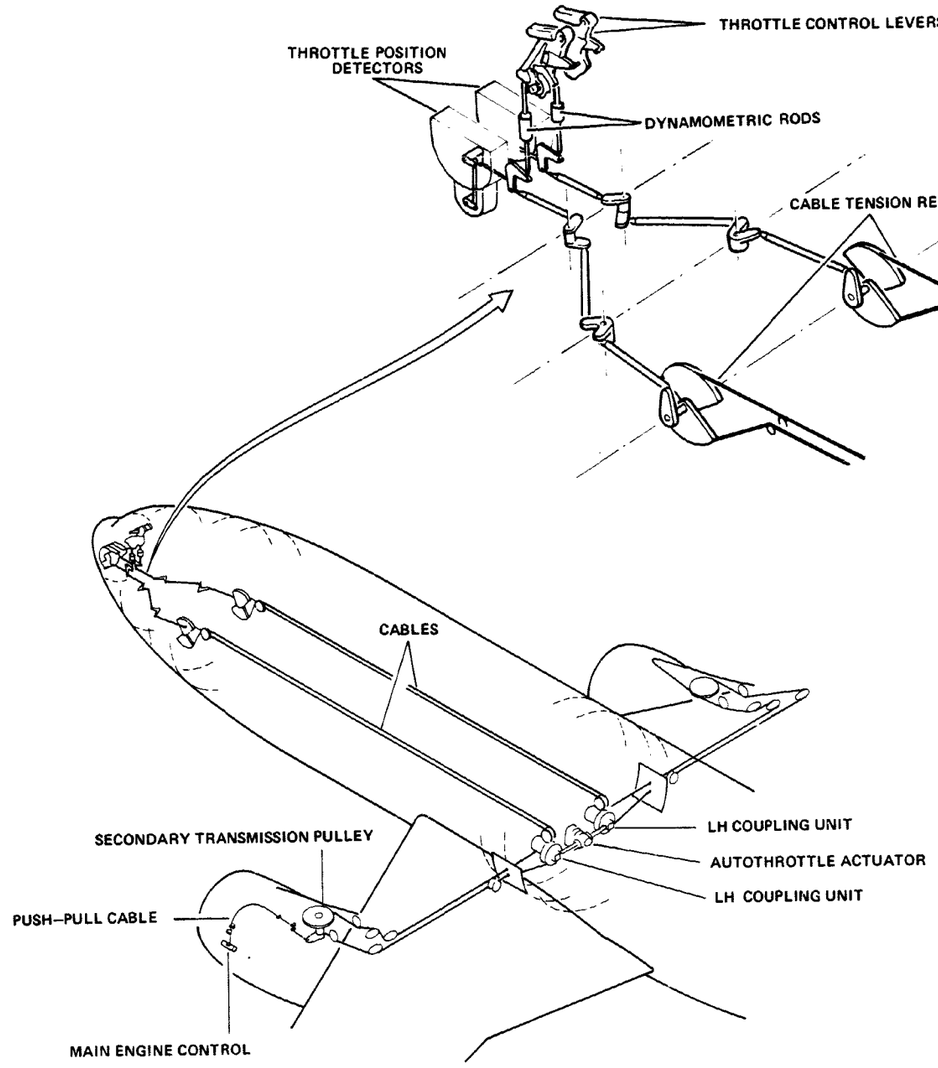

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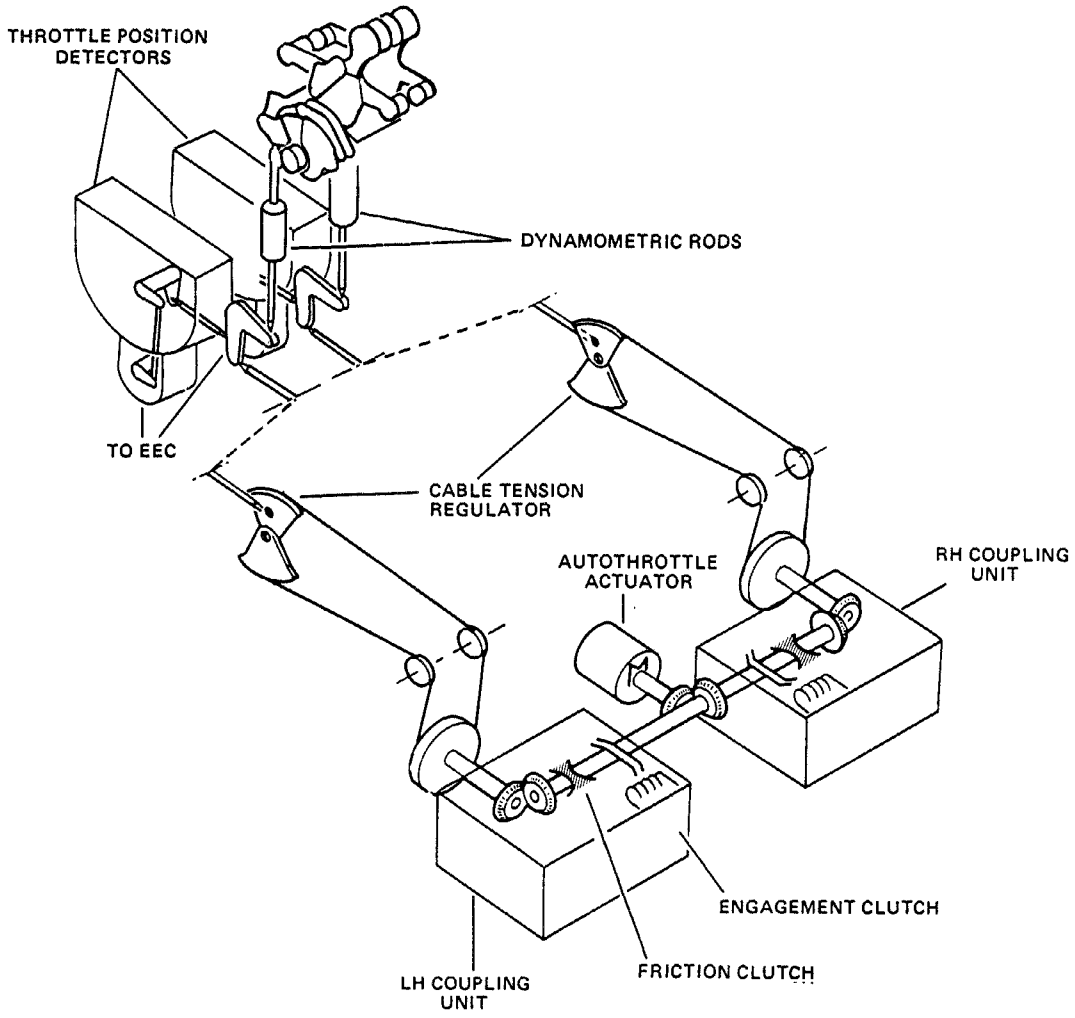
AFS - Engine Control Channel Layout
Figure 013



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AFS - Engine Control Channel Layout
Figure 014

**ON A/C ALL

F. Computer Architecture

(1)Introduction

The AFS design is based on digital technology configured on INTEL series microprocessors :

- 16-bit 8086 microprocessors for the FACs, FCCs and the TCCs,
- 8-bit 8085 microprocessors for the various control and display units (FCU, TRP, MTP).

However, certain sections are configured in analog form : such is the case with : power channels, voting device between the surface control signals and the engage and clutching logic.

The computers (FCC, TCC and FAC) utilize a bi-processor structure, the other components (FCU, MTP and TRP) a mono-processor structure.

However it is to be noted that the FCU comprises two computation channels but only one computation channel is used at a time. Channel 1 has priority, channel 2 is used as a cold back-up (circuits not supplied) and will be used only after a detected failure of channel 1.

(2)FCC, FAC and TCC computation channel design (Ref. Fig. 015)

All the computers comprise two computation channels consisting of a digital part designed around INTEL 8086 microprocessor and an analog part.

(a)Digital part

The two computation channels are also separated as far as possible. In addition, each computation channel uses its own clock, the two clocks are not synchronized.

Some exchange is however necessary so that certain tasks can be synchronized or in view of the fact that some computations are performed by one of the microprocessors only whilst the result must be used by both channels (such cases are explained in the description of each system).

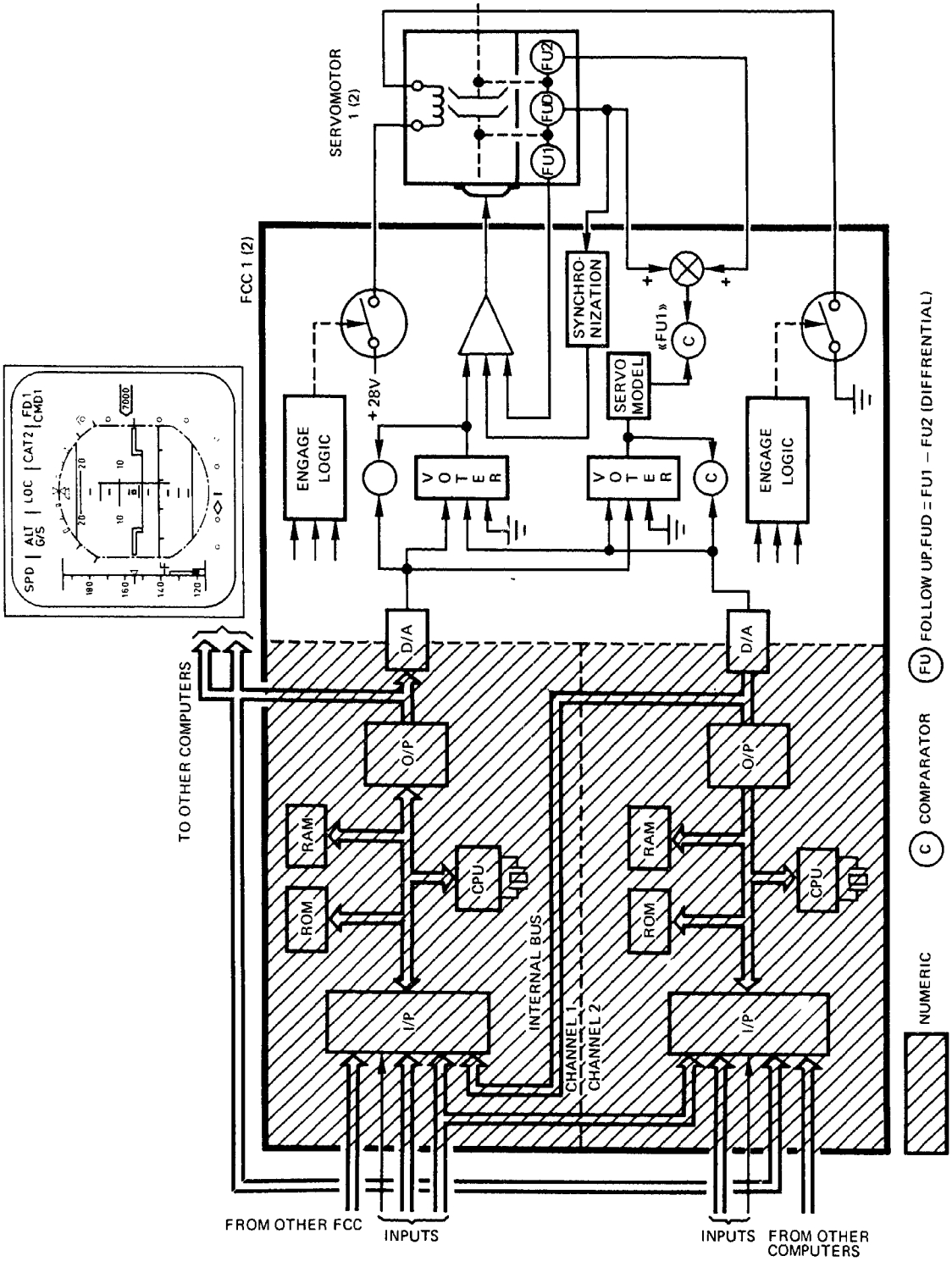
Transmission is then accomplished through two specific data buses in such a way that the output from one microprocessor is the input for the other microprocessor; one of those buses is the main computer output bus (exchange from the command processor to the monitoring processor) the other bus is the computer internal bus (exchange from the monitoring processor to the command processor).

For certain specific functions, it is possible to inhibit exchange between processors.

Each computation channel generates a control command (surface deflection, throttle position...) which is validated, after D/A conversion, either by direct comparison (pitch trim) or by voter circuits (AP yaw damper or autothrottle).

After validation the control command is performed by the power loop which is in analog form. Power loop monitoring is accomplished in analog form except for TCC where the power loop monitoring is performed in software form by both command and monitor processors.

Finally the engage logic is accomplished in hard-wired logic, the mode logic is programmed in the command and/or monitoring processors.



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AFS - Computer Organization
Figure 015

(b)Analog part

Each FCC, TCC and FAC includes an analog part which consists of :

- the power loops
- the power loop monitoring (except for TCC)
- the output voters
- the output comparators (except for TCC)
- the hard-wired engage logic.

These boards are specific to each computer and will be described in the systems corresponding to the FCCs, TCCs and FACs.

However we give the voter operating principle (circuit existing in all the computers) and, as an example, the monitoring principle of the AP power loop.

The voter is a circuit accomplished in analog form.

It ensures selection between several inputs which are :

- the command computed by the command processor (after D/A conversion)
- the command computed by the monitoring processor (after D/A conversion)
- a zero value command since the system works around a zero value most of the time.

The two computation commands (channel 1 and channel 2) are fed to both voter circuits which are composed (for one voter circuit) of : three open-loop input amplifiers which receive the two computation commands and the zero reference.

In fact these amplifiers serve as level detectors between their own input and the system output which is controlled by an output amplifier.

This common output must be identical to inputs.

As the output voltage at each input detector varies to make the common output identical to the inputs by means of counter reactions, a single detector can control the output at any given time as the two other detectors reach saturation level.

It is established that the detector which controls the output is that with the smaller difference between its own input and the common output, therefore, that with an input value which is the mid value of the three inputs (two computation channels and zero reference).

This system therefore provides passivation of obviously wrong deflection commands (hardover type).

Use of identical voter circuits ensures that the same output value is selected which enables the power loop to be monitored. Besides this passivation function, the voter circuit includes safety comparators.

These comparators (one per channel) compare the signal computed by a channel with the signal finally voted.

If the deviation exceeds a threshold set to take into account the possible discrepancies between detectors and tolerances in the electronic circuits, a computer disconnect logic command is generated.

This disconnection occurs at the end of a delay chosen to eliminate transient failures, which are nevertheless passivated.

(c)Power loop monitoring circuit

Monitoring is accomplished from voted values (for the TCC, FCC and yaw damper) in analog form for FCC and yaw damper or from values validated by the output comparators (pitch trim). As an example, figure shows the AP power loop monitoring principle. The command from the monitoring channel voter is compared with position feedbacks FUD + FU2 = FU1.

FU1 = servomotor output upstream of the clutch
FU2 = servomotor output downstream of the clutch
FUD = difference between FU1 - FU2 outputs.

The FU1 signal thus recreated by the sum FUD + FU2 represents the output of the command voter carried out by the servomotor; the output from the monitoring voter is multiplied by a transfer function known as the model (which simulates the servomotor transfer) so that if the power loop slaving is correctly achieved (i.e. without failure) the commands at the power loop comparator input will be identical. Any discrepancy detected by the power loop comparator will cause AP disconnection, the comparator status being processed directly by the engage logic.

With the other systems apart from the AP, the same principle is adhered to, taking into account the specific characteristics of each one (for example, feedback in speed and not in position for the autothrottle).

(3)Design of control and display units

These units (FCU, TRP and MTP) are built around a single microprocessor (INTEL 8085). The FCU and MTP central unit boards are identical. Since each unit has its own architecture a description will be given in the systems corresponding to the FCCs, TCCs and MTP.

(4)Languages used

The program of the various digital units consists of instructions stored in the read-only-memory associated with each CPU. In the AFS two programming languages are used :

- assembler language which has the characteristic of being sufficiently matched to the internal design of the microprocessor; it therefore serves for optimization of the program with respect to computing time and memory capacity.
- PL/M (INTEL language) which is less dependent on the microprocessor internal structure and therefore closer to human language.

These two languages are used in such a way as to intentionally introduce a software dissymmetry whenever necessary.

The MTP, FCU and TRP are programmed both in assembler language and in PL/M 85, selection of the language has been made according to the type of the software module to be programmed.

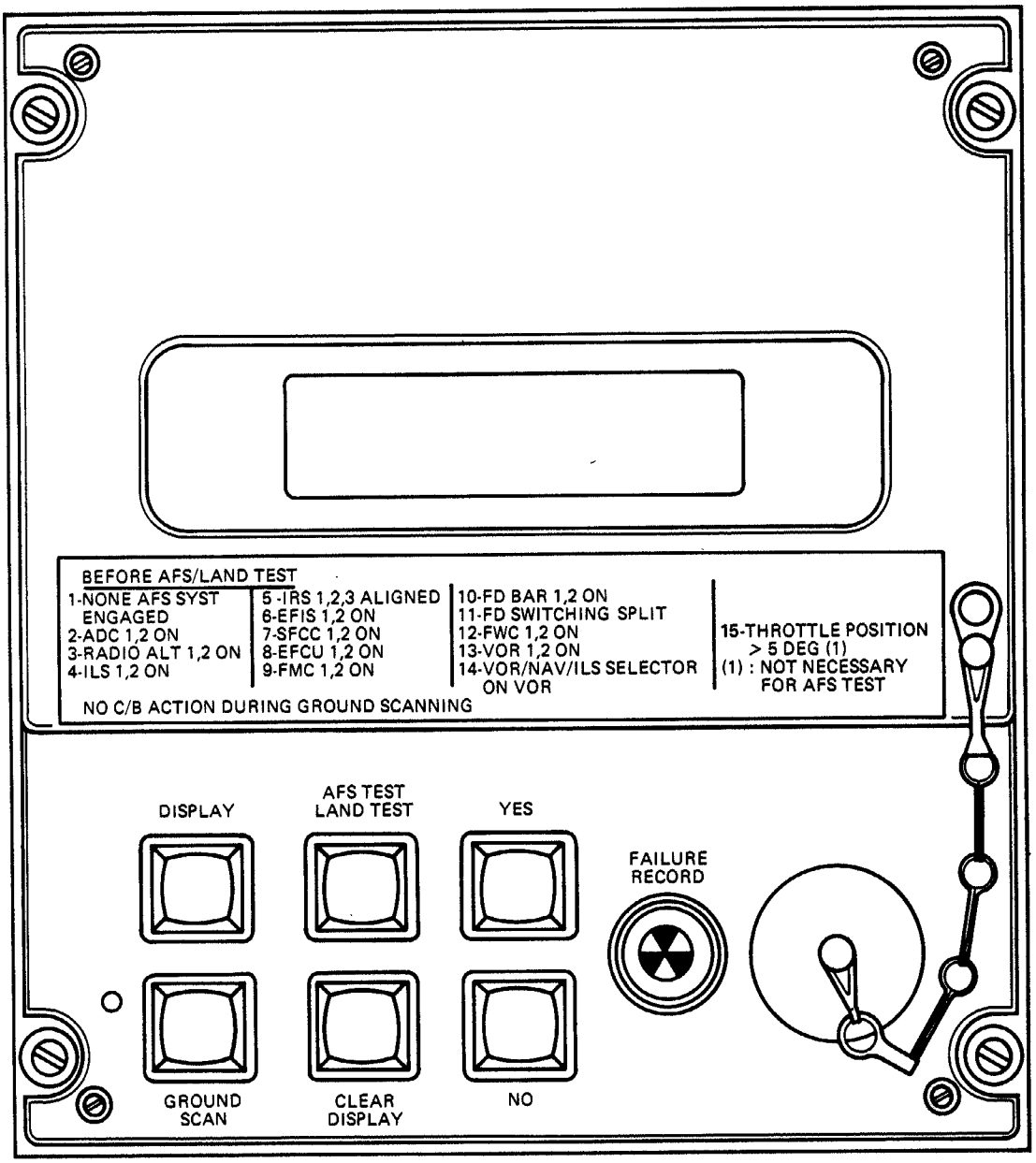
3. Connections Between Components Associated with AFS

A. Connections Between AFS Components
Most connections between the AFS components are accomplished through ARINC Specification 429 data bus lines.
Specific signals e.g. autotrim command, are in conventional form i.e. analog or discrete logic signals.
The method known as the closed loop method has been adopted to check out bus line integrity. Such a method is contingent on the bi-processor architecture and uses the consolidated point principle. The data output bus from a processor supplies the various users (other computers, peripherals, etc.).
At the end of the line, the data bus is directed to the transmitter monitoring processor where a bit by bit comparison is made for some parameters between the message transmitted and the message received.
For the FCC, for example, the flight director commands transmitted to the PFD are monitored that way.
Any fault detection made by this type of monitoring means that the transmitter is faulty :
- AP system, trim etc., disengages
- ARINC transmission is interrupted.

B. Connections Between Peripherals and AFS
Most connections between AFS and peripherals are accomplished through ARINC Specification 429 data buses (ADC , IRS, R/A, ILS receiver etc.).
This interface is described in corresponding section .

4. Maintenance
AFS in-line maintenance is based on use of the MTP system described in 22-40-00.
This system fulfills the following functions :
- in flight : LRU fault isolation
- on the ground : . AFS self-test initiation
. LAND mode availability test
. complex fault isolation.

The test results are displayed on the MTP through a display window comprising 32 characters. Messages are displayed in plain English.
(Ref. Fig. 016)



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AFS - Maintenance Test Panel
Figure 016

1. Operational Test

A. Power Supply Cutoff and Restoration Logic
(1)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(2)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS.
- Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- Make certain that ADS is operating
- Align the IRS.
- Make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU.
- On left and right instrument panels make certain that FD pushbutton switches legends are off
- On glareshield, FD/FPV selector switches in ON position
- Close circuit breaker 5RK on panel 22VU, line 204/F27.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 engage lever in ON position	Lever remains in this position. Check that the following AFS components are energized : - flight augmentation computers : SPD LIM out of view on airspeed scale on PFDs - flight control computers : white FD1 indication appears on captain PFD White FD2 indication appears on First Officer PFD - flight control unit : all displays show indications - thrust control computer :

ACTION	RESULT
	THR LIMIT value appears on TRP
2. Place PITCH TRIM 1 engage lever in OFF position	Check that, after a few seconds, the following components are de-energized : - flight augmentation computers : SPD LIM message appears on PFDs - flight control computers : white FD indication disappears on PFDs - flight control unit : indications disappear - thrust control computer : no value on TRP.
3. Place PITCH TRIM 2 levers successively in ON and then OFF position	At engagement, same result as in step 1 At disengagement, same result as in step 2.
4. On maintenance test panel	
- press GROUND SCAN pushbutton switch (green indicator light comes on)	Wait for few seconds (safety test duration) then same result as in step 1
- press GROUND SCAN pushbutton switch again (indicator light goes off)	Same result as in step 2
- press AFS TEST/LAND TEST pushbutton switch.	Wait for few seconds (safety test duration) then same result as in step 1
- press CLEAR pushbutton switch.	On MTP display goes off then same result as in step 2.
5. On panel 22VU	
- open circuit breaker 5RK (line 204/F27)	Check that the following AFS components are energized - flight augmentation computers : SPD LIM out of view on airspeed scale on PFDs - flight control computers : white FD1 indication appears on Captain PFD white FD2 indication appears on First Officer PFD - flight control unit : all displays show indications - thrust control computer : TAT value appears on TRP.
- close circuit breaker 5RK	Same result as in step 2.

ACTION	RESULT
6. On panel 21VU - open circuit breaker 306CC1 (line 103 G/3)	On MTP - display comes on for one second On flight control unit - all displays show indications.
- close circuit breaker 306CC1	On flight control unit - indications disappear after approx. 1 minute.
- open circuit breaker 306CC2 (line 103 G/12) - close circuit breaker 306CC2	On flight control unit - all displays show indications On flight control unit - indications disappear after approx. 1 minute.

NOTE : Energization of AFS components by means of the ANN LT selector switch will be checked in the next test.

(c)Close-up

- De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

B. AFS Light Test and Display

(1)Reason for the Job

- Test of lighting and display.
- Operational test of windshear warning system through light test with single FWS energized.
- Light test with single FWS energized.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(3)Procedure :

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Close overhead panel circuit breakers particularly those associated with AFS.
- Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- Open FWC1 circuit breaker 7WW on panel 21VU, line 108/B8.

(b)Test

NOTE : The test is performed with only FWC2 energized. To perform the

test with FWC1 energized open FWC2 C/B 32WW on panel 21VU, line 109/A8.

ACTION	RESULT
1. On overhead panel 436VU - place ANN LT selector switch in TEST position	Check that the following lights, displays or indications come on : On main instrument panel : - amber light on altimeter Two red AUTO-LAND warning lights Two red MASTER WARN warning lights. On the PFD's : - red WINDSHEAR legend is displayed On CAPT and F/O loudspeakers - WINDSHEAR is heard 3 times. On flight control unit : - SPD/MACH display : 888 - ALT SEL display : 88800 - HDG SEL display : 888 - V/S display : + 88 - PRESET indication - all pushbutton switches On L and R instrument panels : - all pushbutton switches. On thrust rating panel : - all pushbutton switches. - TAT display : + 88 - THR LIMIT display : 8.8.8.8 - LIM MODE display : all segments used for mode display - TARGET display : 8.8.8.8 - FLX TO TEMP display : + 88
- place ANN LT selector switch in BRT position	Check that after a few seconds, lights, displays and indications listed above go off.

(c)Close-up
- De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

1. Introduction

The FCC groups in a same computer the auto pilot (AP) and flight director (FD) functions.

The functions performed by the FCC are as follows :

- stabilizing the aircraft around its center of gravity while holding vertical speed and heading (AP basic modes in CMD and FD mode)
- stabilizing the aircraft around its center of gravity while holding roll and pitch attitudes, the reference values of these parameters can be modified by pilot action on control column (AP function in CWS mode)
- flying the aircraft on an acquired flight path (hold modes available in FD or in AP (CMD))
- Acquisition of new flight paths (modes with arming, capture and hold phases, functions available in FD or in AP (CMD))
- Take off (mode available in FD as soon as ground roll is initiated and in AP (CMD) only after rotation and takeoff)
- Automatic landing, roll out included (available in FD and in AP (CMD)).

The FCC generates AP and FD commands for the three axes, pitch, roll and yaw ; yaw axis being used only in approach or at takeoff (slats extended).

These commands are sent to :

- The pitch, roll and yaw actuators for the autopilot in CMD
- The pitch and roll actuators for the autopilot in CWS
- The PFD pitch, roll and yaw bars for manual piloting through the flight director.

The FCCS also sends an autotrim command to the FACS for the pitch autotrim function and engagement modes to the TCCs for automatic selection of THR, SPD, MACH, P.SPD or RETARD modes.

R (Ref. Fig. 001)

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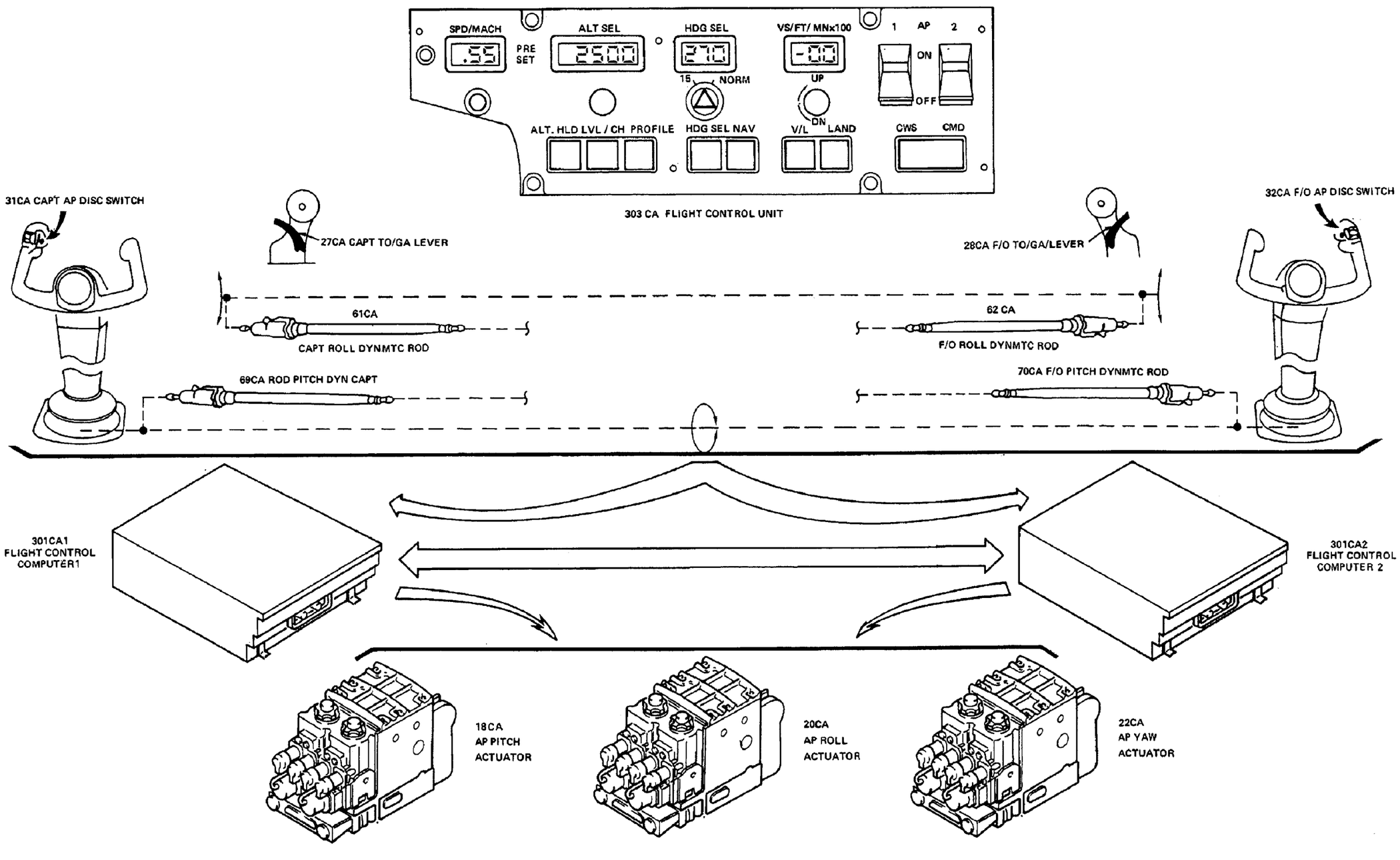
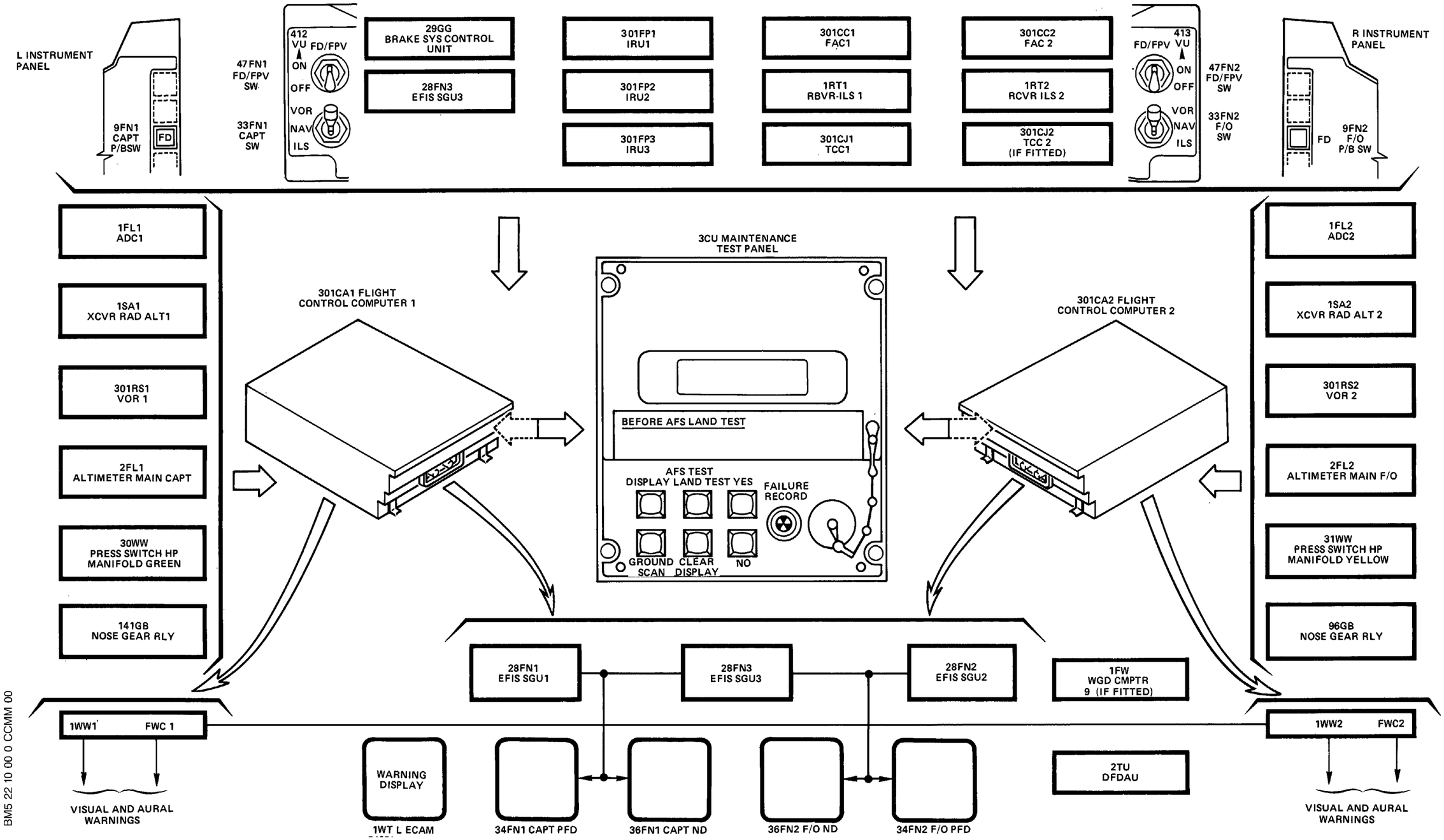


FCCS - Interface (Sheet 1/2)
Figure 001



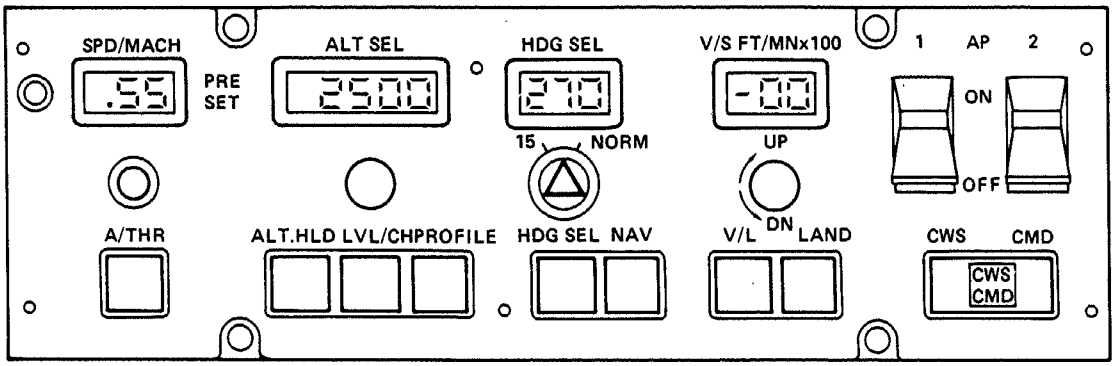
FCCS - Interface (Sheet 2/2)
Figure 001

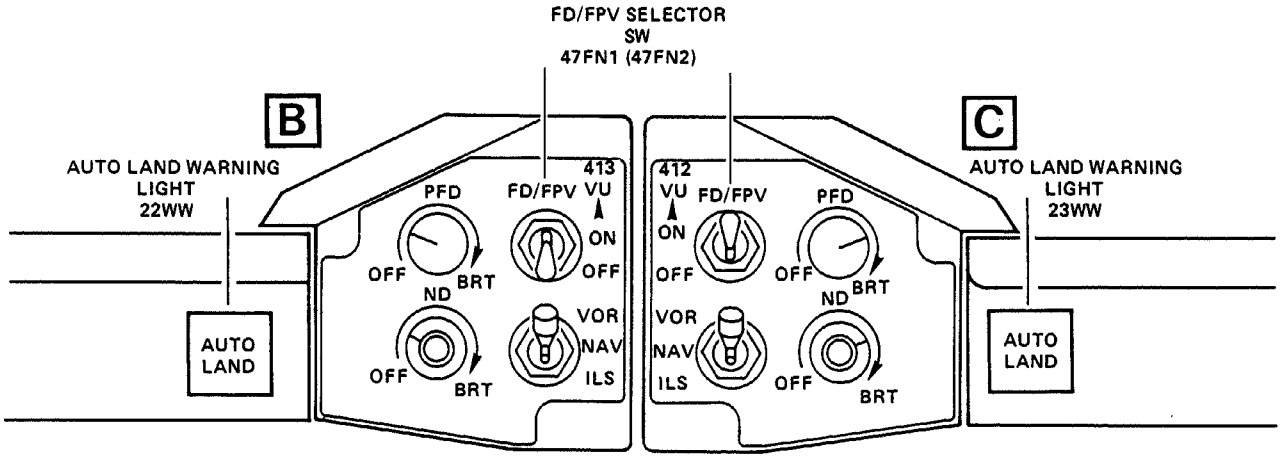
R

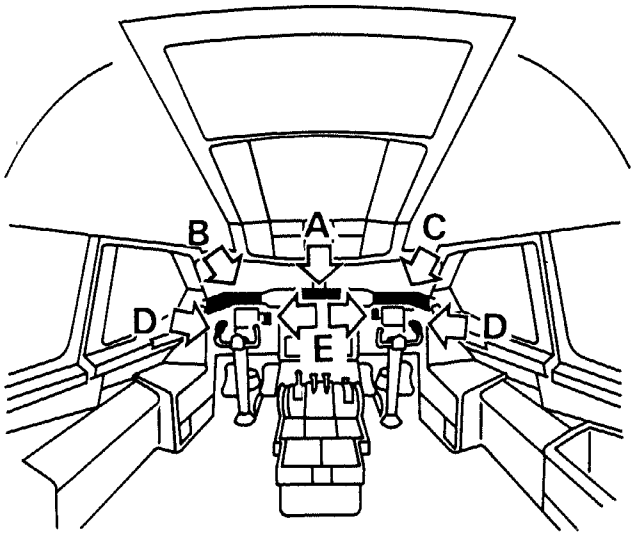
2. Component Location

R (Ref. Fig. 002)

FIN	FUNCTIONAL DESIGNATION	PANEL	ZONE	ACCESS DOOR	ATA REF.
301CA1	FCC-1	83VU	121	121BL	22-18-34
301CA2	FCC-2	84VU	121	121BL	22-18-34
61CA	ROD-ROLL DYNAMOMETRIC, CAPT		121	121BL	27-11-19
62CA	ROD-ROLL DYNAMOMETRIC, F/O		122	121BL	27-11-19
69CA	ROD-PITCH DYNAMOMETRIC, CAPT		121	121BL	27-31-25
70CA	ROD-PITCH DYNAMOMETRIC, F/O		122	121BL	27-31-25
18CA	ACTUATOR-AP PITCH		334	334AB/AT	22-13-23
20CA	ACTUATOR-AP ROLL		675	675AB	22-13-25
22CA	ACTUATOR-AP YAW		311	312AR	22-13-27
36FN1	ND-CAPT	3VU	211		34-75-22
36FN2	ND-F/O	5VU	212		34-75-22
27CA	LEVER-GO AROUND, L	9VU	211		
28CA	LEVER-GO AROUND, R	9VU	211		
31CA	P/BSW-AP DISCONNECT, CAPT	12VU	211		
32CA	P/BSW-AP DISCONNECT, F/O	14VU	212		
47FN1	SELECTOR SWITCH-FD/FPV, CAPT	413VU	211		
47FN2	SELECTOR SWITCH-FD/FPV, F/O	412VU	212		
36WW	LIGHT-MASTER WARN, CAPT	3VU	211		
38WW	LIGHT-MASTER WARN, F/O	5VU	212		
9FN1	P/BSW-FD, CAPT	402VU	211		
9FN2	P/BSW-FD, F/O	406VU	212		
22WW	WARNING LIGHT-AUTOLAND, CAPT	13VU	211		
23WW	WARNING LIGHT-AUTOLAND, F/O	13VU	212		
34FN1	PFD-CAPT	3VU	211		34-74-22
34FN2	PFD-F/O	5VU	212		34-74-22
1WT	DISPLAY UNIT-ECAM, L	4VU	210		31-57-22
303CA	FCU	13VU	211		22-15-12







MASTER WARN LIGHT
36WW (38WW)

AP DISCONNECT P/B SW
31CA/32CA

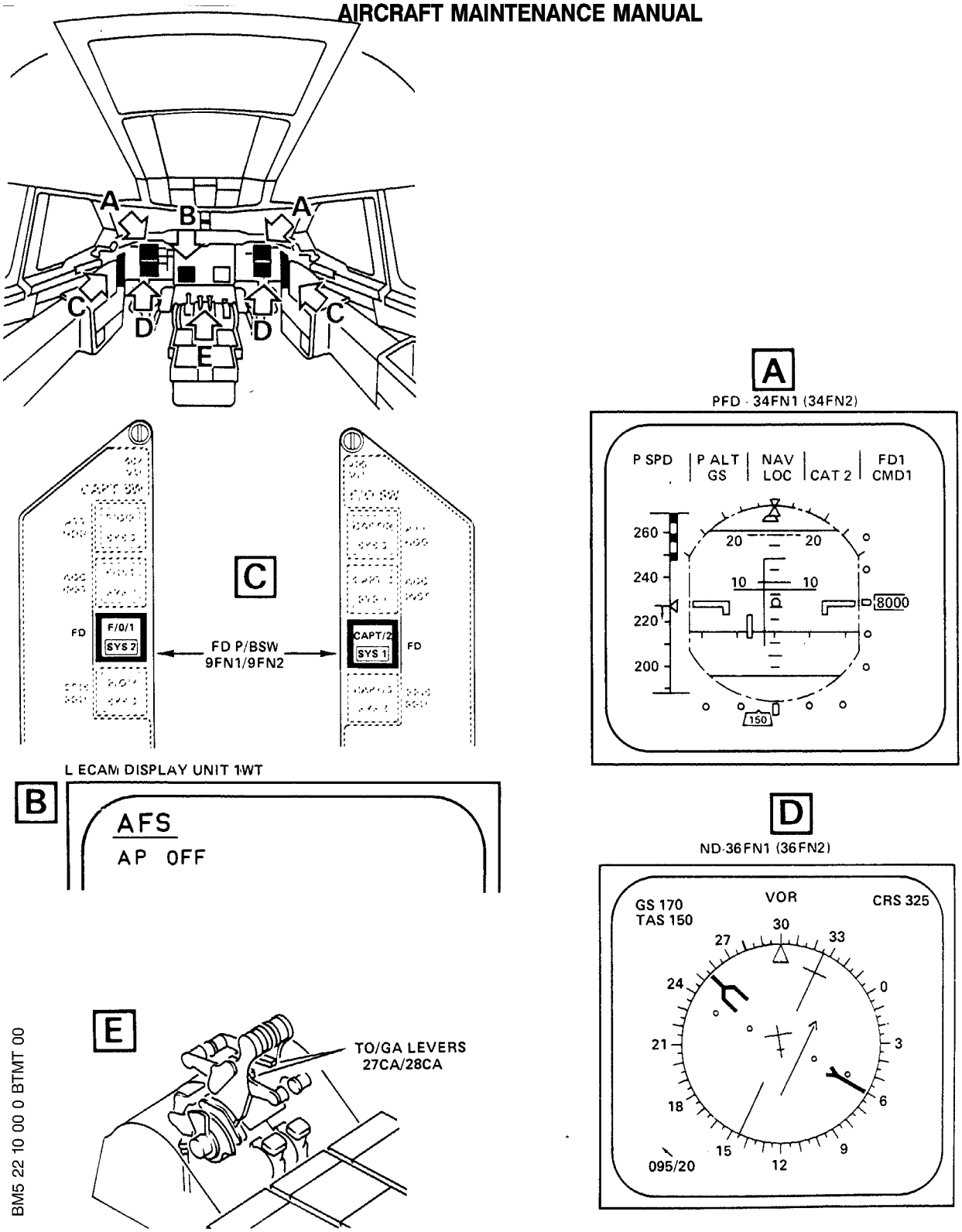
D

CAPT / F/O CONTROL WHEEL

BM5 22 10 00 0 BTMF 00

FCCS - Component Location (Sheet 1/3)
Figure 002

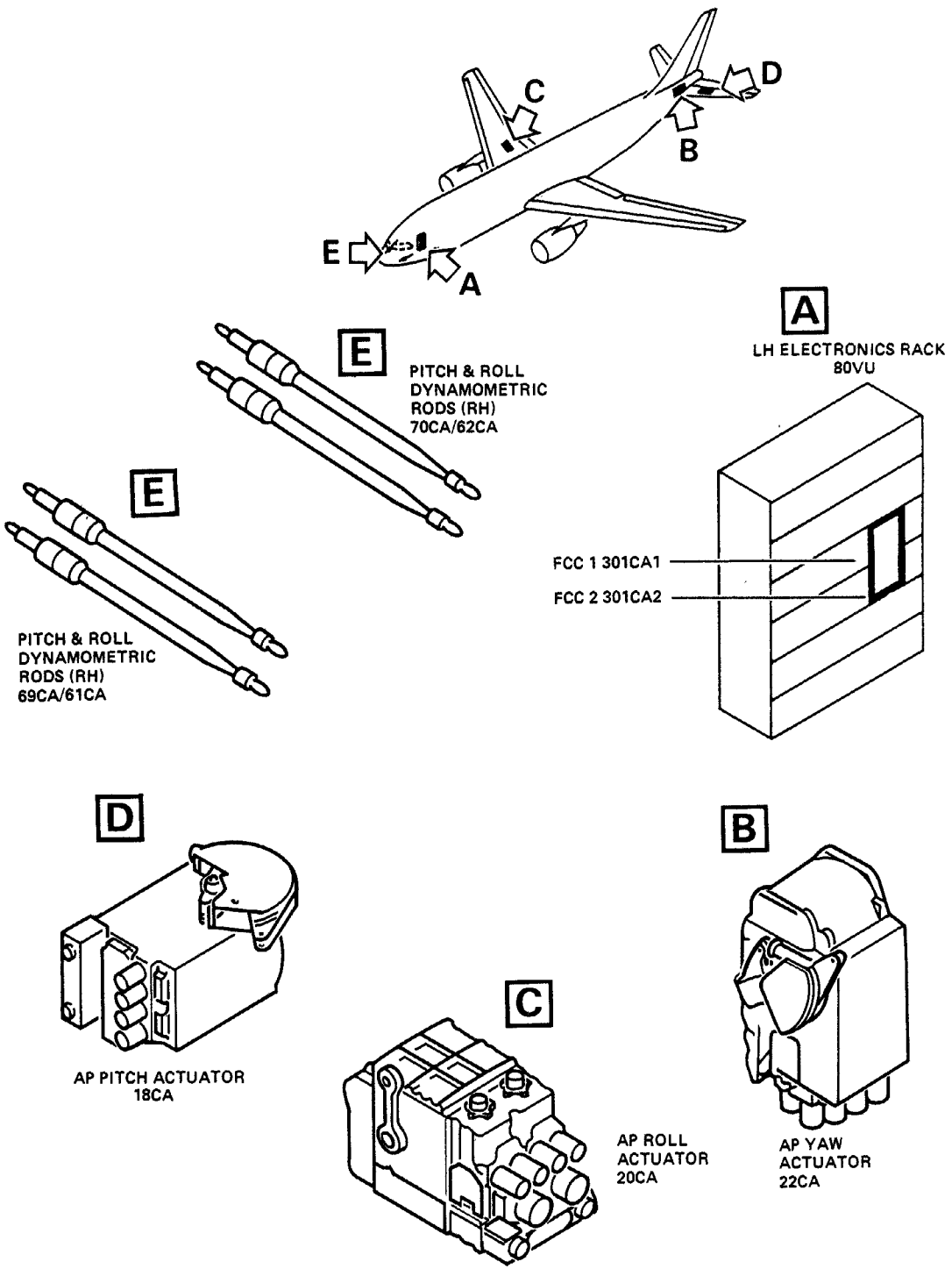
R



BM5 22 10 00 0 BTMT 00

FCCS - Component Location (Sheet 2/3)
Figure 002

R



BM5 22 10 00 0 BTMX 00

FCCS - Component Location (Sheet 3/3)
Figure 002

R

3. Functions Provided - Sub System Architecture

A. Functions Provided

(1)Modes available

The FCC carries out autopilot and flight director functions.

The following table lists the modes available in AP and/or FD.

(2)AP/FD engagement and mode selection.

(a)As regards engagement ;

1 The FD is automatically engaged upon energization of the FCC ; there is no associated engage switch. However the Captain and the First Officer can clear the FD bars displayed on the PFD by acting on the FD/FPV control. Clearing the bars on the two PFD's is equivalent to FD disengagement.

R 2 The AP is engaged through a two-position (ON-OFF) engage lever. In
R flight the AP is always engaged initially in CMD and the result of
R this is the illumination of green CMD legend on the FCU/CWS-CMD
R pushbutton switch.
R Then change to CWS is performed by pressing CWS-CMD pushbutton switch.
R This action results in extinguishing of CMD legend and illumination of
R green, CWS legend on corresponding pushbutton switch and vice versa.
R On ground with at least one engine running, the AP is engaged
R initially in CWS and action on CWS-CMD pushbutton switch is not taken
R into account. Thus this action makes engagement of AP in CMD
R impossible at take-off.

R (b)As regards mode selection

R Selection of a mode is performed either by pressing the corresponding
R mode pushbutton switch, or, in certain specific cases, (Ref. NOTE1) by
R pulling reference display selector knobs on the FCU. This enables the
R crew to rapidly select certain functions, i.e.
R Heading display and engagement of HDG SEL mode by action on the same
R selector knob.
R The selection of an upper mode causes illumination of the corresponding
R pushbutton switch.
R Further actuation of pushbutton switch causes return to basic mode (on
R corresponding axis) or disarming of an arming mode if it was only armed
R (Ref. NOTE 3).
R NOTE 1 : TO/GA modes are selected by action on specific controls
R located on throttle control levers.
R NOTE 2 : In the case of the LAND mode ; if LOC and/or GLIDE function
R was active (after capture), a second action on LAND pushbutton
R switch will cause return to basic mode on both axes.
R NOTE 3 : ALT and ALT ACQ modes are not provided with engagement
R pushbutton switches.
R ALT ACQ mode is permanently armed (except in certain specific
R cases) and is engaged when the aircraft reaches the altitude
R selected on the FCU.
R ALT mode is automaticaly engaged when the aircraft reaches the
R flight level displayed on the FCU.

AP/FD Mode Availability Table

	DESCRIPTION	AVAILABILITY	PHASES	NOTE
BASIC MODES	Vertical speed hold	FD, CMD		
	Pitch hold	CWS		
	Heading hold (HDG)	FD, CMD		
	Bank hold	CWS		
LONGITUDINAL MODES	Altitude hold (ALT)	FD, CMD		
	Altitude acquisition (ALT ACQ)	FD, CMD	ARM, CPT	No pushbutton switch
	Level change (LVL CHG)	FD, CMD		
	Speed hold (SPD)	FD, CMD		- engagement through LVL/CH Mode
	Mach hold (MACH)	FD, CMD		- engagement through LVL/CH Mode
	Profile (PROF)	FD, CMD	ARM, TRACK	AP/AT Common Mode

AP/FD Mode Availability Table (Cont'd)

	DESCRIPTION	AVAILABILITY	PHASES	NOTE
LATERAL MODES	Heading selection (HDG/S)	FD, CMD		
	LOC mode (LOC)	FD, CMD	ARM, CPT, TRK	
	NAVIGATION mode (NAV)	FD, CMD	ARM, CPT, TRK	
R R	VOR	FD, CMD	ARM, CAPT, TRK	
COMMON MODES	LANDING (LAND)	FD, CMD	LOC ARM LOC CPT | LAND GS ARM | LAND GS CPT | ARM LOC TRK | GS TRK - LAND TRK ALIGN, FLARE ROLL OUT NOSE DOWN	
R	TAKEOFF (TO)	FD, CMD		AP/FD --- SRS RWY ATS --- THR
	GO AROUND (GA)	FD, CMD		AP/FD --- SRS HDG ATS --- THR

NOTE : These modes are explained in detail in chapter 22.17.

(c)As regards functions achieved ;
- the AP/FD associated to the autothrottle always control the aircraft speed :
  The engagement of an AP/FD mode always causes engagement of an ATS mode so that control of aircraft speed is performed either by the AP/FD or the ATS.
  In case of uncompatibility : V/S mode engaged in AP and SPD mode engaged in ATS ; if the required vertical speed is excessive, priority is given to speed hold (vertical speed is no longer maintained)
- the following table gives the corresponding ATS modes with respect to the AP/FD modes engaged :

	AP/FD Modes	Corresponding ATS Modes	Remarks
	V/S	SPD	
	ALT	SPD	
	ALT ACQ	SPD	Capture Phase
	G/S	SPD	Capture and track phase
R	LVL/CH (SPD or MACH) climb descent	THR THR RETARD	
	PROFILE	PROFILE	Speed control will be performed by the AP or the ATS as per FMC commands with respect to the flight case

B. Sub System Architecture
The description of the various components, FCU, AP actuators, dynamometric rods will be found in 22-15 and 22-13.
FCC description.
(1)Dimensions
The computer is a 10MCU type box (in accordance with ARINC 600) equipped with a rear connector, size 3, ARINC 600 type and two front test connectors
(2)Weight
The computer weight is less than 16Kgs.
(3)Current consumption
The permanent current consumption of a FCC computer is 100 W from the 28 VDC network.

R

1. Operational Test

A. FD Engage Logic

(1)Reason for the job

To check FD engagement, bars removal logic and switching.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- on left and right instrument panels 402VU and 406VU, make certain that FD pushbutton switches legends are off.
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- on glareshield, place FD/FPV selector switches in ON position.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 engage lever in ON position	Lever remains in this position. On Captain PFD - white FD1 indication appears - FD pitch and roll bars appear. On First Officer PFD - white FD2 indication appears - FD pitch and roll bars appear.
2. On Captain's complementary control panel 413VU - place FPV/FD swicth in FPV/FD	On Captain PFD - FD pitch and roll bars disappear - white FD1 indication disappears.

ACTION	RESULT
position and release - place FPV/FD switch in FPV/FD position again and release	On Captain PFD - FD pitch and roll bars appear - white FD1 indication appears.
- place this switch in OFF position	On Captain PFD - FD pitch and roll bars disappear - white FD1 indication disappears.
3. On First Officer's complementary control panel 412VU - place FPV/FD switch in FPV/FD position and release	On First Officer PFD - FD pitch and roll bars disappear - white FD2 indication disappears.
- place FPV/FD switch in FPV/FD position again and release	On First Officer PFD - FD pitch and roll bars appear - white FD2 indication appears.
- place this switch in OFF position	On First Officer PFD - FD pitch and roll bars disappear - white FD2 indication disappears.
4. On complementary control panels - place both FPV/FD switches in ON position	On Captain PFD - FD pitch and roll bars appear - white FD1 indication appears. On First Officer PFD - FD pitch and roll bars appear - white FD2 indication appears.
5. On left instrument panel 402VU - press FD pushbutton switch	On this pushbutton switch - SYS 2 comes on. On right instrument panel, on FD pushbutton switch - CAPT/2 comes on. On Captain PFD - white FD2 indication replaces FD1 On First Officer PFD - white FD2 indication is visible.
6. On right instrument panel 406VU - press FD pushbutton switch	On this pushbutton switch - SYS 1 comes on - CAPT/2 goes off. On left instrument panel, FD pushbutton switch unlatches and : - SYS 2 goes off - F/O/1 comes on. On PFDs - white FD1 indication is visible
- press FD pushbutton switch again	On this pushbutton switch : - SYS 1 goes off. On left instrument panel, - F/O/1 goes off.

ACTION	RESULT
	On Captain PFD - white FD1 indication is visible. On First Officer PFD - white FD2 indication replaces FD1.

7. On overhead panel
   - place PITCH TRIM 1 engage lever in OFF position.

(c)Close-up
- De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

B. AP Test in CWS Mode

(1)Reason for the job

To check that actions on control column and control wheel provide corresponding control surface deflection.

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW)

  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).

- close overhead panel circuit breakers, particularly those associated

with AFS, NAVIGATION, ECAM and ADS
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- on center pedestal, adjust ECAM display unit brightness and select FLT CTL page
- on overhead panel, make certain that PITCH TRIM 1, YAW DAMPER 2 are engaged.

NOTE : The test described below, concerns side 1. For side 2 which is identical, refer to information between parentheses.

(b)Test

ACTION	RESULT
1. On flight control unit - place AP1 (2) lever in ON position	Lever remains in this position On FCU - green CWS comes on. On PFDs - amber CWS1 (2) indication appears.
2. Move control column in Nose up then Nose down direction	On R ECAM display unit - index on ELEV scale reflects control column movement.
3. Turn control wheel to the right then to the left	On R ECAM display unit - L and R AIL and RUD index movement reflects control wheel movement.
4. Place AP1 (2) lever in OFF position and cancel warnings by pressing AP disconnect pushbutton switch.	
5. De-energize AFS components by placing PITCH TRIM 1 engage lever in OFF position.	

(c)Close-up
- Shut down and disconnect hydraulic ground power cart.
- De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Restore aircraft and system to normal operating condition.

C. AP Test in CMD Mode, AP Instinctive Disconnect P/B Test, AP OFF Warning Test

(1)Reason for the job

To check one longitudinal mode and one lateral mode.

To check AP instinctive disconnect P/B function.
To check AP OFF warning function.

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW)

NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).

NOTE : In order to avoid the effects of a possible insufficient Green hydraulic flow when performing AP2 test, it is recommended to use the ground cart instead of electric pumps and PTU, or to set the Green servo control to OFF when using the alternative hydraulic generation.

- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD and ND image brightness on panels 412VU and 413VU
- on Captain and First Officer EFIS control panels, make certain that mode selector switch is in ROSE position
- on overhead panel, make certain that PITCH TRIM 1, YAW DAMPER 2 are engaged
- make certain that slats are extended.
- close circuit breaker 5RK (panel 22VU, line 204/F27).
- on glareshield, place FD/FPV selector switches in ON position.

NOTE : The test described below, concerns side 1. For side 2 which is identical, refer to information between parentheses.

(b)Test

ACTION	RESULT
1. On Flight Control Unit (FCU)	
- place AP1 (2) lever in ON position	Lever remains in this position On FCU - green CWS comes on. On PFDs - amber CWS1 (2) indication appears.
- press CWS/CMD pushbutton switch	On this pushbutton switch - green CWS goes off - green CMD comes on. On PFDs - white CMD1 (2) indication appears - amber CWS1 (2) indication disappears - green V/S and HDG are displayed.
NOTE : If AP not in V/S mode, On FCU - select altitude of 5000 ft. - pull V/S knob	
- with V/S selector knob, select positive vertical speed (Vz = +300 ft/mn)	- control column moves in nose up direction - pitch trim control wheel rotates in nose up direction.
- select negative vertical speed (Vz = -300 ft/mn)	- control column moves in nose down direction - pitch trim control wheel rotates in nose down direction. - select null vertical speed.
- with HDG SEL selector knob select heading deviation of 10° to right	On NDs - heading index moves accordingly.
- pull HDG SEL selector knob	On FCU - HDG SEL pushbutton switch comes on green. On PFDs - green HDG goes off - green HDG/S comes on - FD roll bar moves to the right Control wheel rotates to the right. Rudder pedals move accordingly.
- press HDG SEL pushbutton switch	This pushbutton switch goes off. On PFDs - green HDG/S goes off - green HDG comes on - FD roll bar returns to center position.
- pull HDG SEL selector knob and	On NDS

ACTION	RESULT
gradually turn **HDG SEL** selector knob to the left until rudder pedals and control wheel return to neutral position, then with **HDG SEL** selector knob, select heading deviation of 10° to left.	- heading index moves accordingly This pushbutton switch comes on green. On **PFDs** - green **HDG** goes off - green **HDG/S** comes on - **FD** roll bar moves to the left. control wheel rotates to the left. Rudder pedals move accordingly.
R - gradually turn heading selector knob to the right until rudder pedals and control wheel return to neutral position then select aircraft heading.	Control wheel returns towards neutral position. Rudder pedals return towards neutral position.
R - press **HDG SEL** pushbutton switch	This pushbutton switch goes off. On **PFDs** - green **HDG/S** goes off - green **HDG** comes on - **FD** roll bar returns to center position.
2. On control wheel - disconnect **AP1 (2)** with **CAPT** instinctive disconnect pushbutton.	On **FCU** - **AP1 (2)** lever returns to **OFF** position. On **CAPT** and **F/O** main instrument panel - a red **MASTER WARN** light flashes **NOTE** : Check that the upper and lower parts of **"MASTER WARN"** words are illuminated. On **L ECAM CRT** - a red **AP OFF** warning message is displayed. In **CAPT** and **F/O** loudspeaker - an aural warning (cavalry charge) sounds.
3. On control wheel - cancel warnings by pressing **CAPT AP** disconnect pushbutton.	Warnings are cancelled.
4. On Fligh Control Unit - place **AP1 (2)** lever in **ON** position and press **CWS/CMD** pushbutton	On **FCU** - **AP1 (2)** remains in this position - green **CMD** comes on. On **PFDs**

ACTION	RESULT
switch.	- white CMD1 (2) indication appears - green V/S and HDG are displayed.
5. Perform again steps 2 thru 4 but press F/O instinctive disconnect pushbutton instead of CAPT in steps 2 and 3.	
6. On panel 22VU - open circuit breaker 5RK (line 204/F27).	AP1(2) lever returns to OFF position. On FCU, on CWS/CMD pushbutton switch - green CMD goes off. On PFDs - white CMD1 (2) indication disappears.
7. De-energize AFS components by placing PITCH TRIM 1 engage lever in OFF position.	
8. Close circuit breaker 5RK (line 204/F27).	

(c)Close-up
- Shut down and disconnect hydraulic ground power cart.
- De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Restore aircraft and system to normal operating condition.

R D. Check of Pitch Trim Autotrim Circuits

(1)Reason for the job

To ensure integrity of autotrim function for all possible combination of Autopilot (AP) and Pitch Trim.

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart - Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW)
  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).
  NOTE : In order to avoid the effects of a possible insufficient Green hydraulic flow when performing AP2 test, it is recommended to use the ground cart instead of electric pumps and PTU, or to set the Green servo control to OFF when using the alternative hydraulic generation.
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary)
- make certain that EFIS is operating and adjust PFD and ND image brightness on panels 412VU and 413VU
- on Captain and First Officer EFIS control panels, make certain that mode selector switch is in ROSE position
- on overhead panel, make certain that PITCH TRIM 1, YAM DAMPER 2 are engaged
- make certain that flap and slat control lever is in the fourth notch (SLATS/FLAPS position 20/20)
- close circuit breaker 5RK on panel 22VU (line 204/F27)
- on glareshield, place FD/FPV selector switches in ON position.
  NOTE : The test described below, concerns side 1. For side 2 which is identical, refer to information between parentheses.

(b)Test

This test aims at ensuring the integrity of the autotrim function for all possible combinations of Autopilot (AP) and pitch trim.

ACTION	RESULT
1. On Flight Control Unit (FCU) : - place AP1 (2) lever in ON position	- lever remains in this position. On FCU : - green CWS comes on. On PFDs : - amber CWS1 (2) indication appears.

ACTION	RESULT
- press **CWS/CMD** pushbutton switch	On this pushbutton switch : - green **CWS** goes off - green **CMD** comes on. On **PFDs** : - white **CMD1 (2)** indication appears - amber **CWS1 (2)** indication disappears - green **V/S** and **HDG** are displayed
NOTE : If **AP** not in **V/S** mode, on **FCU** : - select altitude of 5000 ft - pull **V/S** knob.	
- with **V/S** selector knob, select positive vertical speed (Vz = +3000 ft/min)	- control column moves in nose up direction - pitch trim control wheel rotates in nose up direction without **PITCH TRIM 1** disconnection
- select negative vertical speed (Vz = -3000 ft/min)	- control column moves in nose down direction - pitch trim control wheel rotates in nose down direction without **PITCH TRIM 1** disconnection.
- place **PITCH TRIM 2** engage lever in **ON** position - disengage **PITCH TRIM 1**	**AP1 (2)** remains in **CMD**
- with **V/S** selector knob, select positive vertical speed (Vz = +3000 ft/min)	- control column moves in nose up direction - pitch trim control wheel rotates in nose up direction without **PITCH TRIM 2** disconnection
- select negative vertical speed (Vz = -3000 ft/min)	- control column moves in nose down direction - pitch trim control wheel rotates in nose down direction without **PITCH TRIM 2** disconnection.
2. On Flight Control Unit (FCU) : - place **AP1 (2)** lever in **OFF** position and cancel warning	**AP1 (2)** lever returns to **OFF** position On **FCU**, on **CWS/CMD** pushbutton switch : - green **CMD** goes off. On **PFDs** : - white **CMD1 (2)** indication disappears.

ACTION	RESULT
3. Engage PITCH TRIM 1 and disengage PITCH TRIM 2 : - perform steps 1 to 2 again using information between parentheses.	
4. De-energize AFS components by placing both PITCH TRIM engage levers in OFF position.	
(c)Close-up - Shut down and disconnect hydraulic ground power cart. - De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301). - Restore aircraft and system to normal operating condition.	

E. AP Pitch Mechanical Override

(1)Reason for the job

To check AP mechanical override and PITCH TRIM disengagement while actuating AP pitch torque limiter lever.

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
(2)98A27903501001	Measuring Equipment - Pitch and Roll Operating Load
(3)97A27003000002 or 97A27003000003 or 97A27003000004 or 97A27003000006 or 97A27003000007	Gage Assy - Push-Pull
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW)

  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION, ECAM and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that AP is not engaged
- make certain that pitch trim control wheel is in neutral position
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that PITCH TRIM 1 and 2 are engaged
- make certain that YAW DAMPER 2 is engaged
- on center pedestal, adjust ECAM display unit brightness and select FLT CTL page
- close circuit breaker 5RK on panel 22VU (line 204/F27)
- on glareshield, place FD/FPV selector switches in OFF position.
- install pitch and roll operating load measuring equipment and push-pull gage assy.

(b)Test

NOTE : Threshold values given in the test are approximate. (They are just an order of magnitude).

ACTION	RESULT
1. On flight control unit (FCU) - place AP1 lever in ON position	On FCU : - green CWS comes on - lever remains engaged. On primary flight displays (PFDs) - amber CWS1 indication appears
2. Press CWS/CMD pushbutton switch	On this pushbutton switch - green CMD comes on - green CWS goes off. On PFDs - white CMD1 indication appears

ACTION	RESULT
	- amber CWS1 indication disappears - green V/S and HDG are displayed
NOTE : If AP not in V/S mode, On FCU - select altitude of 5000 ft. - pull V/S knob.	
3. Select positive vertical speed (Vz + 500 ft/mn)	Check that control column moves in nose up position and pitch trim control wheel rotates in nose up direction.
4. Move control column in aircraft nose down position	Check that, past load threshold (20 daN nose down approx.) : On ECAM FLT CTL system page : - ELEV scale reflects control column movement - STAB scale reflects pitch trim control wheel rotation On FAC/ATS engage unit : - PITCH TRIM 1 lever returns to OFF position (After some delay).
5. Release control column	
6. Select null vertical speed	
7. Place PITCH TRIM 1 lever in ON position	
8. Select negative vertical speed (Vz = -500 ft/mn)	Check that control column moves in nose down position and pitch trim control wheel rotates in nose down direction.
9. Move control column in aircraft nose up position	Check that, past load threshold (46 daN nose up approx.) : On ECAM FLT CTL system page : - ELEV scale reflects control column movement - STAB scale reflects pitch trim control wheel rotation On FAC/ATS engage unit : - PITCH TRIM 1 lever returns to OFF position (After some delay).
10. Release control column.	
11. Disengage AP1 and cancel warnings	

ACTION	RESULT
by pressing AP disconnect pushbutton switch.	
12. De-energize AFS components by placing PITCH TRIM 2 lever in OFF position.	

(c) Close-up
1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

F. AP Roll Mechanical Override

(1) Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
(2) 98A27903501001	Measuring Equipment - Pitch and Roll Operating Load
(3) 97A27003000002 or 97A27003000003 or 97A27003000004 or 97A27003000006 or 97A27003000007	Gage Assy - Push-Pull
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxilary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(2) Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.

(a) Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)

- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW)
  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION, ECAM and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that AP is not engaged
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that PITCH TRIM 1 is engaged
- make certain that YAW DAMPER 2 is engaged
- on center pedestal, adjust ECAM display unit brightness and select FLT CTL page
- close circuit breaker 5RK on panel 22VU (line 204/F27)
- on glareshield, place FD/FPV selector switches in OFF position.
- install pitch and roll operating load measuring equipment and push-pull gage assy.

(b)Test

NOTE : Threshold values given in the test are approximate. (They are just an order of magnitude).

ACTION	RESULT
1. On flight control unit (FCU)	On FCU :
- place AP1 lever in ON position	- green CWS comes on - lever remains engaged. On primary flight displays (PFDs) - amber CWS1 indication appears.
- Press CWS/CMD pushbutton switch	On this pushbutton switch - green CMD comes on - green CWS goes off. On PFDs - white CMD1 indication appears - amber CWS1 indication disappears - green HDG is displayed.
- Pull heading selector knob (selection of HDG SEL mode) release it and select heading deviation of 10° to right	- green HDG SEL comes on - control wheel rotates to the right.
2. Override command by rotating control wheel to the left	Check that, past load threshold (15 daN approx.), on ECAM FLT CTL system page : - AIL scales reflect aileron movement.

ACTION	RESULT
3. Release control wheel	Control wheel returns to position corresponding to command given (automatic resetting of torque limiter lever).
4. Disengage AP1 and cancel warnings by pressing AP disconnect pushbutton switch.	
5. De-energize AFS components by placing PITCH TRIM 1 lever in OFF position.	

(c)Close-up
1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

G. AP Yaw Mechanical Override
(1)Equipment and materials

ITEM	DESIGNATION
(1)	Cart - Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(2)Procedure
WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.

(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW)

NOTE : As an alternative, it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).

- close overhead panel circuit breakers, particularly those asso-

ciated with AFS, NAVIGATION, ECAM and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that slats are extended
- make certain that PITCH TRIM 1 is engaged
- make certain that YAW DAMPER 2 is engaged
- On center pedestal, adjust ECAM display unit brightness and select FLT CTL page.
- on glareshield, place FD/FPV selector switches in OFF position.

(b)Test

NOTE : Threshold values given in the test are approximate. (They are just an order of magnitude).

ACTION	RESULT
1. On flight control unit (FCU) - place AP1 lever in ON position	On FCU : - green CWS comes on - lever remains engaged. On primary flight displays (PFDs) - amber CWS1 indication appears.
- Press CWS/CMD pushbutton switch	On this pushbutton switch - green CMD comes on - green CWS goes off. On PFDs - white CMD1 indication appears - amber CWS1 indication disappears - green HDG is displayed.
- Pull heading selector knob and release it. Then select heading deviation of 10° to right then return to aircraft heading.	- green HDG SEL comes on - rudder pedals move accordingly.
2. Operate rudder pedals	Check that, past load threshold (65 daN approx.) : On ECAM FLT CTL system page : - RUD scale reflects rudder pedals movements.
3. Release rudder pedals	Rudder pedals return to initial position (automatic resetting of torque limiter lever).
4. Disengage AP1 and cancel warnings by pressing AP disconnect pushbutton switch.	
5. De-energize AFS components by placing	

ACTION	RESULT
PITCH TRIM 1 lever in OFF position. (c)Close-up - Shut down and disconnect hydraulic ground power cart. - De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301). - Restore aircraft and system to normal operating condition. H. Altitude Alert Test (1)Reason for the job To check correct operation of altitude alert function. (2)Equipment and materials	

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- close overhead panel circuit breakers, particularly those associated with AFS, ECAM, ADS and NAVIGATION.
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- on captain and First Officier main altimeters push BARO set knob and read altitude of 0 ft. approx.
- make certain that slats are retracted.

NOTE : The test described below concerns side 1. For side 2 which is identical, refer to information between parentheses.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 engage lever in ON position.	Lever remains in this position.
2. On FCU - select an altitude of 1000 ft. using ALT selector knob.	
3. On CAPT (F/O) main altimeter	On CAPT and F/O main altimeters

ACTION	RESULT
- select an altitude of 500 ft. using BARO set knob.	- altitude alert warning lights come on.
- select an altitude of 0 ft.	On CAPT and F/O main altimeters - altitude alert warning lights flash. - C-Chord aural warning sounds continuously.

4. On FCU
   - select 3000 ft. using ALT selector knob to cancel aural warning and warning lights.

5. On overhead panel
   - place PITCH TRIM 1 engage lever in OFF position.

(c)Close-up
- De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

J. TO/GA Lever Test

(1)Reason for the job

To check TO/GA lever function

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- make certain that electronics racks ventilation is correct.
- close overhead panel circuit breakers, particulary those associated with AFS, ECAM, ADS and NAVIGATION.
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that AP is not engaged
- make certain that ADS is operating
- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags

on NDs disappear ; present position entering is not necessary) (Ref. 34-25-00, P. Block 501)
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that PITCH TRIM 1 is engaged
- make certain that YAW DAMPER 2 is engaged
- on glareshield, place FPV/FD selector switches in ON position
- make certain that pitch trim control wheel is in neutral position
- make certain that flaps and slats control lever is in third notch (15/15).

(b)Test

ACTION	RESULT
1. On Captain's EFIS complementary control panel 413VU - place FPV/FD switch in OFF position.	On Captain PFD - FD pitch and roll bars disappear - white FD1 disappears.
2. On center pedestal - place throttle control levers in 0 ° position.	
3. On overhead panel - place ATS1 arming lever in ON position.	Lever remains in this position.
4. On Captain's EFIS complementary control panel 413VU - place FPV/FD switch in ON position.	- green V/S indication appears
5. At least 30 seconds after PITCH TRIM engagement, on center pedestal - press TO/GA lever on LEFT throttle control lever.	Check that on CAPT PFD - green V/S indication disappears - green SRS indication appears. On center pedestal - both throttle control levers move towards full throttle. Warnings appear and associated messages appear on L ECAM display.
6. On center pedestal - press autothrottle disconnect pushbutton on left or right throttle control lever - place throttle control lever in 0 ° position.	On center pedestal - both throttle control levers stop moving.
7. On ECAM control panel, press CLR pushbutton switch as many times as necessary to call up MEMO page on left ECAM display unit.	Warnings are cancelled.

ACTION	RESULT
8. On FCU Pull V/S knob.	On CAPT PFD - green SRS indication disappears - green V/S indication appears.
9. On First Officer's EFIS complementary control panel 412VU - place FPV/FD switch in OFF position.	On First Officer PFD - FD pitch and roll bars disappear - white FD2 indication disappears - green V/S indication disappears
10. On First Officer's EFIS complementary control panel 412VU - place FPV/FD switch in ON position.	- green V/S indication appears.
11. On center pedestal - press TO/GA lever on RIGHT throttle control lever.	Check that on F/O PFD - green V/S indication disappears - green SRS indication appears. On center pedestal - both throttle control levers move towards full throttle. Warnings appear and associated messages appear on L ECAM display.
12. Repeat actions described in step 6 thru 8.	Same as step 6 thru 8.
13. On center pedestal - place SLATS/FLAPS control lever in retracted position.	
14. Place ATS1 arming lever in OFF position and F/O FPV/FD switch in OFF position.	

(c) Close-up
1 De-energize the aircraft electrical network (Ref. 24.41.00, P. Block 301).
2 Restore aircraft and system to normal operating condition.
3 Make certain that throttle control levers are in idle position.

**ON A/C ALL

K. AP Disconnection through Torque Limiters

(1)Reason for the job

R Operational Test of AP Disconnection upon Pitch Override in all Flight Phases.

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart - Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfert Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED. MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW) (Ref. 29-10-00, P. Block 301).

  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).
  If electric pumps are used, exert the force progressively. If loss of AP occurs, use ground hydraulic power cart.
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION, ECAM and ADS
- make certain that ADS is operating
- align IRS 1, 2 and 3 (Ref. 34-25-00, P. Block 501)
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that pitch trim control wheel is in neutral position
- on glareshield, place FD/FPV selector switches in ON position
- close circuit breaker 5 RK on panel 22VU (line 204/F27)
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- on center pedestal, adjust ECAM display unit brightness and select FLT CTL page

- on center pedestal, place throttle control levers in GA position
- on overhead panel, make certain that PITCH TRIM 1 and 2, YAW DAMPER 1 and 2 are engaged
- select slats out configuration (SLATS/FLAPS control lever in notch 2)
- on glareshield, on FCU, select 5000ft on ALT SEL display
- on FCU, pull V/S knob

NOTE : The test described below, concerns side 1. For side 2 which is identical, refer to information between parentheses.

(b)Test

WARNING : BEFORE OPENING THE FOLLOWING CIRCUIT BREAKERS, MAKE SURE THAT PILOT PROTECTIVE DEVICES ARE REMOVED.

ACTION	RESULT
1. On panel 133VU : - Open circuit breakers 1GB, 2GB and 119GB (V52,U51,U52) - after at least 2 seconds, close circuit breakers 1GB, 2GB and 119GB.	
NOTE : From now on you have 30 seconds to do steps 2. and 3.	
2. On overhead panel : - place PITCH TRIM 1 and 2, YAW DAMPER 1 and 2 engage levers in ON position if any of them has disconnected during previous steps.	
3. On center pedestal, on throttle control levers: - press TO/GA levers	On Primary Flight Displays (PFD), on FMA : - green GO AROUND indication appears.
R 4. On Flight Control Unit (FCU) : R - place AP1(2) lever in ON position	On FCU : - Green CWS comes on and lever remains engaged. On PFDs : - Amber CWS 1(2) indication appears.
R 5. Move control column in aircraft nose up position to reach AP actuator electrical limit (control column is suddenly stiffer at about 18 deg. up).	On R ECAM Display Unit, on ECAM FLT CTL system page : - ELEV scale reflects control column movement.
R 6. Pull firmly the control column to override the AP pitch actuator,	On FCU : - AP1(2) lever ramains engaged.

	ACTION	RESULT
R	up to the elevator mechanical stop (about 30 deg.). This requires a force of 70 daN approximately.	
R	7. Release control column.	
	8. On overhead panel 22VU : - open circuit breaker 5RK	
R	9. Move control column in aircraft nose up position to reach AP actuator electrical limit (control column is suddenly stiffer at about 18 deg. up).	On R ECAM Display Unit, on ECAM FLT CTL system page : - ELEV scale reflects control column movement.
R	10. Pull firmly the control column to override the AP pitch actuator, up to the elevator mechanical stop (about 30 deg.). This requires a force of 70 daN approximately.	On FCU : - AP1(2) lever lever disengages
R	11. Release control column. With the circuit breaker 5RK still opened , place AP1(2) lever in ON position .	On FCU : - Green CWS comes on and lever remains engaged On PFDs : - Amber CWS 1(2) indication appears.

NOTE : If AP1(2) lever does not engage, check the pitch torque limiter switches (19CA).

(c)Close-up

1 Shut down and disconnect hydraulic ground power cart (Ref. 29-10-00, p. Block 301).

2 De-energize the aircraft electrical network (Ref. 24.41.00, P. Block 301).

3 Restore aircraft and system to normal operating condition.

Ref. MMEL Sect. 1-22 Item 1-b (Pitch Trim)

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED. BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.

1. Deactivation

A. Deactivation of the Inoperative Pitch Trim Servomotor

(1)Procedure

(a)For Pitch Trim servomotor 1 (3CC1) inoperative :

- Open, safety and tag the following circuit breaker:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC 1/115VAC	309CC1	103/G5

(b)For Pitch Trim servomotor 2 (3CC2) inoperative :

- Open, safety and tag the following circuit breaker:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC 2/115VAC	309CC2	103/G10

R B. Check of Pitch Trim

CAUTION : AFTER THE DEACTIVATION OF THE INOPERATIVE PITCH TRIM SERVOMOTOR, THE FOLLOWING CHECK OF THE PITCH TRIM MUST BE PERFORMED BEFORE EACH FLIGHT.

R (1)Reason for the job

R (a)To check that the servomotor of remaining Pitch Trim system is healthy.

R (2)Equipment and materials

ITEM	DESIGNATION
(a)	Cart - Hydraulic Ground Power Supply (aircraft - Type Pump)

Referenced Procedures

- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)

R (3)Job set-up
R (a)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
R (b)Make certain that electronics racks ventilation is correct.
R (c)Make certain that hydraulic power is available (GREEN, YELLOW) (Ref. 29-10-00, P. Block 301).
NOTE : As an alternative, it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).

R (d)Close overhead panel circuit breakers, particularly those associated with NAVIGATION, ADS and AFS except the one of inoperative Pitch Trim system (309CC1 or 309CC2).
R (e)Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
R (f)Make certain that ADS is operating.

R (4)Procedure
R (a)Test
R 1 Test of remaining Pitch Trim servomotor
This test aims at checking that the servomotor of remaining Pitch Trim system is healthy.
- Select GROUND SCAN on maintenance test panel.
NOTE : Wait for a few seconds until the red SPD LIM message goes out of view on airspeed scale on PFDs.
- On overhead panel, engage remaining pitch trim lever in ON position and disengage it.
- Visually check that there is no Pitch Trim control wheel movement.
NOTE : Dispatch is not allowed if Pitch Trim control wheel movement occurs after disengagement of pitch trim lever.
- Deselect GROUND SCAN on maintenance test panel.

R (5)Close-up
R (a)Shut down and disconnect hydraulic ground power cart.
R (b)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
R (c)Restore aircraft and system to normal operating condition.
NOTE : Make sure that the circuit breaker of the inoperative Pitch Trim servomotor is open (309CC1 or 309CC2).

R 2. Reactivation

R A. Reactivation of the Inoperative Pitch Trim Servomotor
R (1)Procedure
R (a)For Pitch Trim servomotor 1 (3CC1) inoperative :
R - Remove the safety clips and tags and close the following circuit breaker:

R PANEL	SERVICE	IDENT.	LOCATION
R 21VU	AFS/FAC 1/115VAC	309CC1	103/G5

R (b)For Pitch Trim servomotor 2 (3CC2) inoperative :

R - Remove the safety clips and tags and close the following
R circuit breaker:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC 2/115VAC	309CC2	103/G10

R (c)Refer to TSM 22-20-00.

1. Electrical Power Supply

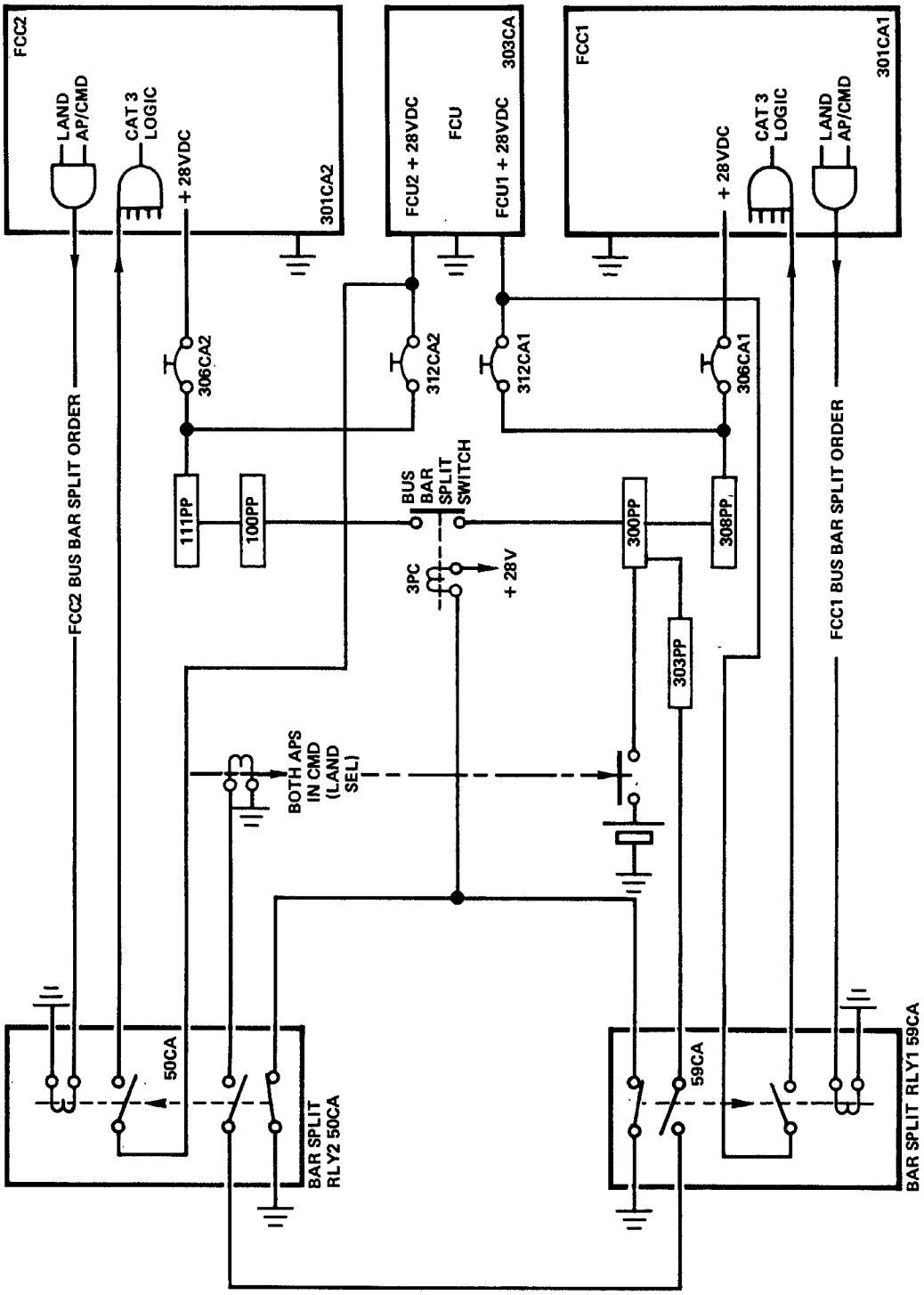
A. Aircraft Electrical Supplies

(1)28VDC power supply

This power supply constitutes the computer main supply. The system is supplied by two independent sources :

- FCC1 is supplied from main busbar through 10A calibrated circuit breaker 306CA1.
- FCC2 is supplied from normal busbar through 10A calibrated circuit breaker 306CA2.

The FCU groups two independent computers in the same box.
Each FCU computer is supplied independently from the same busbars as the FCC's but through a 5A calibrated circuit breaker ;
312CA1 for the FCU1
312CA2 for the FCU2.
In normal operation, normal and main busbars are coupled. In approach, 28V power supplies are independent (uncoupling of normal and main busbars) when both APS are engaged in CMD and with LAND mode selected.
The 28VDC power supply is shown in following figure
Monitoring of 28V power supply, description and monitoring of internal supply are described in 22.00.
(Ref. Fig. 001)



BM5 22 12 00 0 ACR2-15-10
DM5

28VDC Power Supply and Busbar Split - Simplified Schematic
Figure 001

(2)26V/400 Hz power supply

(a)Description

This power supply is used to energize the synchros and to obtain required demodulation reference.
The system is supplied with single-phase alternating current.
FCC1 is supplied through 7.5 A calibrated circuit breaker 305CA1.
FCC2 is supplied through 7.5 A calibrated circuit breaker 305CA2.
Each side of servomotor and dynamometric rod is supplied in parallel with associated circuit breaker.

(Ref. Fig. 002)

(b)26V/400 Hz monitoring

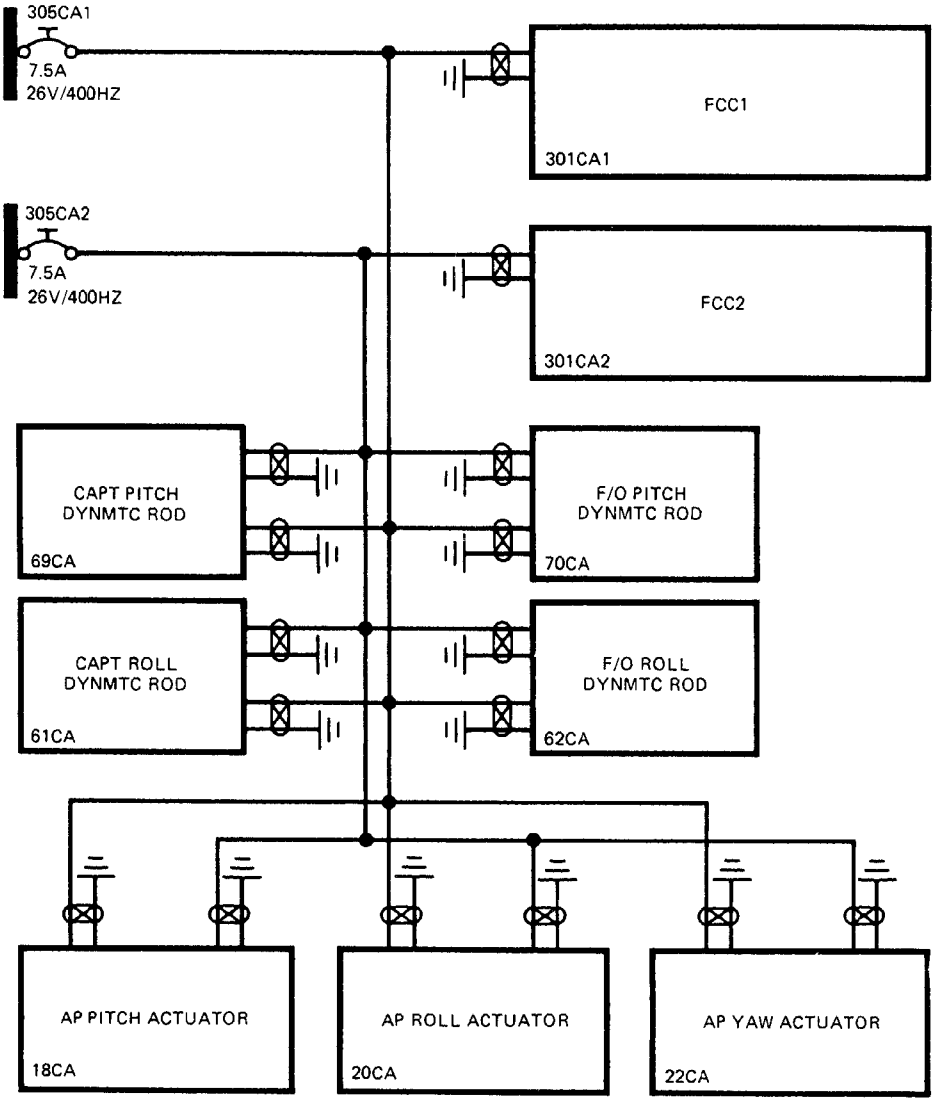
The 26V/400 Hz monitoring protects the system of short 26V cut off periods and enables system disengagement in the event of permanent loss of the 26V/400 Hz power supply.
To this end a ± 10 % calibrated detector on the 8V/400 Hz line generates a warning signal.
During the cut off period this signal ensures that the servomotor data are no longer taken into account.
For this purpose :
- the servovalve signals are cut off
- the power comparators are forced to good state.

This is carried out by :
- a signal transmitted to the central unit (inputs, synchros... are not taken into account)
- a relay controlling the servovalve channel opening
- a circuit applying a - 15V at each input of the power comparators.

Beyond a certain period of time the system is disconnected.

B. Internal Power Supply Automatic Cut Off.

To improve the computer MTBF, their internal power supply is automatically cut off when the aircraft is on ground.
The automatic cut off and restoration principle of the internal power supply is described in chapter 22-00.



BM5 22 12 00 0 AEM2 09 04
DM5

26V/400 Hz - Power Supply
Figure 002

2. Hydraulic Power Supply

The AP pitch, roll and yaw actuators are powered by the Green hydraulic system for side 1 and the Yellow hydraulic system for side 2.
The actuators are effectively powered only when the SERVO CTLS pushbutton switches are selected ON (switch pressed in).
Two pressure switches provide the AP engage logic with Green and Yellow pressure validation signals and give a ground signal when pressure is higher than the calibration pressure.
The hydraulic power supply principle is described in 22-00.

1. Main Flight Controls

A. AP Actuator

The AP commands are followed up by three identical dual electro-hydraulic actuators which drive pitch, roll and yaw flight controls through torque limiter levers. The face features holes accommodating special pins when adjustment of flight controls is carried out.

(1)Description

Each actuator consists of two identical lanes, one per side (AP1, AP2). Each lane comprises :

- one hydraulic pressure inlet electrovalve (main valve)
- one engagement electrovalve with electrical contacts for confirmation of engagement
- one servovalve
- one actuator body with chambers, pistons, clutch mechanisms, mechanical linkage, pipes and filters
- position feedback synchros (FU1, FU2, FUD).

(2)Operation

- At AP engagement the main valve and the engagement valve are energized if the engage logic conditions of the corresponding system are met (logic circuits are located in the corresponding FCC).
  When the engagement valve is energized, the associated slide valve is displaced in direction of arrow and drives the cam.
  When engagement is effective, (confirmed by closed microswitches) pivoting pin of bellcrank is in contact with cam slot and seesaw motions of the working pistons are strictly followed by the output shaft and lever. The engagement status of microswitches is taken into account in the system engage logic.
- The servovalve transforms control electrical orders from FCC into pressure orders ensuring output lever positioning.
  The output shaft is thus controlled either by side 1 (when AP1 is engaged) or side 2 (when AP2 is engaged).
  When both APs are simultaneously engaged, operation is possible even if position differences of pistons exist between side 1 and side 2, owing to the profile of the engagement cams. The side 1 cam features a deeper gradient than that of side 2, thus ensuring mechanical priority.
  In the case of loss of side 1, an automatic change-over to side 2 is carried out, ensuring system survival. In principle this change-over is smoothly carried out as the standby side (side 2) is synchronized with the active side (side 1).
  The actuator is physically limited in speed (total travel in 1.6 s).

B. Torque Limiter Levers

A torque limiter lever is installed between the output shaft of each actuator and the corresponding flight control channel. The torque limiter levers are different for the three axes (pitch, roll and yaw). It allows for mechanical override of the FCC command throughout the flight envelope by disconnecting the actuator from the flight control kinematics.

However AP remains engaged but inoperative. When override effort is relieved, the AP is reactivated.
The operation principle is as follows :
- beyond a certain load threshold, a ball and cam device triggers and the actuator output shaft separates from the corresponding flight control. The approximate triggering thresholds (from neutral position of flight controls), expressed in loads to be applied to the control column, handwheel, pedals, are as follows :

- pitch : 20 kgf nose-down _! at low
          46 kgf nose-up   _! speed

- roll : 15 kgf

- yaw   : 65 kgf

**ON A/C ALL

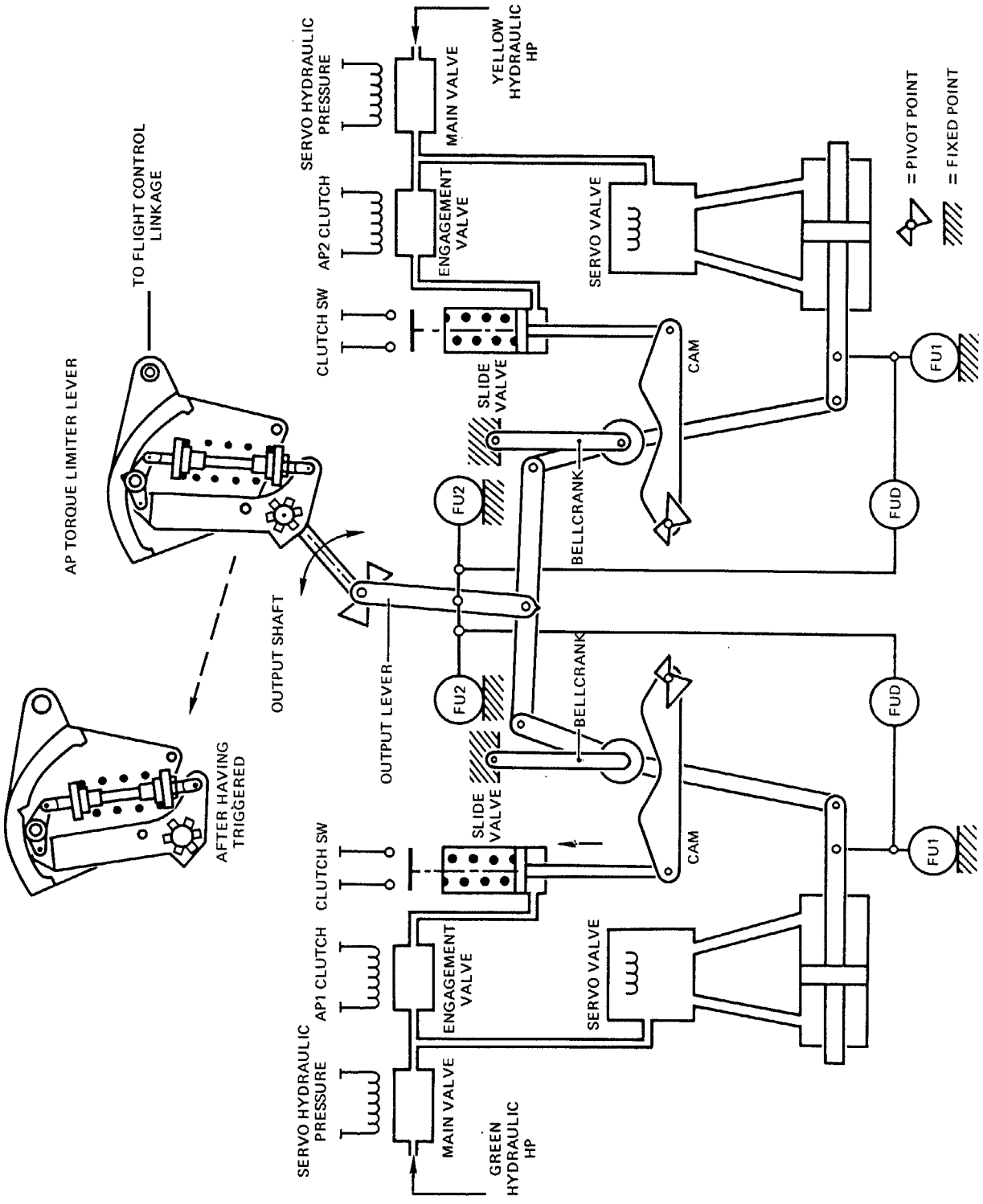
R

NOTE : The pitch torque limiter lever only is fitted with a microswitch whose action causes the automatic change over on TRIM 2 by disengaging TRIM 1 (thus the crew has an additional means for preventing a trim runaway when the AP is engaged) and AP CMD disconnection in case of pitch actuator override.

**ON A/C ALL

(Ref. Fig. 001)

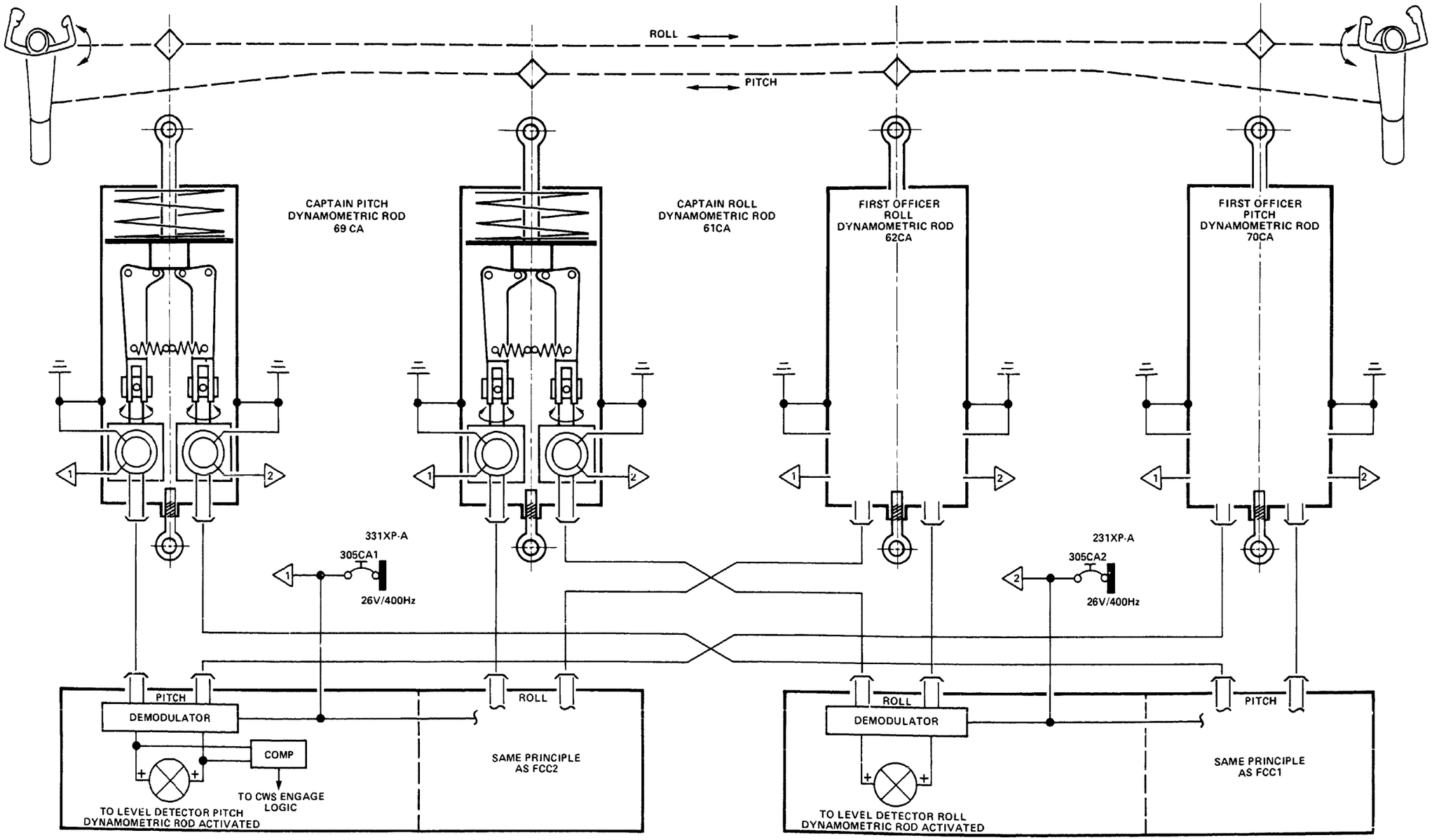
C. Dynamometric Rods
For the control wheel steering function, the FCC uses signals from dynamometric rods which represents the Captain's and First Officer's loads on the controls.
There are two rods for the pitch axis and two for the roll axis.
Each dynamometric rod includes two synchros (one associated with FCC1, the other with FCC2). There is no rod on the yaw axis. The rods are placed in series in the flight control linkage.
The measuring range is ±20 daN for a maximum extension of ±0.5 mm. The transmitted signal is approximately proportional to the load applied.
The rods are supplied with a minor tooling (jumper) used to lock them in their neutral position (effort null). This facilitates length adjustment when the rods are installed.
A discrete signal (ground) enables the FCC to make certain of the correct connection of the sensors (CWS sensor connected signal). The pitch dynamometric rods enable the AP disconnection except in GO AROUND, LAND TRACK or GLIDE CAPTURE modes.
It can be noticed that the First Officer's roll dynamometric rod wiring is inverted with respect to that of the Captain's roll dynamometric rod. As the loads detected in each rod are summed up in the FCC's, this enables,



BM5 22 13 00 0 ACM2 09 04
DM5

Autopilot Actuator
Figure 001

taking into account the principle of the roll flight control operation, the signals to be cancelled when the AP is operative and no loads are applied to the aileron control wheels. On the contrary, when the pilot applies a load, in CWS mode, to the control wheel, the rod which bears against the AP actuator detects the load exerted by the pilot (the actuator representing a "friction point"), the other rod detects a negligible load (no friction point), and the sum of the signals performed in the FCC's represents the actual load applied by the pilot.
(Ref. Fig. 002)



BM5 22 13 00 0 ADM2 13-06
DM5

Dynamometric Rods and FCC's Interconnection
Figure 002

2. Secondary Flight Controls

The flap and slat configuration data is transmitted to the FCC by the SFCC and then sent to the other AFS computers. The flap and slat control lever position is taken into account by the FCC for :

- the engagement of AP yaw axis when the slats are extended
- the authorization of the GO AROUND and TAKEOFF mode engagement when slats are extended.
- the modification of the piloting laws with respect to flap and slat extension.

The main landing gear information is transmitted to the FCC via bus connnections between the FACs and the FCCs.

The FACs monitor the landing gear relay supply : in the event of a detected supply failure the FACs send a warning signal thus causing the FCC loss.

Each FCC receives directly the nose landing gear information.

The FCC monitors landing gear data validity :

- a disagreement between the main landing gear (L and R) data during 10 seconds results in loss of the FCC
- a disagreement between the nose and main landing gear data results in loss of the FCC.

The nose landing gear data is used to authorize ground tests.

The main landing gear data is used for :

- distinction between takeoff and go around modes
- NAV mode engagement logic
- inhibition of AP engagement in CMD mode on the ground with engines running
- piloting laws (CWS and TO/GA).

The wheel speed information greater than 70 kts (directly linked to FCCs) is used for ROLL OUT mode engagement.

Each FCC is connected to both FACS. The logic of the FAC bus selection by the FCCs depends on the engagement of pitch trim and yaw dampers.

AUTOPILOT PITCH ACTUATOR - REMOVAL/INSTALLATION

WARNING: CHECK THAT THE LANDING GEAR GROUND SAFETIES INCLUDING WHEEL CHOCKS ARE IN POSITION.
BEFORE APPLYING OR RELIEVING HYDRAULIC SYSTEM PRESSURE, MAKE SURE THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, CHECK THAT ALL CONTROLS ARE SET TO CORRESPOND WITH THE ACTUAL POSITION OF THE SERVICES THEY OPERATE.
BEFORE PROCEEDING WITH MAINTENANCE WORK ON OR NEAR MECHANICAL FLIGHT CONTROLS OR PRIMARY FLIGHT CONTROL SURFACES, LANDING GEARS, ASSOCITED DOORS OR ANY MOVING COMPONENT, MAKE SURE THAT GROUND SAFETIES AND/OR WARNING NOTICES ARE IN CORRECT POSITION TO PREVENT INADVERTENT OPERATION OF THE CONTROLS.
ELECTRICAL POWER - BEFORE POWER IS SUPLIED TO THE AIRCRAFT MAKE SURE THAT ANY ELECTRICAL CIRCUIT UPON WHICH WORK IS IN PROGRESS IS ISOLATED.

1. Autopilot Pitch Actuator

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 8.30 m (27 ft.)
B. 98A27207543000	Pin - Rigging, Auto-Pilot, Servo-Motor
C. 98A27307549000	Pin - Rigging, Elevator Control Bellcranks
D. 97A27302009000	Locking Equipment - Pitch Trim Control
E. 98A27303001000	Tool - Locking Elevator (optional)
F.	Torque Wrench, up to 0.1 m.daN (8.85 lbf.in.)
G.	Circuit Breaker Safety Clips
H.	Warning Notices
J.	Electrical Ground Power Unit 3-Phase, 115/220 V, 400 Hz
K.	Corrosion-Resistant Steel Lockwire, 0.8 mm (0.032 in.) dia.
L.	Cotter Pins
M.	Shims
N. Material No. 05-005	Special Materials (Ref. 20-31-00)
P. Material No. 05-027	Special Materials (Ref. 20-31-00)
Q. Material No. 09-018	Sealants (Ref. 20-31-00)
R. Material No. 11-004	Cleaning Agents (Ref. 20-31-00)
Referenced Procedures:	
- 20-28-11, P. Block 1	Electrical Bonding
- 22-10-00, P. Block 501	Autopilot
- 22-13-24, P. Block 401	Pitch Torque Limiter Lever
- 24-41-00, P. Block 301	AC External Power Control
- 27-35-00, P. Block 501	Control Surface Position Indicating (Elevtor)
- 29-10-00, P. Block 301	Main Auxiliary Power - Pressurization/

ITEM	DESIGNATION
	Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power - Servicing
- 29-22-00, P. Block 301	Blue Auxiliary Power - Servicing
- 29-23-00, P. Block 301	Yellow Auxiliary Power - Servicing
- 51-75-10, P. Block 301	Repair of Paint Coatings
- 55-13-13, P. Block 401	Trailing Edge Access Panels

B. Procedure

R NOTE: Obey the AMM 20-17-00 instructions to prevent damage to the equipment
R when you do this task.

(1)Job Set-Up (Ref. Fig. 402)

(a)Put the access platform.
(b)Make sure that the pitch trim control wheels and control column are set at zero and install the locking equipment P/N 97A27302009000.
R (c)Make sure that the elevator is set at zero position. (Ref. 27-35-00,
R P. Block 501).

R NOTE: The elevator can also be set to zero position with the
R locking elevator optional-tool P/N 98A27303001000.

R (d)Put the warning notices in the flight compartment to tell persons not
R to operate the flight controls.
R (e)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AP1 FD1 FCC1/26VAC	305CA1	102/H02
21VU	AP1 FD1 FCC1/28VDC	306CA1	102/H03
21VU	AFS AP1 FD1 FCU	312CA1	102/H04
21VU	AFS AP2 FD2 FCU	312CA2	102/H05
21VU	AFS AP2 FD2 FCC2/28VDC	306CA2	102/H06
21VU	AFS AP2 FD2 FCC2/26VAC	305CA2	102/H07
21VU	AFS MTP	301CU	102/H01

(f)Remove the access panels 334AB/AT and 344AB/AT (Ref. 55-13-13, P. Block 401).
R (g)Depressurize the green and yellow hydraulic systems (Ref. 29-21-00,
R P. Block 301) (Ref. 29-23-00, P. Block 301).
R (h)Put the warning notices to tell persons not to pressurize the
R the hydraulic systems.
R (j)Install the rigging pin P/N 98A27307549000 in the bellcrank at
R RIB4 and RIB9 servo-controls (Ref. Fig. 401).

(2)Removal

(a)Disconnect the electrical connectors, (1) and (12), and install the blanking caps.
(b)Disconnect the hydraulic lines (11), and install the blanking plugs.
(c)Remove the upper and lower reducers (20) and discard the packing (21).

NOTE: Keep the upper and lower reducers autopilot (20) of the removed autopilot pitch actuator (13) to be used again in the replacement.

(d)Remove the nut (14) and bolt (16), and disconnect the bonding strap (15).
(e)Remove the pushrod (10) from the pitch torque limiter lever (17).
1 Remove the cotter pin (6), nut (7) and washer (5).
2 Remove the bolt (2), spacer (4), disconnect the pushrod (10) and secure to the structure.
(f)Removal of the autopilot pitch actuator (13) (Ref. Fig. 401)
1 Put and remove the lockwire, and remove the bolts (28).
2 Remove the covers (29).

CAUTION: AUTOPILOT PITCH ACTUATOR WEIGHS APPROX. 20 kg (44 lb.).

3 Support the autopilot pitch actuator (13), and remove the pins (30).
4 Remove the autopilot pitch actuator (13).

5 Remove the bushes (31).

(3)Preparation of the Replacement Component

(a)Transfer the pitch torque limiter lever (17) from removed to replacement component (Ref. 22-13-24, P. Block 401).
(b)Transfer the upper and lower reducers (20) from the removed component to the replacement component, and install the new packing (21).
(c)Clean all bolts and bushes with cleaning agent (Mat. No. 11-004).

(4)Installation

(a)Installation of the autopilot pitch actuator (13).(Ref. Fig. 401)
1 Apply a coat of sealant (Material No. 09-018) to the mating surfaces of the bushes (31) and actuator attach fittings.
2 Install the bushes (31).

NOTE: Do the installation of the autopilot pitch actuator with two persons:
- one of them holds the autopilot pitch actuator from below
- another attaches it.

3 Position the autopilot pitch actuator (13).
4 Install the pins (30).
5 Position the covers (29), and install the bolts (28).
6 Safety the bolts (28) with lockwire.

CAUTION: MAKE SURE THAT YOU CONNECT THE HYDRAULIC LINES TO THE CORRECT HYDRAULIC CONNECTIONS ON THE ACTUATOR. IF YOU DO NOT CONNECT THE LINES CORRECTLY, THE TEST (AFTER YOU HAVE COMPLETED THE INSTALLATION) WILL FAIL.

R (b)Remove the blanking caps and connect the electrical connectors, (1) and (12).
R (c)Apply corrosion-preventive compound (Mat. No. 05-005) to all bolt heads, protruding washers and nuts.
R (d)Make sure that the hydraulic systems are depressurized. (Ref. 29-21-00, P. Block 301) (Ref. 29-22-00, P. Block 301) (Ref. 29-23-00, P. Block 301)
R (e)Install the rigging-pin P/N 98A27207543000.
R (f)Installation of the pushrod (10).
1 Position the pushrod (10) to pitch torque limiter levers (17), install the spacer (4) and bolt (2).
2 If bolt (2) cannot be easily inserted, adjust pushrod (10) as follows:
a Cut and remove the lockwire and loosen the nuts (8).
b Adjust the length of the pushrod (10) with the adjuster (9) until the bolt (2) can be easily inserted. Make sure that the adjuster (9) is in safety.
c Tighten the nuts (8) and safety with lockwire to the lockwasher (9).
3 Install the washer (5) and nut (7).
4 TORQUE the nuts (7) to between 0.08 and 0.095 m.daN (7 and 8 lbf.in.) and safety with the cotter pin (6).
R (g)Connect the bonding strap (15), and install the bolt (16) and nut (14). (Ref. 20-28-11, P. Block 1)
R (h)Remove the blanking plugs, and connect and tighten the hydraulic lines (11).
R (j)Remove the rigging pins, P/N 98A27307549000 and P/N 98A27207543000.

R NOTE: If necessary, remove the locking elevator optional-tool P/N 98A27303001000.

R (k)Pressurize the hydraulic systems. (Ref. 29-21-00, P. Block 301) (Ref. 29-22-00, P. Block 301) (Ref. 29-23-00, P. Block 301)
R (l)Remove the safety clips and tags, and close the circuit breakers previously open.
R (m)Make sure that the control surface travel ranges are clear.
R (n)Check that the hydraulic connections are free from leaks.

(5)Test

(a)Initial conditions :

1 Electrical power supply correct (115VAC, 28VDC, 26VAC).
2 Forward services ventilation system in operation.
3 AFS, NAVIGATION, ADS and ECAM circuit breakers on overhead panel closed.
4 Aircraft in ground configuration (LDG GEAR PROX DET & RELAYS circuit breakers on panel 133VU closed).
5 ADS operating.
6 IRS 1, 2 and 3 aligned (perform ALIGN procedure until red HDG flags on NDS disappear, present position entering is not necessary).
7 EFIS operating (adjust PFD and ND brightness on panels 412VU and 413VU).
8 On center pedestal, ECAM display unit brightness adjusted and FLT CTL page selected.
9 PITCH FEEL and RUDDER TRAVEL systems ON.
10 PITCH TRIM 1 system ON.
11 YAW DAMPER 1 system ON.

(b) Do the test that follows, to make sure that the electrical components were installed correclty during the removal/installation procedure.

ACTION	RESULT
1. On panel 21VU: - open the AP2-FD2/FCC2/28VDC circuit breaker 306CA2 (line 102/H6)	- the FCC2 is de-energized
2. On the flight control unit (FCU): - put the AP1 lever in the ON position. - push (in) the CWS/CMD p/bsw. - if the AP1 lever does not stay engaged, stop the test and check electrical connections.	- the AP1 lever is engaged - the AP1 lever stays engaged - CMD lights ON
NOTE: If AP is not in V/S mode, select altitude of 5000 ft. on FCU and pull V/S knob.	
3. On flight control unit (FCU): - with V/S selector knob, select positive vertical speed (Vz = +300 ft/min).	- control column moves in nose up direction. - pitch trim control wheel rotates in nose up direction.
- select zero vertical speed position.	- control column returns to neutral

ACTION	RESULT
4. Disengage the AP1 lever	- the AP1 lever is disengaged
5. On panel 21VU: - close the AP2-FD2/FCC2/28VDC circuit breaker 306CA2 line(102/H6) - open the AP1-FD1/FCC1/28VDC circuit breaker 306CA1 line(102/H3)	 - the FCC2 is enegized. - the FCC1 is de-enegized.
6. On the flight control unit (FCU): - put the AP2 lever in the ON position. - push (in) the CWS/CMD p/bsw. - if the AP2 lever does not stay engaged, stop the test and check electrical connections.	 - the AP2 lever is engaged - the AP2 lever stays engaged - CMD lights ON
7. On flight control unit (FCU): - with V/S selector knob, select positive vertical speed (Vz = +300 ft/min). - select zero vertical speed position.	 - control column moves in nose up direction. - pitch trim control wheel rotates in nose up direction. - control column returns to neutral
8. Disengage the AP2 lever	- the AP2 lever is disengaged
9. On panel 21VU: - close the AP1-FD1/FCC1/28VDC circuit breaker 306CA2 line(102/H6)	 - the FCC1 is enegized.

(c)Do the test that follows, to make sure that the hydraulic lines to the actuator were connected correctly during the removal/installation procedure.

NOTE: Failure of the correctly engange the autopilot systems 1 and 2 in the comand (CMD) mode during this test will show that the hydraulic lines are not connected correctly.

ACTION	RESULT

NOTE: Hydraulic presssure initial configuration:
- Blue system available for yaw damper 1 engagement (Ref. 29-22-00 p. block 301)
- Green system available for AP1 engagement (Ref. 29-21-00 p. block 301)

ACTION	RESULT
- Yellow system must be isolated to check the hydraulic line connection.	
1. On the flight control unit (FCU): - put the AP1 lever in the ON position. - push (in) the CWS/CMD p/bsw. - if the AP1 lever does not stay engaged, stop the test and do the procedure to correct the connections of the hydraulic lines (Ref. para.2.B.(6)).	- the AP1 lever is engaged - the AP1 lever stays engaged
2. Disengage the AP1 lever	- the AP1 lever is disengaged
3. Change the hydraulic presssure configuration: - yellow system available for yaw damper 2 and AP2 engagement (Ref. 29-23-00 p. block 301). - green system must be isolated to check the correct hydraulic line connection. - green system can be isolated at the servo control panel if the yellow system is pressurized through the PTU.	
4. On the flight control unit (FCU): - put the AP2 lever in the ON position. - push (in) the CWS/CMD p/bsw. - if the AP2 lever does not stay engaged, stop the test and do the procedure to correct the connections of the hydraulic lines (Ref. para. 2.B.(6)).	- the AP2 lever is engaged - the AP2 lever stays engaged
5. Disengage the AP2 lever.	- the AP2 lever is disengaged
(d)Do the test that follows, to make sure that the elevators do not move when the autopilot is disenganged	

ACTION	RESULT
1. On SERVO CONTROL section of panel 428VU: - release Y pushbutton switch.	- Corresponding OFF legend comes

ACTION	RESULT
	on. - Corresponding SERVO CTL/LO PR/Y warning light comes on.
- press G pushbutton switch.	- Corresponding ON legend comes on.
2. On the flight control unit (FCU): - place the AP1 lever in the ON position.	- Lever remains in this position: - on FCU green CWS comes on. - on PFDs ambert CWS1 indication appears.
- press CWS/CMD pushbutton switch.	- On the CWS/CMD pushbutton switch: - green CWS goes off. - green CMD comes on. - On PFDs: - white CMD1 indications appears. - amber CWS1 indication disappears. - green V/S and HDG are displayed.
NOTE: If AP is not in V/S mode, select altitude of 5000 ft. on FCU and pull V/S knob.	
3. On flight control unit (FCU): - with V/S selector knob, select positive vertical speed (Vz = +300 ft/min).	- control column moves in nose up direction. - pitch trim control wheel rotates in nose up direction.
- select zero vertical speed position.	- control column returns to neutral
NOTE: Put one person in position adjacent to each elevator when you do step 4.	
4. Disengage the AP1 lever and cancel the warning.	- Check the the elevator surfaces do not move.
5. On SERVO CONTROL section of panel 428VU: - press Y pushbutton switch.	On panel 428VU: - SERVO CTL/LO PR/Y warning light goes off. - On SERVO CTL/Y pushbutton

ACTION	RESULT
- release G pushbutton switch.	switch OFF legend goes off. - corresponding OFF legend comes on. - corresponding SERVO CTL/LO PR warning light comes on.
6. On flight control unit (FCU): - engage AP2 in CMD.	On PFDs: - white CMD2 is displayed. - green V/S and HDG are - displayed
7. Carry out same operations as in step 3.	Same results as in step 3.
NOTE: Put one person in position adjacent to each elevator when you do step 9.	
8. Disengage AP2 and cancel warning.	- check for absence of elevator surfaces movement.
9. On SERVO CONTROL section of panel 428VU: - press G pushbutton switch.	On panel 428VU: - SERVO CTL/LO PR/G warning ligth goes off. - On SERVO CTL/G pushbutton switch OFF legend goes off.
10. On overhead panel: - place PITCH TRIM 1 lever in OFF position.	

(e) Carry out AP the pitch mechanical override test (Ref. 22-10-00, P. Block 501).

(f) Correction of the hydraulic connections.

NOTE: Use this procedure to correct the hydraulic line connections to the pitch actuator when the test to engage the autopilot systems 1 and 2 in the command mode fails.

1 Depressurize the green, yellow and blue hydraulic systems (Ref. 29-10-00 P. Block 301)
2 Open, safety and tag the circuit breakers 305CA1, 306CA1, 312CA1, 312CA2, 306CA2, 305CA2 and 301CU (Ref. 1. B. (1) (d)).
3 Place a container under the hydraulic connections.
4 Identify and mark lines and connections for correct

installation. Disconnet the hydraulic delivery and return lines (11) from the actuator.

CAUTION: MAKE SURE THAT YOU CONNECT THE HYDRAULIC LINES TO THE CORRECT HYDRAULIC CONNECTIONS ON THE ACTUATOR. IF YOU DO NOT CONNECT THE LINES CORRECTLY, THE TEST (AFTER YOU HAVE COMPLETED THIS PROCEDURE) WILL FAIL AGAIN

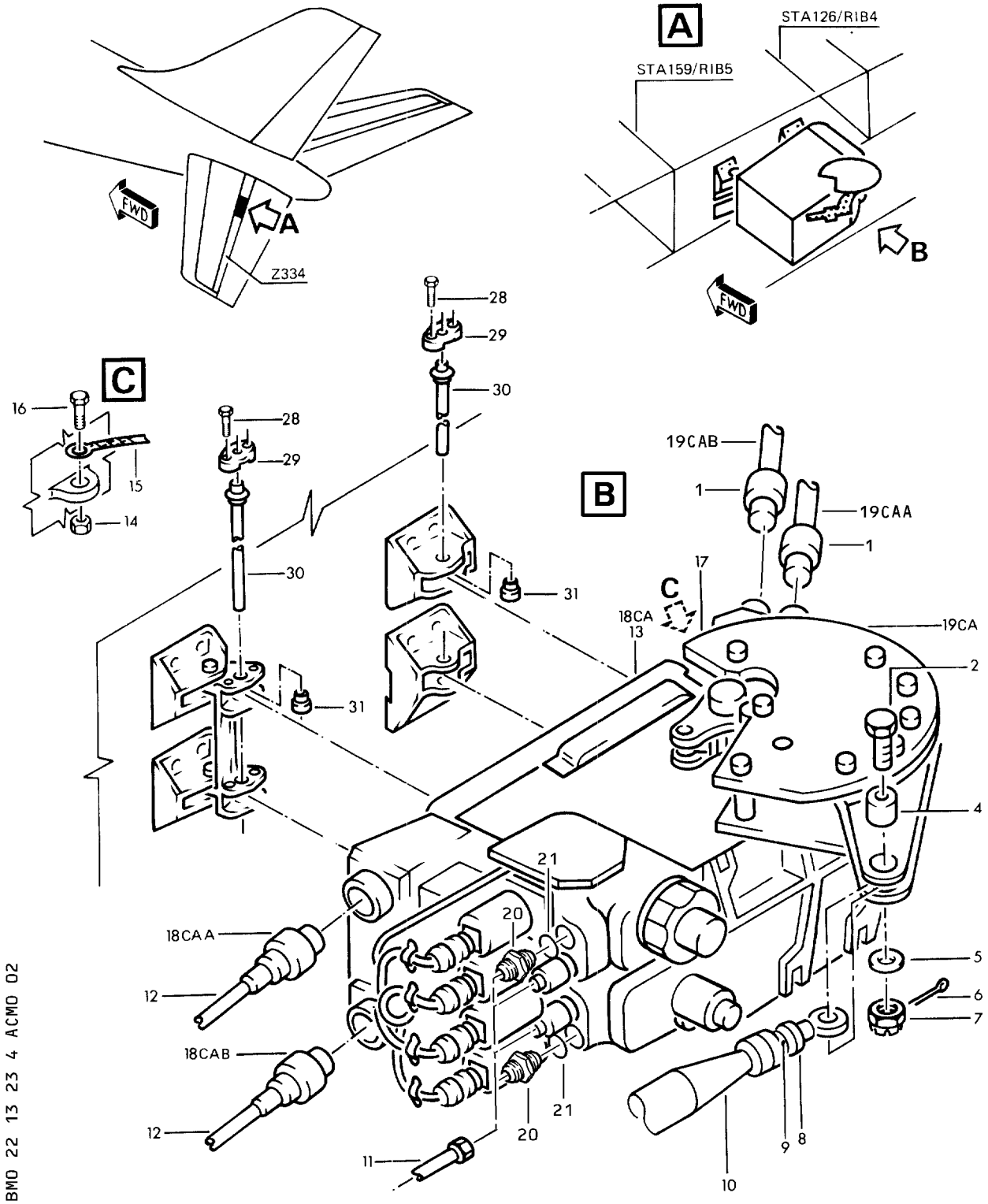
5 Connect the hydraulic and delivery return lines (11) to the correct positions on the actuator
6 Tighten the hydraulic lines (Ref. 20-23-12 P. Block 1).
7 Remove the container from under the hydraulic connetions.
8 Remove safety clips and tag and close the circuit breakers previously open. (Ref. 1.B. (1) (d)).
9 Make sure that the control ranges are clear and pressurize green, yellow and blue hydraulic systems (Ref. 29-10-00, P. Block 301).
10 Do the test, Ref 1. B. (5) (c), to make sure that the hydraulic are connected correctly

(6)Test completion:

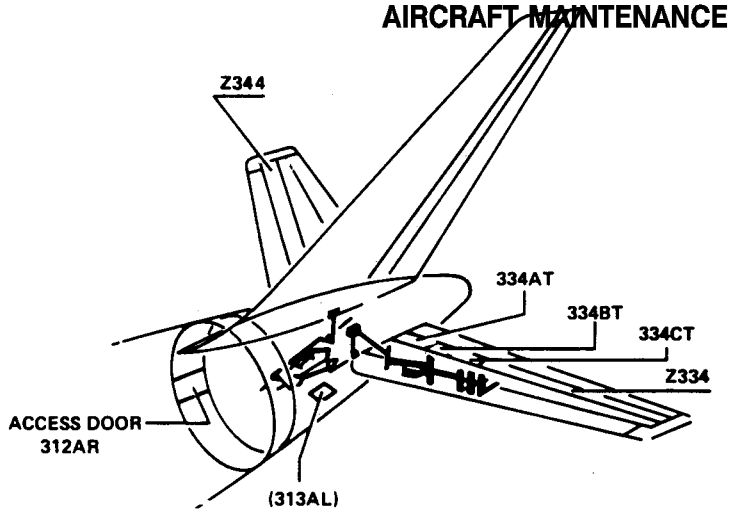
(a)Shut down and disconnect the hydraulic ground power cart. (Ref. 29-10-00 P. Block 301).
(b)De-energize the aircraft electrical network and disconnect the electrical ground power unit (Ref. 24-41-00, P. Block 301).
(c)Restore the aircraft and system to normal operating condition.

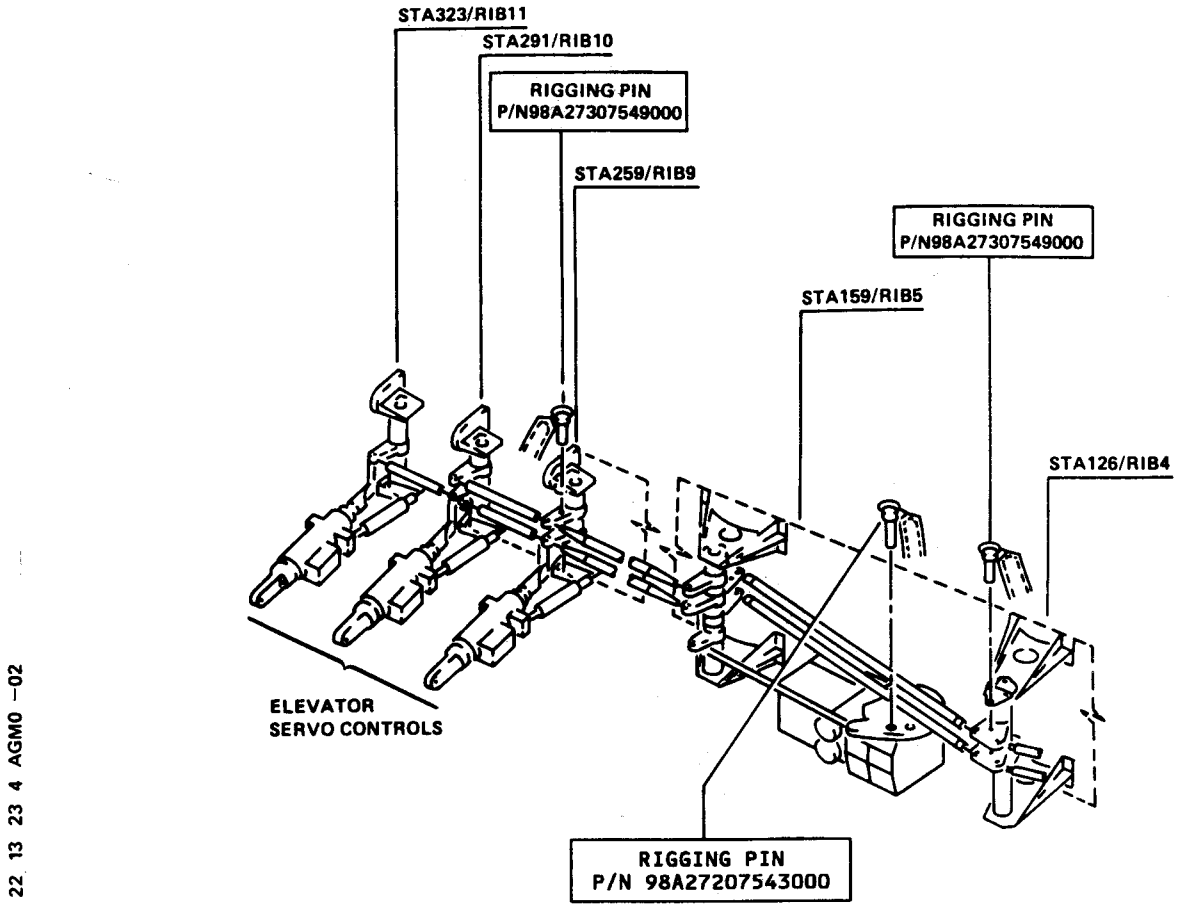
(7)Close-Up

(a)Make sure that the work area is clean and clear of tools and other items.
(b)Apply moisture-removing oil (Mat. No. 05-027) to the autopilot pitch actuator (Ref. 51-75-10, P. Block 801).
(c)Install the trailing-edge access-panels 334AB/AT, (Ref. 55-13-13, P. Block 401).
(d)Remove the access platform.
(e)Remove the warning notices.



Autopilot Pitch Actuator
Figure 401





Rigging Location
Figure 402

2. Servo-Valve, Pitch-Actuator

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 8.30 m (27 fl)
B. 98A27207543000	Pin Rigging - Auto Pilot Actuator
C. 98A27307549000	Pin Rigging - Elevator Control Bellcranks
D. 98A27302009000	Locking Equipment - Pitch Trim Control Whell
E. OU130560	Adaptor
F.	Torque Wrench 0 - 0.5 m.daN (0-3.69 lbf.ft)
G.	Circuit Breacker Safety Clips and Tags
H.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
J.	Warning Notice - Prohibiting Use of Hydraulic Systems
K.	Blanking Caps and Plugs - Electrical
L.	Brush
M. Material No. 04-015	Common Grease (Ref. 20-31-00)
N. Material No. 04-004	Common Grease (Ref. 20-31-00)
P. Material No. 05-005	Special Materials (Ref. 20-31-00)
Q. Material No. 05-027	Special Materials (Ref. 20-31-00)
R. Material No. 11-004	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures:	
- 22-13-23, P. Block 401	Autopilot Roll Actuator
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depreessurization)
- 51-75-10, P. Block 801	Repair Paint Coatings
- 55-13-13, P. Block 401	Trailing Edge Access Panels

B. Procedure

NOTE: The removal/installation procedure is the same for the LH or RH servo-valve (5).

(1) Job Set-Up

(a) Put the access platform in position below zone 334.
(b) Make sure that the pitch trim control-wheels and control column are set at zero and install the locking equipment PN 97A27302009000.
(c) Put the warning notice in the flight compartment prohibiting operation of the flight controls
(d) On panel 21VU, open, safety and tag the following AUTO FLIGHT SYSTEM circuit breakers:

PANEL	SERVICE	DENT.	LOCATION
21VU	AFS/MTP	301CA1	102/H01
21VU	AP1/FD1/FCC1/26VAC	305CA1	102/H02
21VU	AFS/FD1/FCC1/28VDC	306CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312CA1	102/H04
21VU	AFS/AP2/FCU	312CA2	102/H05
21VU	AFS/AP2/FD2/FCC2/28VDC	306CA2	102/H06
21VU	AFS/AP2/FCC2/26VAC	305CA2	102/H07

(e)Remove the access panel 334AB/AT and 344AB/AT (Ref. 55-13-13, P. Block 401).
(f)Install the rigging pin PN 98A27307549000 in the bellcrank at RIB4 and RIB9 servo-controls.
(g)Make sure that the riggin pin PN 98A27207543000 can be installed in the actuator (1). Remove the riggin pin inmediately after check.
(h)Put a warning notice in the flight compartment prohibiting use of the aircarft hydraulic systems and de-pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).

(2)Removal

CAUTION: TOOLS AND OTHER EQUIPMENT, INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69 mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE, STORE THE REMOVED SERVO-VALVE IN A PROTECTIVE COVER AND AWAY FROM MAGNETIC MATERIALS. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

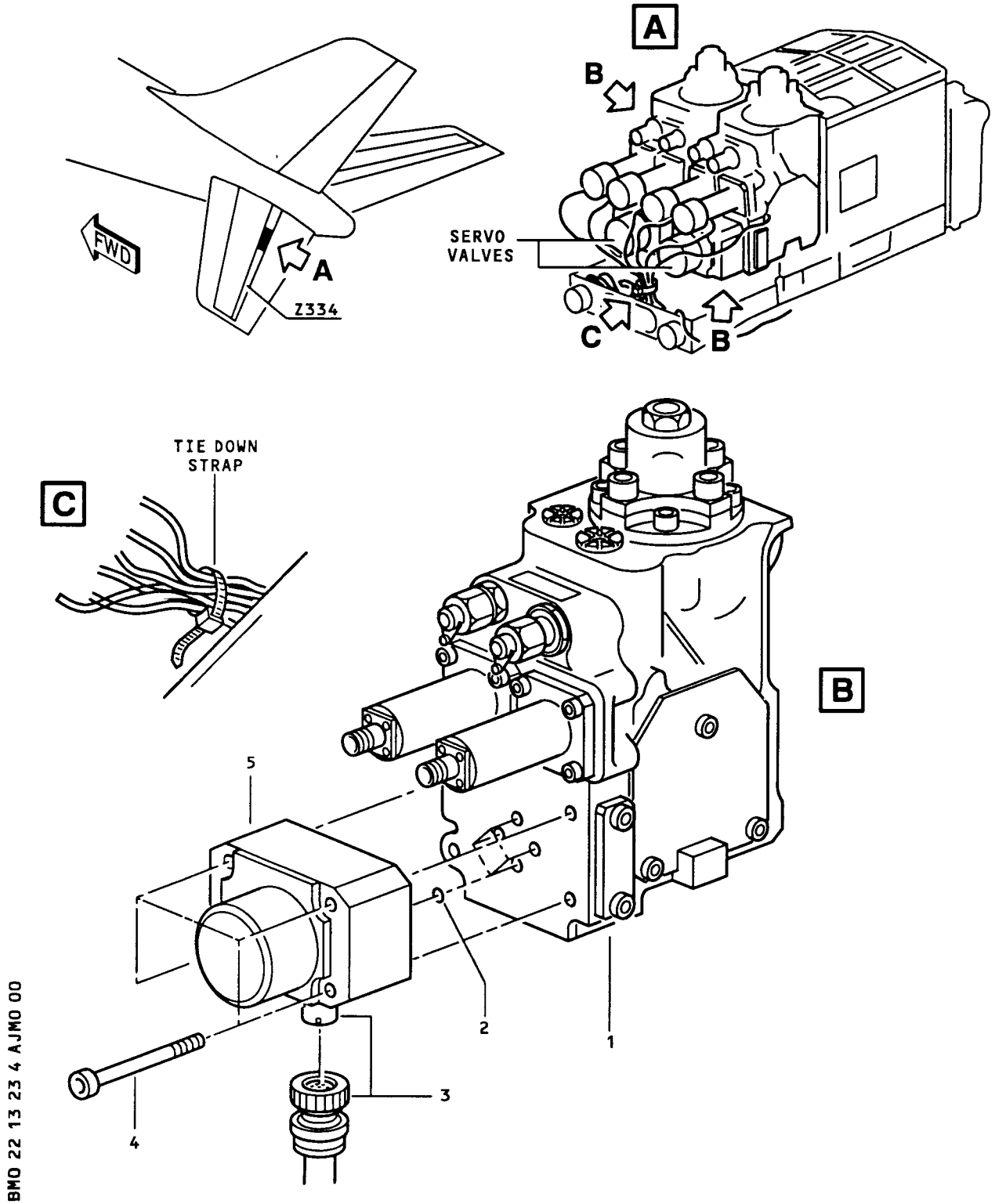
(Ref. Fig. 403)

CAUTION: DO NOT ALLOW DUST, DIRT OR METAL FILINGS TO CONTAMINATE THE INTERIOR PASSAGE OF THE SERVO-VALVE OR THE ROLL ACTUATOR MANIFOLD. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

(a)Remove and discard the tie down strap that holds the electrical
(b)Disconnect the applicable electrical connector (3) from the servo-valve (5).
(c)Cap the open end for the electrical connector (3).
(d)Remove the screws (4) from the applicable servo-valve (5).
(e)Carefully remove the servo-valve (5) from the actuator (1).
(f)Remove and discard the O-rings (2) and inmediately blank the open mainfold of the actuator (1).
(g)Put the removed servo-valve (5) into a protective cover.

(3)Preparation of Replacement Component

(a)Remove the protective cover from the servo-valve (5) and the blank



Autopilot Pitch Actuator, Servo-Valve
Figure 403

from the manifold face of the actuator (1).
(b)Clean the actuator (1) and servo-valve (5) manifold faces with a lint-free cloth moist in cleaning agent (Mat. No. 11-004) until contamination is removed. Do not let cleaning agent dry on surfaces.
(c)Apply a coat of common grease (Mat. No. 04-004) to the manifold faces of the actuator (1) and the servo-valve (5) and to the threads of the screws (4).

(4)Installation

(a)Install the rigging pin PN 98A27207543000 in the actuator (1).
(b)Put the new O-rings (2) in position on the mating face of the servo-valve (5).
(c)Put the servo-valve (5) in position on the actuator (1) and install the screws (4).
(d)Use the adaptor 0U130560, and TORQUE the screws (4) to between 0.39 and 0.49 m.daN (34.5 and 43.3 lbf.in).
(e)Remove the caps, examine the ends of the electrical connector (3) for correct condition then connect the electrical connector (3) for to the servo-valve (5).
(f)Put back the electrical harness of the actuator (1) between the servo-valve (4) and install a new tie down strap.

(g)Remove safety clips and tags and close circuit breakers previously open.
(h)Make sure that the control ranges are clear.
(j)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(k)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301).
(l)Remove the riggin pins 98A27207543000, 98A27307549000 and 97A27302009000.
(m)Make sure that the actuator (1) and the servo-valves (5) are free from hydraulic leaks.

(5)Test

(a)Do the test of the autopilot pitch-actuator system (Ref. 22-13-23, P. Block 401, only the test (5) (b)).

(6)Close-up

(a)De-pressurize the aircraft hydraulic system (Ref. 29-10-00, P. Block 301).
(b)De-energize aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c)On the actuator (1) and the servo-valve (5), examine the coating of the special material (Mat. No. 05-027) for good condition and apply again as necessary (Ref. 51-75-00, P. Block 801).
(d)Install the access panels 334AB/AT (Ref. 55-13-13, P. Block 401).
(e)Make sure that the work area is clean and clear of tools and

other items.
(f)Remove the access platform.
(g)Remove the warning notices.

3. Solenoid, Pitch Actuator

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 3 to 6 m (10 to 20 ft)
B. 98A27307549000	Pin - Rigging, Elevator Control Bellcranks
C. 97A27302009000	Pin Rigging - Pitch Trim Control Wheel
D. OU129455	Pre-cashing Bush
E. OU110423	Pre-cashing Bush
F. OU131396 Adaptor	Torque Wrench 0-0.6 m.daN (0 - 53 lbf.in).
G.	Circuit Breaker Safety Clips and Tags
H.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
J.	Warning Notice - Prohibiting Unauthorized Use of Hydraulic Systems
K.	Brush
L. Material No. 04-015	Common Grease (Ref. 20-31-00)
M. Material No. 04-004	Common Grease (Ref. 20-31-00)
N. Material No. 05-005	Special Materials (Ref. 20-31-00)
P. Material No. 05-027	Special Materials (Ref. 20-31-00)
Q. Material No. 11-004	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures:	
- 22-13-23, P. Block 401	Autopilot Roll Actuator
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization)
- 51-75-10, P. Block 801	Repair Paint Coatings
- 55-13-13, P. Block 401	Trailing Edge Access Panels

B. Procedure

(1)Job Set-Up

(a)Position the access platform.
(b)Make sure that the pitch trim control wheels and control column are set at zero and install locking equipment P/N 97A27302009000.
(c)Display warning notices in fligth compartment prohibiting operation of flight controls.
(d)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301CU	102/H01
21VU	AP1/FD1/FCC1/26VAC	305CA1	102/H02
21VU	AFS/FD1/FCC1/28VDC	306CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312CA1	102/H04
21VU	AFS/AP1/FO1/FCU	312CA2	102/H05
21VU	AFS/AP2/FD2/FCC2/28VDC	306CA2	102/H06

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP2/FD2/FCC2/26VAC	305CA2	102/H07

(e)Depressurize the hydraulic systems (Ref. 29-10-00, P. Block 301), and display warning notices, prohibiting repressurization.
(f)Remove the access panels 334AB/AT and 344AB/AT.
(g)Install the riggin pin P/N 98A27307549000 and 97A27302009000 and control wheels.
(h)Cautiuosly mark the solenoids to be removed and their applicable electrical connetors.

(2)Removal (Ref. Fig. 404)

**On solenoids with fixed flange

CAUTION: DO NOT REMOVE MORE THAN ONE SOLENOID FROM THE ACTUATOR OR DISCONNECT MORE THAN ONE ELECTRICAL CONNECTOR AT THE SAME TIME. YOU MUST INSTALL AGAIN OR REPLACE THE SOLENOID BEFORE YOU CONTINUE TO REMOVE MORE SOLENOIDS FROM THE ACTUATOR.
THIS IS TO PREVENT CROSS CONNECTION OF THE SOLENOIDS.

(a)Cut and remove the lockwire, and disconnect the electrical connector (1) and install blanking caps.

CAUTION: MAKE SURE THAT THE PUSHROD IS NOT PUSHED OUT BY INTERNAL RESIDUAL PRESSURE OF THE AUTOPILOT ACTUATOR.

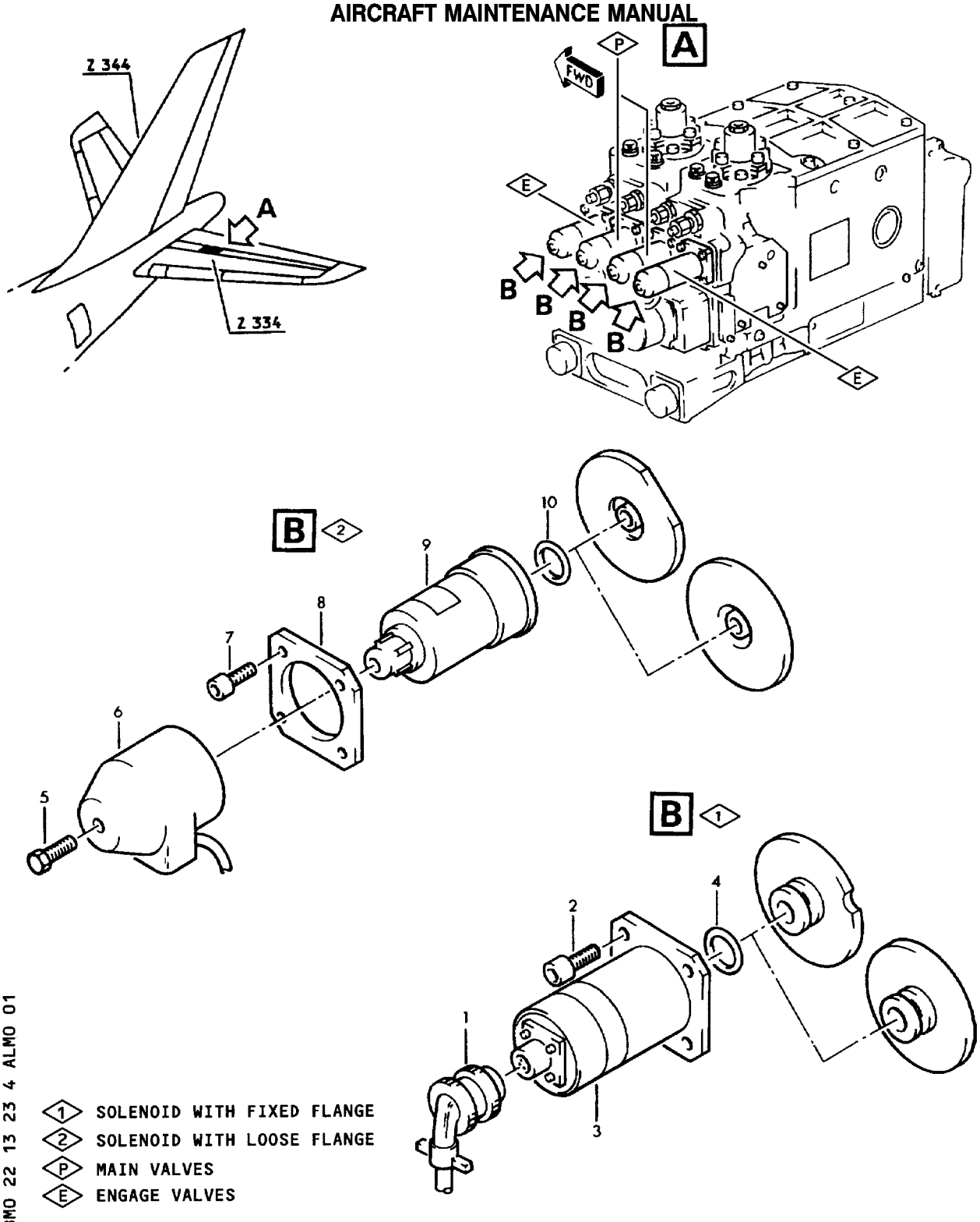
(b)Cut and remove the lockwire, and remove the screws (2) and solenoid (3).
(c)Remove and discard the packing (4).

**On solenoids with loose flange
(2)Removal (Ref. Fig. 404)

CAUTION: DO NOT REMOVE MORE THAN ONE SOLENOID FROM THE ACTUATOR OR DISCONNECT MORE THAN ONE ELECTRICAL CONNECTOR AT THE SAME TIME. YOU MUST INSTALL AGAIN OR REPLACE THE SOLENOID BEFORE YOU CONTINUE TO REMOVE MORE SOLENOIDS FROM THE ACTUATOR.
THIS IS TO PREVENT CROSS CONNECTION OF THE SOLENOIDS.

(a)Cut and remove the lockwire, and remove the screw (5)
(b)Disconnect the electrical connector (6), and install the blanking caps.

CAUTION: MAKE SURE THAT THE PUSHROD IS NOT PUSHED OUT BY INTERNAL RESIDUAL PRESSURE OF THE AUTOPILOT ACTUATOR.



Autopilot Actuator Solenoids
Figure 404

(c)Cut and remove the lockwire, and remove the screws (7), support flange (8) and solenoid (9).
(d)Remove and discard the O-ring (10).

**On solenoids with fixed flange

(3)Installation (Ref. Fig. 404)

(a)Lubricate the new packing (4) with grease (Mat. No. 04-015).
(b)Apply a coat of common grease (Mat. No. 04-004) to the screws (2).
(c)Install the new packing (4) on the valve using the mounting cone PN 0U130411.
(d)Put the solenoid (3) over the valve and install the screws (2).
(e)Torque the screws (2) to between 0.22 and 0.29 m.daN (19.5 and 25.6 lbf.in) using the adaptor PN 0U130423.
(f)Safety the screws (2) with lockwire 0.8 mm (0.032 9 in.) dia.
(g)Use a brush to the apply special material (Mat. No. 05-005) to the heads of the screws (2).
(h)Remove blanking caps, connect the electrical connector (1) back to the applicable solenoid respecting the job set-up mark.
(i)Safety the electrical connector (1) with lockwire 0.6 mm (0.24 in.) dia.

**On solenoids with loose flange

(3)Installation (Ref. Fig. 404).

(a)Lubricate the new O-ring (10) with grease (Mat. No. 04-015).
(b)Apply a coat of common grease (mat. No. 04-004) to the screws (7) and (5).
(c)Install the new O-ring (10) to the valve.
(d)Put the solenoid (9) over the valve, and install the support flange (8) with screws (7).
(e)Torque the screws (7) to between 0.22 and 0.29 m.daN (19.5 and 25.6 lbf.in) using the adaptor PN 0U130423.
(f)Safety the screws (7) with lockwire 0.8 mm (0.032 in.) dia.
(g)Connect the electrical connector (6) back to the applicable solenoid respecting the job set-up mark, and install the screw (5).
(h)Safety the screw (5) with lockwire 0.8 mm (0.032 in.) dia.
(j)Use a brush to apply the special material (Mat. No. 05-005) to the heads of the screws (7) and (5).

**On A/C ALL

(k)Remove the safety clips and tags and close circuit breakers previously open.
(l)Make sure that the control-surface travel-ranges are clear.
(m)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301).
(n)Remove the riggin pins PN 98A27307549000, 97A27302009000
(p)Check that the hydraulic connections are free from leaks.

(4)Test

(a)Do the test of the autopilot pitch-actuator system (Ref. 22-13-23, P.Block 401, only the test (5) (b)).

(5)Close-up

(a)Make sure that the work area is clean and clear of tools and other items.
(b)On the autopilot pitch actuator, check the coating of moisture-removing oil (Mat. No. 05-027) for good condition and apply again as necessary (Ref. 51-75-10, P. Block 801).
(c)Close the access panels 344 AB/AT.
(d)Remove the warning notices.
(e)Remove the access platform.

4. Filter, Pitch Actuator

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 8.30 m (27 ft.)
B. 97A27302009000	Pin Rigging - Pitch Trim Control Wheel
C. 0U129493	Pre-casting Bush
D. 0U129485	Pre-casting Bush
E. 0U134344	Adaptor
F.	Torque Wrench 0 - 0.6 m.daN (0 - 53 lbf.in).
G.	Circuit Breaker Safety Clips and Tags
H.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
J.	Warning Notice - Prohibitng Unauthorized Use of Hidraulic System
K. Material No. 04-015	Common Grease (Ref. 20-31-00)
L. Material No. 04-004	Common Grease (Ref. 20-31-00)
M. Material No. 05-005	Special Materials (Ref. 20-31-00)
N. Material No. 05-027	Special Materials (Ref. 20-31-00)
P. Material No. 11-004	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures:	
- 22-13-23, P. Block 401	Autopilot Pitch Actuator
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization)
- 51-75-10, P. Block 801	Repair Paint Coatings
- 55-13-13, P. Block 401	Trailing Edge Access Panels

B. Procedure

NOTE : The removal/installation procedure is the same for the LH or RH filter (8).

(1)Job Set-up.

(a)Put the access platform in position below zone 334
(b)Make sure that the pitch trim is set at zero.
(c)Make sure that pitch trim control wheel is central and install the rigging pin 97A27302009000.
(d)Put warning notices in the flight compartment, prohibiting operation of the flight controls.
(e)Open safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	API/FD1/FCC1/28VAC	305CA1	102/H02
21VU	API/FD1/FCC1/28VDC	305CA1	102/H03
21VU	AFS/AP1-FD1/FCU	306CA1	102/H04
21VU	AFS/AP2/FD2/FCU	306CA2	102/H05

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP2/FD2/FCC2 28VDC	306CA2	102/H06
21VU	AFS/AP2/FP2/FCC2/26VAC	305CA2	102/H07
21VU	AFS MTP	301CU	102/H01

(f)Open the access panels 334 AB/AT and 344AB/AT (Ref. 55-13-13, P.Block 401).
(g)Put a warning notice in the flight compartment prohibiting use of the hydraulic systems.
(h)De-pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).

(2)Removal (Ref. Fig. 405)

CAUTION: TOOLS AND OTHER EQUIPMENTS; INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69 mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE.
DO NOT ALLOW DUST, DIRT OR METAL FILINGS TO CONTAMINATE THE FILTER SURFACES OR TO ENTER THE INTERIOR OF THE PITCH ACTUATOR.
FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

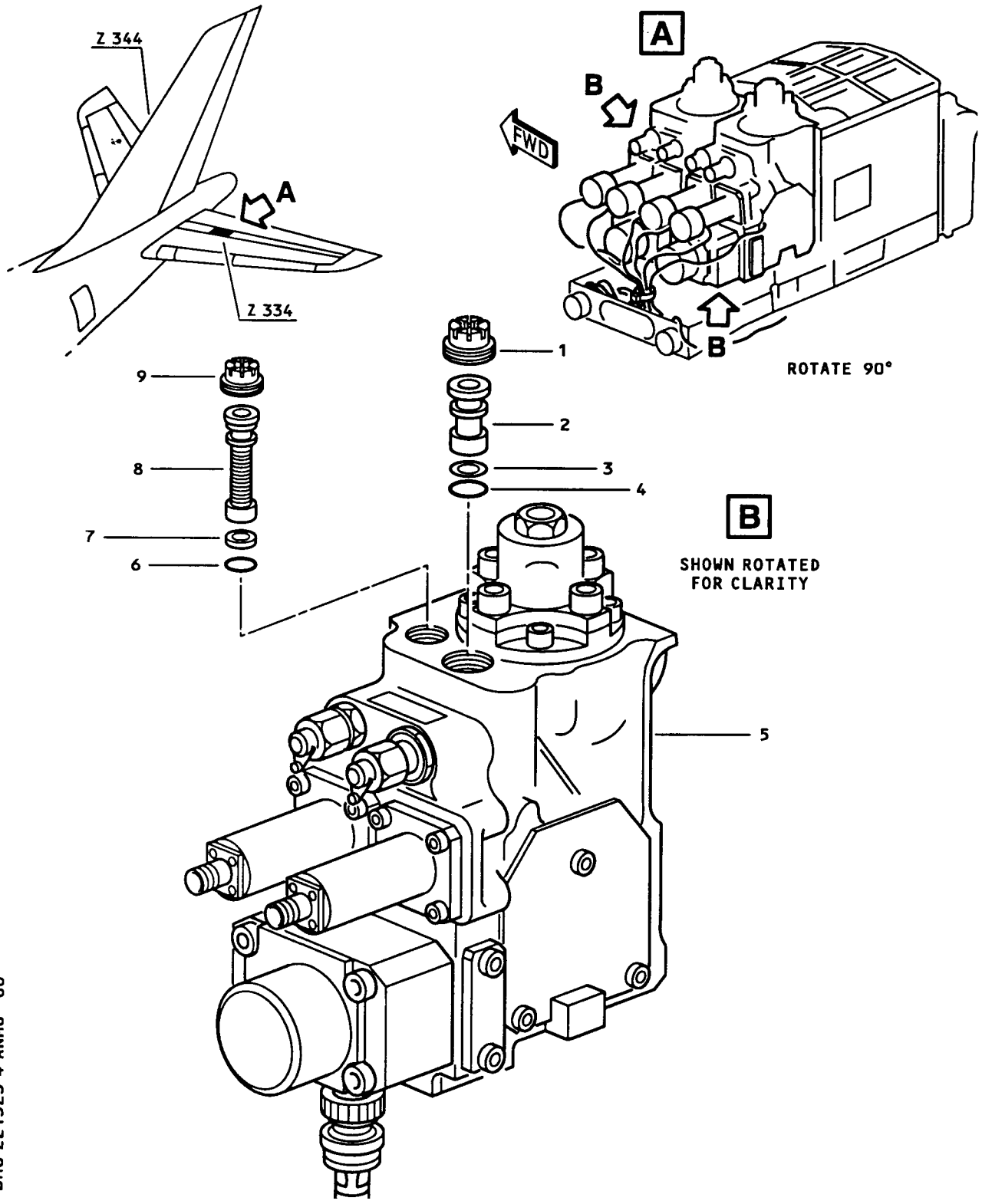
(a)Cut and remove the lockwire.
(b)Use the adaptor OU134344 to remove the nut (9).
(c)To remove the filter (8), install a threaded rod (8-32 UNC 3A) into the body of the filter (8) and remove it complete from the hydraulic unit (5). Remove the threaded rod from the filter (8).

(3)Preparation of Replacement Component.

(a)Clean the interface threads of the hydraulic unit (5) and the filter (8) with lint-free cloth moist in cleaning agent (Mat. No. 11-004) until contamination is removed. Do not let cleaning agent dry on the surfaces.
(b)Use common grease (Mat. No. 04-015) and lubricate the packing (6) and the O-ring (7).
(c)Use the pre-casting bush OU129493 and install the packing (6) on the filter (8).
(d)Use the pre-casting bush OU129485 and install the O-ring (7) on the filter (8).

(4)Installation

(a)Install the filter (8) into the hydraulic unit (5).
(b)Apply a coat of common grease (Mat. No. 04-015) to the threads of the nuts (9).
(c)Use the adaptor OU134344 to install the nut (9). TORQUE the nut



Autopilot Actuator Filter and Strainer
Figure 405

(9) to between 0.4 and 0.5 m.daN (35 to 44 lbf.in.).
(d)Safety the nut (9) with corrosion-resistan steel lockwire 0.8 mm (0.32 in) dia.
(e)Use a brush to apply special material (Mat. No. 05-005) to the head of the nut (9).
(f)Remove the rigging pin PN 97A27302009000.
(g)Remove safety clips and tags, and close circuit breakers previously open.
(h)Make sure that the control ranges are clear.
(j)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(k)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301)
(l)Make sure that there are no leaks from the filter (8).

(5)Test

(a)Do the test of the autopilot pitch-actuator system (Ref. 22-13-23, P. Block 401, only the test (5) (b)).

(6)Close-up

(a)Depressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).
(b)De-energize aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c)On the area around the nut (9), examine the coating of the special material (Mat. No. 05-027) for good condition and apply again as necessary (Ref. 51-75-10, P. Block 801).
(d)Close the access panels 334AB/AT (Ref. 55-13-13, P. Block 401).
(e)Make sure that the working area is clean and clear of tools and other items.
(f)Remove the access platfrom.
(g)Remove the warning notices.

5. Strainer, Pitch Actuator

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 3 to 6 m (10 to 20 ft)
B. 97A27302009000	Pin Rigging - Pitch Trim Control Wheel
C. 0U129455	Pre-casting Bush
D. 0U131397	Pre-casting Bush
E. 0U131396	Adaptor
F.	Torque Wrench 0 - 0.6 m.daN (0 - 53 lbf.in)
G.	Circuit Breaker Safety Clips and Tags
H.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
J.	Warning Notice - Prohibiting Unauthorized Use of Hydraulic Systems
K.	Brush
L. Material No. 04-015	Common Grease (Ref. 20-31-00)
M. Material No. 04-004	Common Grease (Ref. 20-31-00)
N. Material No. 05-005	Special Materials (Ref. 20-31-00)
P. Material No. 05-027	Special Materials (Ref. 20-31-00)
Q. Material No. 11-004	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures:	
- 22-13-23, P. Block 401	Autopilot Roll Actuator
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization)
- 51-75-10, P. Block 801	Repair Paint Coatings
- 55-13-13, P. Block 401	Trailing Edge Access Panels

B. Procedure

NOTE: The removal/installation procedure is the same for the LH or RH strainer (2).

(1) Job Set-Up

(a) Put the access platform in position below zone 334.
(b) Make sure that the pitch trim is set at zero.
(c) Make sure that the pitch control wheel is central and install the rigging pin PN 97A27302009000.
(d) Put warning notices in the flight compartment, prohibiting operation of the flight controls.
(e) On panel 21VU, open, safety and tag the following AUTO FLIGHT SYSTEM circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301/CU	102/H01
21VU	AFS/AP1/FO1/FCC1 26VAC	305/CA1	102/H02
21VU	AFS/AP1/FO1/FCC1 28VDC	306/CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312/CA1	102/H04
21VU	AFS/AP1/FO1/FCU	312/CA2	102/H05
21VU	AFS/AP2/FO2/FCC2 28VDC	306/CA2	102/H06
21VU	AFS/AP2/FO2/FCC2 26VAC	305/CA2	102/H07

(f)Open the access panels 334AB/AT and 344AB/AT (Ref. 55-13-13, P. Block 401).
(g)Put a warning notice in the flight compartment prohibiting use of the hydraulic systems.
(h)De-pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).

(2)Removal (Ref. Fig. 405)

CAUTION : TOOLS AND OTHER EQUIPMENTS, INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69 mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE.
DO NOT ALLOW DUST, DIRT OR METAL FILINGS TO CONTAMINATE THE FILTER SURFACES OR TO ENTER THE INTERIOR OF THE PITCH ACTUATOR.
FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

(a)Cut and remove the lockwire.
(b)Use the adaptor OU131396 to remove the nut (1).
(c)To remove the strainer (2), install a threaded rod (8 - 32 UNC 3A) into the body of the strainer (2) and remove it complete from the hydraulic unit (5). Remove the threaded rod from the strainer (2).
(d)Remove and discard the packing (4) and O-ring (3).

(3)Preparation of Replacement Component

(a)Clean the interface threads of the hydraulic unit (5) and the strainer (2) with lint-free cloth moist in cleaning agent (Mat. No. 11-004) until contamination is removed. Do not let cleaning agent dry on the surfaces.
(b)Use common grease (Mat. No. 04-015) and lubricate the packing (4) and O-ring (3).
(c)Use the pre-casting bush OU129455, and install the packing (4) on the strainer (2).
(d)Use the pre-casting bush OU131397, and install the O-ring (3) on the strainer (2).

(4)Installation

(a)Install the strainer (2) into the hydraulic unit (5).
(b)Apply a coat of common grease (Mat. No. 04-004) to the threads of the nut (1).
(c)Use the adaptor 0U131396 to install the nut (1). TORQUE the nut (1) to between 0.4 and 0.5 m.daN (35 to 44 lbf. in.).
(d)Safety the nut (1) with corrosion-resistant steel lockwire 0.8 mm (0.032 in.) dia.
(e)Use a brush to apply the special material (Mat. No. 05-005) to the head of the nut (1).
(f)Remove the rigging pins PN 97A27302009000.
(g)Remove safety clips and tags, and close circuit breakers previously open.
(h)Make sure that the control ranges are clear.
(j)Energize the aircraft electrical network (Ref. 24-41-00, P. Block k 301).
(k)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301).
(l)Make sure that there are no leaks from the strainer (2).

(5)Test

(a)Do the test of the autopilot pitch-actuator system (Ref. 22-13-23, P, Block 401. Only the test (5)(b)).

(6)Close-up

(a)De-pressurize the aircraft hydraulic system (Ref. 29-10-00, P. Block 301).
(b)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c)On the area around the nut (1), examine the coating of the special material (Mat. No 05-027) for good condition and apply again as necessary (Ref. 51-75-10, P. Block 801).
(d)Close the access panels 334AB/AT(Ref. 55-13-13, P.Block 401.
(e)Make sure that the work area is clean and clear of tools and other items.
(f)Remove the access platform.
(g)Remove the warning notices.

PITCH TORQUE LIMITER LEVER 19CA - REMOVAL/INSTALLATION

WARNING : CHECK THAT THE LANDING GEAR GROUND SAFETIES INCLUDING WHEEL CHOCKS ARE IN POSITION.
BEFORE APPLYING OR RELIEVING HYDRAULIC SYSTEM PRESSURE, MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, CHECK THAT ALL CONTROLS ARE SET TO CORRESPOND WITH THE ACTUAL POSITION OF THE SERVICES THEY OPERATE.
BEFORE PROCEEDING WITH MAINTENANCE WORK ON OR NEAR MECHANICAL FLIGHT CONTROLS OR PRIMARY FLIGHT CONTROL SURFACES, LANDING GEARS, ASSOCIATED DOORS OR ANY MOVING COMPONENT, MAKE CERTAIN THAT GROUND SAFETIES AND/OR WARNING NOTICES ARE IN CORRECT POSITION TO PREVENT INADVERTENT OPERATION OF CONTROLS.

1. Equipment and Materials

	ITEM	DESIGNATION
	A.	Access Platform, up to 8.3 m (27 ft.)
	B. 98A27207543000	Pin - Rigging, Autopilot Servomotor
	C. 98A27307549000	Pin - Rigging, Elevator Control Bellcranks
R	D. 98A27303001000	Tool - Locking Elevator (optional)
	E.	Torque Wrench, up to 1.0 m.daN (88.5 lbf.in.)
	F.	Blanking Caps - Electrical Connector
	G.	Cotter Pins
	H.	Circuit Breaker Safety Clips
	J.	Corrosion-Resistant Steel Lockwire, 0.8 mm (0.031 in.) dia.
	K.	Warning Notices
	L. Material No. 05-005	Special Materials (Ref. 20-31-00)

Referenced Procedures

-	22-10-00, P. Block 501	Autopilot
-	27-35-00, P. Block 501	Control Surface Position Indicating (Elevator)
-	29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
-	55-13-13, P. Block 401	Trailing Edge Access Panels

2. Procedure

A. Job Set-Up
(1)Position access platform.
(2)Make certain that pitch trim control wheels and control column are in neutral position.
(3)Display warning notices in flight compartment prohibiting operation of flight controls.
(4)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-AP1 FD1/FCC1/26VAC	305CA1	102/H02
21VU	AFS-AP1 FD1/FCC1/28VDC	306CA1	102/H03
21VU	FS-AP1 FD1/FCU	312CA1	102/H04
21VU	AFS-AP2 FD2/FCU	312CA2	102/H05
21VU	AFS-AP2 FD2/FCC2/28VDC	306CA2	102/H06
21VU	AFS-AP2 FD2/FCC2/26VAC	305CA2	102/H07
21VU	AFS-MTP	301CU	102/H01

(5)Remove trailing edge access panels 334AB/AT (Ref. 55-13-13, P. Block 401).
(6)Open access door 313AL.
(7)Pressurize the Green and Yellow Hidraulic Systems (Ref. 29-10-00, P. Block 301) and make sure that the elevator is set at zero position. (Ref. 27-35-00, P. Block 501).
(8)Display warning notices prohibiting depressurization.
(9)Insert rigging pin P/N 98A27307549000 in bellcrank at RIB4 and RIB9 servo controls (Ref. Fig. 402). As an option the elevator can also be set to zero position by using the Tool Locking Elevator P/N 98A27303001000.

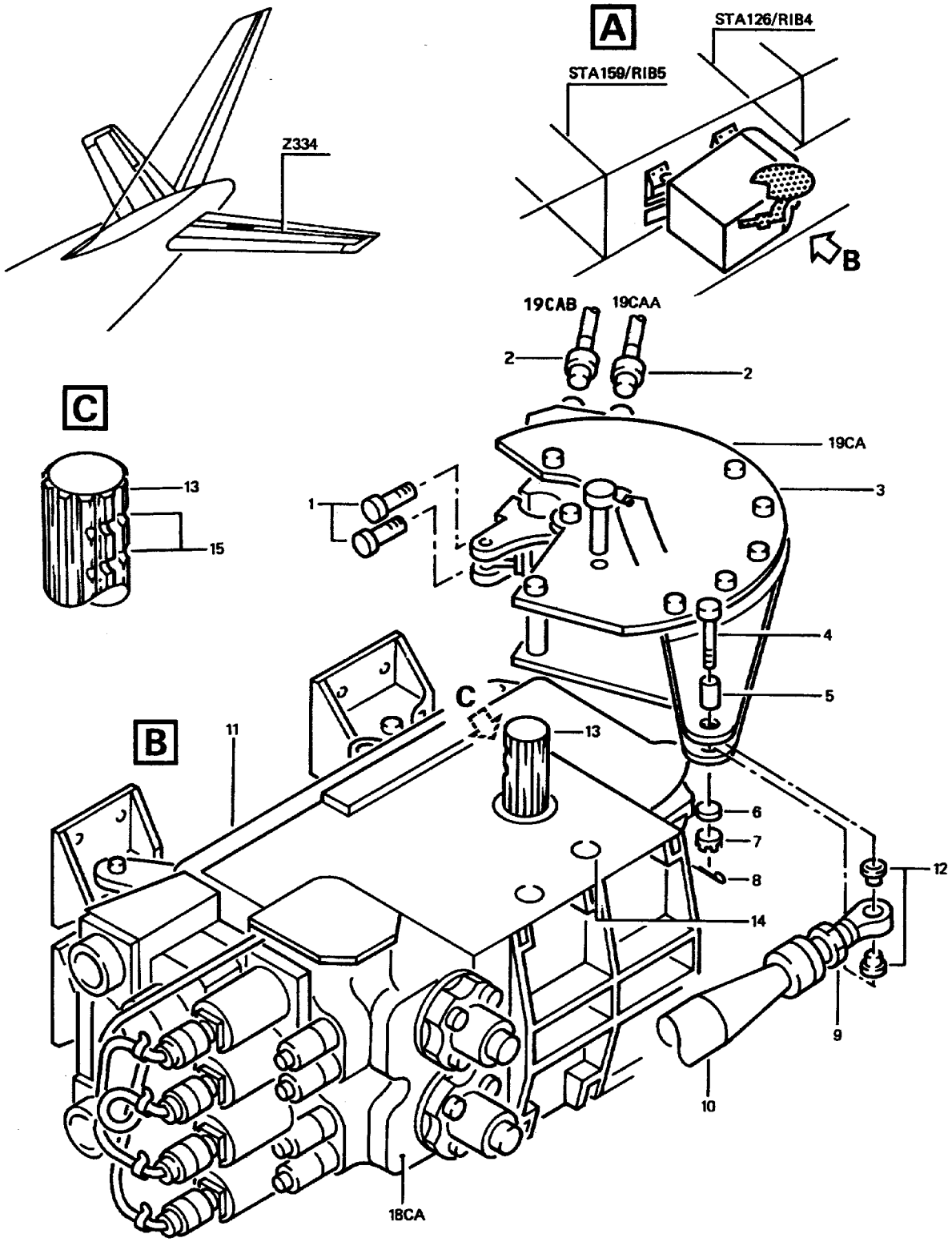
R

B. Removal (Ref. Fig. 401)
(1)Disconnect pushrod (10).
(a)Remove cotter pin (8), nut (7), washer (6), bolt (4) and spacer (5).
(b)Disconnect pushrod (10) and secure to structure.
(c)Secure bushes (12) temporarily.
(2)Disconnect electrical connectors 19CAA and 19CAB (2).
(3)Remove torque limiter lever (3).
(a)Cut and remove lockwire and remove bolts (1).
(b)Remove torque limiter lever (3) from shaft (13).

C. Installation (Ref. Fig. 401)
(1)Install torque limiter lever (3).
(a)Make certain that bolts (1) are removed from torque limiter lever (3).
(b)Position torque limiter lever (3) on shaft (13) so that grooves (15) for bolts (1) align.
(c)Install bolts (1).
(d)TORQUE bolts (1) to between 0.39 and 0.48 m.daN (34.5 and 42.5 lbf.in.) and safety with lockwire.
(2)Install rigging pin P/N 98A27207543000in bush (14) of autopilot pitch actuator.
(3)Install pushrod.
(a)Make certain that bushes (12) are positioned.
(b)Connect pushrod (10) to torque limiter lever (3) and install spacer (5) and bolt (4).
(c)If bolt (4) cannot be easily inserted, adjust pushrod (10) as follows:
1 Cut and remove lockwire and loosen nuts (9).
2 Adjust length of pushrod (10) until bolt (4) can be easily inserted.
3 Tighten nuts (9) and safety with lockwire.
(d)Install washer (6) and nut (7).
(e)TORQUE nut (7) to between 0.30 and 0.35 m.daN (26.5 and 31.0 lbf.in.) and safety with cotter pin (8).
(4)Connect electrical connectors 19CAA and 19CAB (2).
(5)Remove rigging pins P/N 98A27307549000 and P/N 98A27207543000.
(6)Remove safety clips and tags, and close circuit breakers 305CA1, 306CA1, 312CA1, 312CA2, 306CA2, 305CA2 and 301CU.
(7)Depresurize hidraulic system (Ref. 29-10-00 P. Block 301)
(8)Make certain that elevator control travel ranges are clear.

D. Test
(1)Carry out AP pitch mechanical override test (Ref. 22-10-00, P. Block 501).
(2)Do the AP Disconnection through torque limiters test (Ref. 22-10-00, P. Block 501).

E. Close-Up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Apply corrosion-preventive compound (Mat. No. 05-005) to rod fittings.
(3)Install trailing edge access panels (Ref. 55-13-13, P. Block 401).
(4)Close access door 313AL.
(5)Remove warning notices.
(6)Remove access platform.



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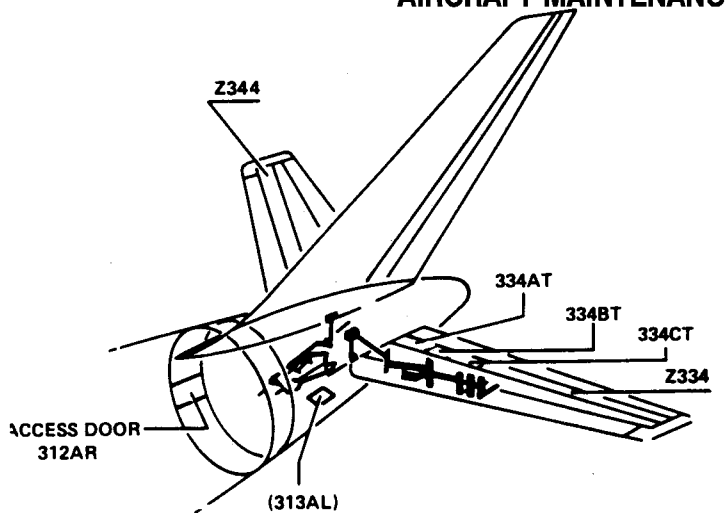
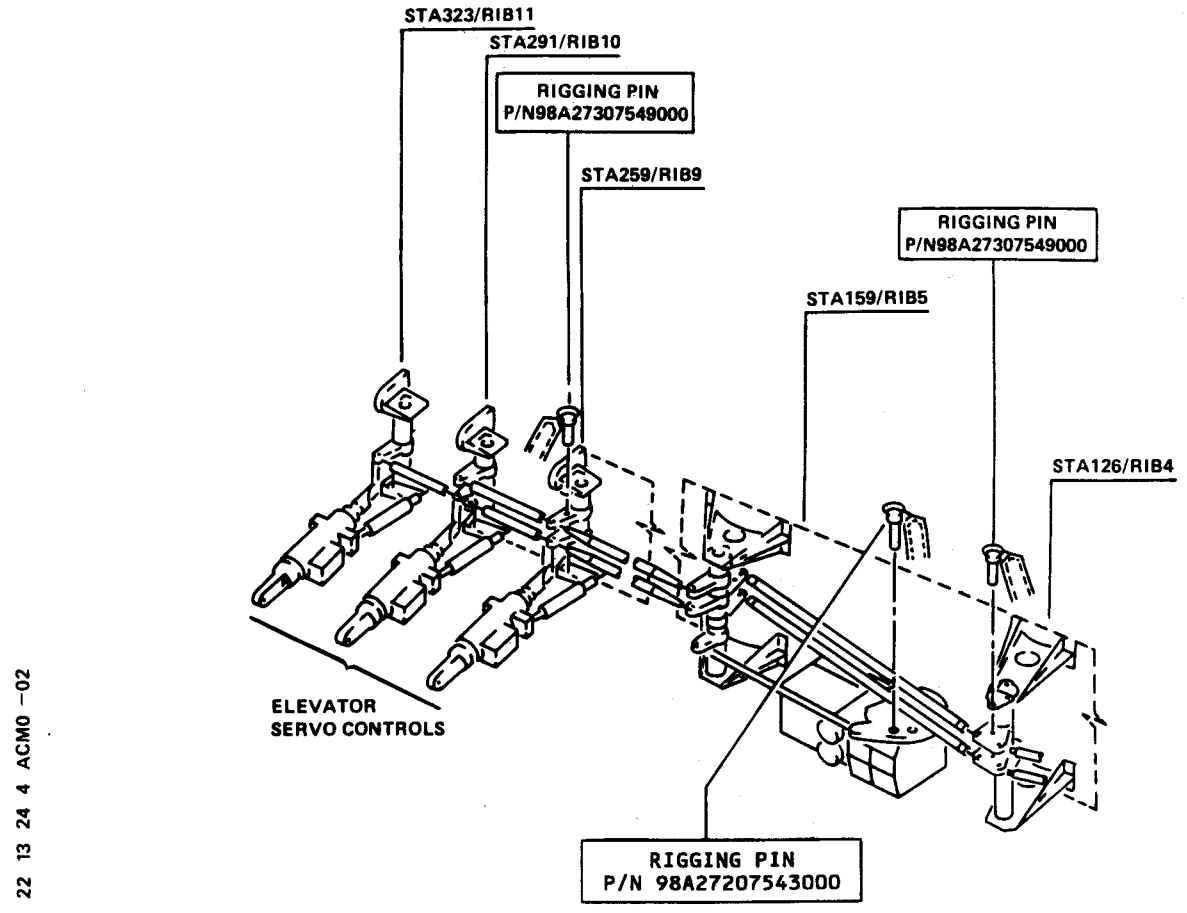
Pitch Torque Limiter Lever
Figure 401





Rigging Location
Figure 402

AUTOPILOT ROLL ACTUATOR - REMOVAL/INSTALLATION

WARNING : LANDING GEAR - MAKE SURE THAT THE GROUND SAFETIES AND CHOCKS ARE IN POSITION.

HYDRAULICS - BEFORE YOU PRESSURIZE THE SYSTEM, MAKE SURE THAT THE CONTROL SURFACE POSITIONS AND THE FLIGHT CONTROL POSITIONS ARE THE SAME.

MAKE SURE THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR BEFORE YOU PRESSURIZE/DE-PRESSURIZE A HYDRAULIC SYSTEM.

PUT THE SAFETY DEVICES AND THE WARNING NOTICES IN POSITION BEFORE YOU START A TASK ON OR NEAR:
- THE FLIGHT CONTROLS
- THE FLIGHT CONTROL SURFACES
- THE LANDING GEAR AND THE RELATED DOORS
- COMPONENTS THAT MOVE.

MAKE SURE THAT YOU ISOLATE THE ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS

1. Reason(s) for the Job

A. Roll Actuator - Removal/Installation
B. Servo-Valve - Removal/Installation
C. Solenoids - Removal/Installation
D. Filter - Removal/Installation
E. Strainer - Removal/Installation.

2. ROLL ACTUATOR

A. Equipment and Materials

	ITEM	DESIGNATION
	A.	Access Platform 3 m (10 ft.)
	B.	Circuit Breaker Safety Clips and Tags
	C.	Container
	D.	Warning Notice, Prohibiting Unauthorized Use of Flight Controls
	E.	Warning Notice, Prohibiting Use of Aircraft Hydraulic Systems
	F. 98A27108014000	Pin - Rigging
	G. 98A27207543000	Pin - Rigging
R	H. E5112	AP Actuator Locking Tool (installed on the
R		replacement actuator)
R	J.	Applicable Support to prevent droop of the Flight Control Surface(s)
R	K. 98A27608249000	R/I Tool - Roll Actuator
R	L. 98A27901000000	Pin - Rigging

	ITEM	DESIGNATION
R	M.	Blanking Caps and Plugs - Electrical
R	N.	Blanking Caps and Plugs - Hydraulic
R	P.	Lockwire, Corrosion Resistant Steel, 0.8 mm (0.032 in) dia.
	Referenced Procedures	
	- 20-23-12, P. Block 1	Specified Torque Values for Unions
	- 20-28-11, P. Block 1	Electrical Bonding
	- 22-26-00, P. Block 1	Yaw Damper
	- 22-27-00, P. Block 1	Electric Pitch Trim
	- 24-41-00, P. Block 301	AC External Power Control
	- 27-80-00, P. Block 301	Lift Augmenting (Slats and Krueger Flaps)
	- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/
	- 34-10-00, P. Block 501	Flight Environment Data
	- 34-25-00, P. Block 501	Inertial Reference System

B. Procedure
(Ref. Fig. 401)

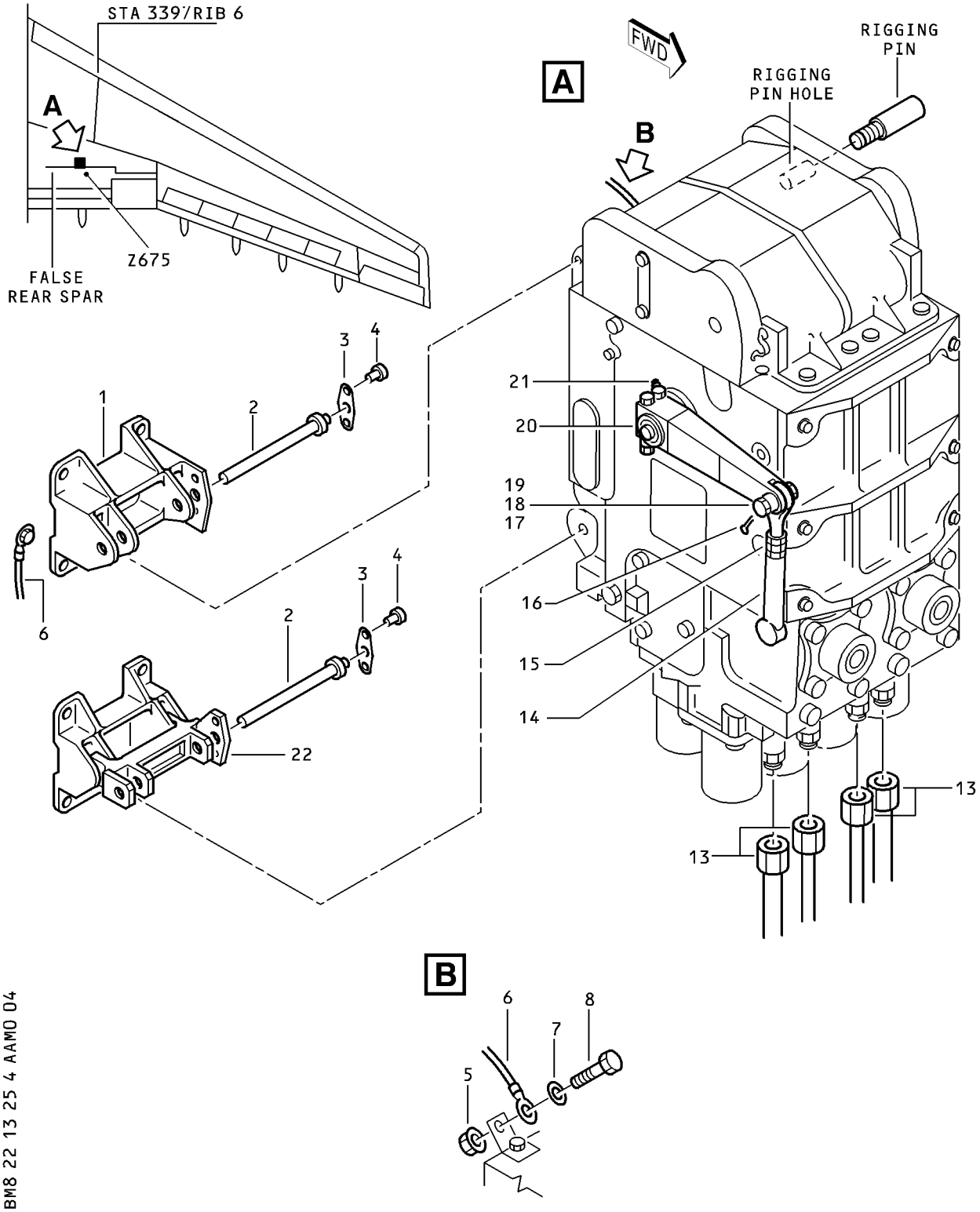
(1)Job Set-Up
(a)Put the access platform in position below zone 675.
(b)Make sure that the control ranges are clear and pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

(c)Make sure that the aileron trim is set at zero.
(d)Make sure that the aileron control wheel is central and install the rigging pin PN 98A27901000000.
(e)Put a warning notice in the flight compartment to prevent the unapproved operation of the flight controls.
(f)On panel 21VU, open, safety and tag the AUTO FLIGHT SYSTEM circuit breakers that follow:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301CU	102/H1
21VU	AFS/AP1-FD1/FCC1/26VAC	305CA1	102/H2
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H3
21VU	AFS/AP1/FD1/FCU	312CA1	102/H4
21VU	AFS/AP2/FD2/FCU	312CA2	102/H5
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H6
21VU	AFS/AP2-FD2/FCC2/26VAC	305CA2	102/H7

(g)Open the access door 675AB.



BM8 22 13 25 4 AAMO 04

Actuator Autopilot Roll - Removal/Installation
Figure 401

R

CAUTION : MAKE SURE THAT YOU PRESSURIZE THE HYDRAULIC SYSTEM(S) BEFORE YOU INSTALL THE RIGGING PINS. IF YOU DO NOT PRESSURIZE THE HYDRAULIC SYSTEM(S), THE FLIGHT CONTROL SURFACES WILL NOT STAY AT ZERO, BUT WILL DROOP AND PUT A FORCE ON THE RIGGING PINS.

(h)Install the rigging pin PN 98A27207543000 into the outboard side of the actuator, and the rigging pin PN 98A27108014000, into the aileron cable-drum differential droop unit.
(j)Put a support in position below the applicable flight control surface(s), to prevent droop.
(k)De-pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).
(l)Put a warning notice in the flight compartment to prevent the unapproved operation of the hydraulic system(s).

(2)Removal
(a)Disconnect the two electrical connectors on the actuator and install the blanking caps and plugs onto the fixed and free ends.
(b)Make sure that the hydraulic systems are depressurized (Ref. 29-10-00, P. Block 301).
(c)Put the container in position below the hydraulic lines (13).
R (d)Identify and mark the hydraulic lines and connections for subsequent
R correct installation.
(e)Disconnect the hydraulic delivery and return lines (13) and cap the open line ends and the hydraulic connections on the actuator.
(f)Remove the container.
(g)Remove the nut (5), the washer (7) and the bolt (8) from the attachment bracket of the actuator bonding lead. Disconnect the bonding lead (6) from the actuator.
(h)Remove and discard the cotter-pin (16).

CAUTION : MAKE SURE THAT THE PUSHROD (14) IS NOT PUSHED OUT BY THE RESIDUAL INTERNAL PRESSURE OF THE AUTOPILOT ACTUATOR.

(j)Remove the nut (17), the washer (18) and the bolt (19) and disconnect the push rod (14) from the lever (20).
(k)Remove the bolts (4) from the upper and lower mounting brackets. Remove the retaining plates (3).
(l)Put the platter in position below the actuator on the inside wing skin and engage it over the camloks and the bolt tails.
(m)Install the frame onto the actuator and hold in position using the two pins. Screw the pins into the captive nuts on the frame to fully secure it.
(n)Put the wedge in position between the platter and the frame. Tap the wedge with a mallet until the weight of the actuator, approximately 20 kgs (44 lbs), is supported.
(p)Withdraw the top locating pin (2) using the slide hammer assembly.
(q)Withdraw the lower locating pin (2) using the extractor assembly.
(r)Remove the autopilot roll actuator from the aircraft.

(3)Preparation of the Replacement Component

R (a)On the actuator you have removed:
R 1 Unscrew and remove the bolts (21) from the lever (20), and remove the
R lever.
R 2 Remove the rigging pin PN 98A27207543000.
R (b)On the replacement actuator:
R 1 Remove the AP actuator locking tool PN E5112.
R 2 Install the rigging pin PN 98A27207543000 into the outboard side.

CAUTION : MAKE SURE YOU DO NOT INSTALL THE LEVER (20) 180 DEG. OUT OF
R POSITION. THE LEVER (20) MUST BE INSTALLED IN THE DOWN
R POSITION.

R 3 Install the lever (20), in the same related position on the actuator
R splined shaft. Safety the lever (20) with the threaded bolts (21).
R (c)Install the AP actuator locking tool PN E5112, on the removed
R actuator.

(4)Installation
(a)Put the frame in position on the actuator and hold it in position with the pins, (put in from the opposite side to the lever (20)). Move the lever (20) to the up position above the upper pin.
(b)Turn the pins into the captive nuts on the frame to fully safety the actuator.
(c)Put the platter in position on the inside wing skin, and engage it above the camloks and the bolt tails.
(d)Put the actuator and frame, correctly aligned, on the platter.
(e)Put the wedge between the platter and the frame. Tap the wedge with a mallet (to raise the actuator until the holes in the actuator are aligned with the upper mounting bracket (1)).
(f)Install the top locating pin (2) into the upper mounting bracket (1). Remove the wedge and install a second locating pin (2) through the link (22).
(g)Remove the platter from the wing skin.
(h)Turn the pins and remove them from the frame. Remove the frame from the actuator.
(j)Put the container in position below the hydraulic lines (13).
(k)Remove the caps from the actuator hydraulic connections and hydraulic lines.

CAUTION : MAKE SURE THAT YOU CONNECT THE HYDRAULIC LINES TO THE CORRECT HYDRAULIC CONNECTIONS ON THE ACTUATOR. IF YOU DO NOT CONNECT THE LINES CORRECTLY, THE TEST (AFTER YOU HAVE COMPLETED THE INSTALLATION) WILL FAIL.

(l)Loosely connect the hydraulic delivery and return lines (13) to the actuator.
(m)Install the retaining plates (3) and safety them with the bolts (4).
(n)TORQUE the loose hydraulic delivery and return lines (13) (Ref. 20-32-12, P. Block 1).
(p)Remove the container.
(q)Make sure that the control ranges are clear and pressurize the aircraft

hydraulic system(s) (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

CAUTION : YOU MUST MAKE SURE THAT THE PUSHROD (14) IS CORRECTLY ADJUSTED WHEN THE AIRCRAFT HYDRAULIC SYSTEM(S) ARE PRESSURIZED. IF THE PUSHROD (14) IS NOT ADJUSTED WHEN THE HYDRAULIC SYSTEM(S) ARE PRESSURIZED, THE CONTROL SURFACES WILL NOT STAY AT ZERO, BUT WILL DROOP AND PUT A FORCE ON THE RIGGING PINS. THIS CAN CAUSE THE CONTROL SURFACES TO MOVE DURING THE DISENGAGEMENT OF THE AP1/AP2.

: MAKE SURE THAT THE PUSHROD (14) IS STILL IN POSITION.

(r)Align the pushrod (14) with the fork end of the lever (20), install the bolt (19) (and safety with the washer (18) and the nut (17)).

NOTE : If the pushrod (14) will not align, adjust it as follows:

1 Remove and discard the lockwire from the locknut (15) and loosen the eye-end locknut (15).
2 Turn the eye end of pushrod (14) until it aligns with the forked end of lever (20) and install the bolt (19).
3 Tighten the eye-end locknut (15) and safety with lockwire.
4 Tighten the bolt (19), with the washer (18) and the nut (17) and safety with a new cotter-pin (16).

(s)Remove the support from below the flight control surface(s).
(t)Remove the rigging pins PN 98A27108014000, PN 98A27207543000 and PN 98A27901000000.
(u)Depressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).
(v)Install the bonding lead (6) to the actuator attachment bracket with the bolt (8), the washer (7) and the nut (5) (Ref. 20-28-11, P. Block 1).
(w)Remove the blanking caps and plugs and examine the electrical connections for the correct condition. Connect them to the actuator.
(x)Remove the safety clips and tags, and close the circuit breakers that follow:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301CU	102/H01
21VU	AFS/AP1/FD1/FCC1 26VAC	305CA1	102/H02
21VU	AFS/AP1/FD1/FCC1 28VDC	306CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312CA1	102/H04
21VU	AFS/AP2/FD2/FCU	312CA2	102/H05
21VU	AFS/AP2/FD2/FCC2 28VDC	306CA2	201/H06
21VU	AFS/AP2/FD2/FCC2 26VAC	305CA2	102/H07

(y)Remove the warning notices from the flight compartment.

(z)Make sure that the control ranges are clear and pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

(aa)Make sure that the hydraulic connections are free from leaks.

R (ab)Depressurize the aircraft hydraulic system(s) (Ref. 29-10-00,
R P. Block 301).

(5)Test

(a)Initial conditions:

1 Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

2 Make sure that the Flight Environment Data is operating (Ref. 34-10-00, P. Block 501).

3 Make sure that the Inertial Reference Systems 1, 2 and 3 are aligned, (perform the align procedure until the red HDG flags on the NDS disappear, present position entering is not necessary) (Ref. 34-25-00, P. Block 501).

R 4 Make sure that the control ranges are clear and pressurize the
R aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

R 5 Make sure that the pitch trim control wheel and the control surfaces are in the neutral position.

R 6 Adjust the ECAM display unit brightener on the center pedestal as required and select the FLT CTL page.

R 7 On the overhead panel, make sure that the Pitch Trim 1 system is operating, engage the lever into the ON position (Ref. 22-27-00, P. Block 1).

R 8 On the overhead panel, make sure that the Yaw Damper 1 system is operating, engage the lever into the ON position (Ref. 22-26-00, P. Block 1).

R 9 Make sure that the EF1S is operating, by adjustment on panel 412VU, 413VU of PFD and ND brightness.

(b)Do the test that follows, to make sure that the electrical components of the actuator were connected correctly during the removal/installation procedure:

ACTION	RESULT
1. On panel 21VU: - open the AP2-FD2/FCC2/28V DC circuit breaker 306CA2 (line 102/H6).	- the FCC2 is de-energized.
2. On the flight control unit (FCU): - put the AP1 lever in the ON position - push (in) the CWS/CMD p/bsw - if the AP1 lever does not stay engaged, stop the test and check the electrical connections.	- the AP1 lever is engaged - the AP1 lever stays engaged.
3. On the flight control unit (FCU):	On the NDS:

ACTION	RESULT
- with HDG SEL selector knob select a heading deviation of 10 deg. to the right	- the heading index moves in relation.
- pull the HDG SEL selector knob	On the FCU: - the HDG SEL P/BSW comes on green. On the PFDs: - the green HDG goes off - the green HDG/S comes on. The control wheel rotates to the right.
- select a heading deviation equal to the heading.	The control wheel returns towards the neutral position.
4. Disengage the AP1 lever.	- the AP1 lever is disengaged.
5. On panel 21VU: - close the AP2-FD2/FCC2/28V DC circuit breaker 306CA2 (line 102/H6)	- the FCC2 is energized
- open the AP1-FD1/FCC1/28V DC circuit breaker 306CA1 (line 102/H3).	- the FCC1 is de-energized.
6. On the flight control unit (FCU): - put the AP2 lever in the ON position	- the AP2 lever is engaged
- push (in) the CWS/CMD p/bsw	- the AP2 lever stays engaged.
- if the AP2 lever does not stay engaged, stop the test and check the electrical connections.	
7. On the flight control unit (FCU): - with HDG SEL selector knob select a heading deviation of 10 deg. to the right	On the NDS: - the heading index moves in relation
- pull the HDG SEL selector knob	On the FCU: - the HDG SEL P/BSW comes on green. On the PFDs: - the green HDG goes off - the green HDG/S comes on. The control wheel rotates to the right
- select a heading deviation equal to the heading.	The control wheel returns towards the neutral position.
8. Disengage the AP2 lever and	

ACTION	RESULT
cancel the warning.	
9. On panel 21VU: - close the AP1-FD1/FCC1/28V DC circuit breaker 306CA1 (line 102/H3).	- the FCC1 is energized.

(c) Do the test that follows, to make sure that the hydraulic lines to the actuator were connected correctly during the removal/installation procedures.

NOTE : Failure to correctly engage the autopilot systems 1 and 2 in the command mode during this test will show that the hydraulic lines are not connected correctly.

ACTION	RESULT
1. Set the hydraulic pressure initial configuration: - Blue system available for yaw damper 1 engagement (Ref. 29-22-00 P. Block 301) - Green system available for AP 1 engagement (Ref. 29-21-00, P. Block 301) - Yellow system isolated to check the hydraulic line connection.	
2. On the flight control unit (FCU): - put the AP1 lever in the ON position	- the AP1 lever is engaged
- push (in) the CWS/CMD p/bsw	- the AP1 lever stays engaged.
- if the AP1 lever does not stay engaged, stop the test and do the procedure to correct the connections of the hydraulic lines (Ref. para. 2.B.(6)).	
3. Disengage the AP1 lever.	- the AP1 lever is disengaged.
4. Change the hydraulic pressure configuration: - Yellow system available for yaw damper 2 and AP2 engagement (Ref. 29-23-00, P. Block 301) - Green system isolated to check correct hydraulic line connection	

ACTION	RESULT
- the Green system can be isolated at the servo control panel if the yellow system is pressurized through the PTU.	
5. On the flight control unit (FCU):	
- put the AP2 lever in the ON position	- the AP2 lever is engaged
- push (in) the CWS/CMD p/bsw	- the AP2 lever stays engaged.
- if the AP2 lever does not stay engaged, stop the test and do the procedure to correct the connections of the hydraulic lines (Ref. para. 2.B.(6)).	
6. Disengage the AP2 lever.	- the AP2 lever is disengaged.

(6)Correction of hydraulic line connections

NOTE : Use this procedure to correct the hydraulic line connections to the roll actuator. Do the procedure when the test to engage the autopilot systems 1 and 2 in the command mode fails.

(a)Depressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).
(b)Open, safety and tag the circuit breakers 305CA1, 306CA1, 312CA1, 312CA2, 306CA2, 305CA2 and 301CU (Ref. para. 2.B.(1)(f)).
(c)Place a container under the hydraulic connections.
(d)Where installed, do not remove clamp (16). Identify and mark lines and connections for correct installation. Disconnect the hydraulic delivery and return lines (5 thru 8) from the actuator.

CAUTION : MAKE SURE THAT YOU CONNECT THE HYDRAULIC LINES TO THE CORRECT HYDRAULIC CONNECTIONS ON THE ACTUATOR. IF YOU DO NOT CONNECT THE LINES CORRECTLY, THE TEST (AFTER YOU HAVE COMPLETED THIS PROCEDURE) WILL FAIL AGAIN.

(e)Connect the hydraulic delivery and return lines (5 thru 8) to the correct positions on the actuator.
(f)Tighten the hydraulic lines (Ref. 20-23-12, P. Block 1).
(g)Remove the container from under the hydraulic connections.
(h)Remove the safety clips and tags and close circuit breakers 305CA1, 306CA1, 312CA1, 312CA2, 306CA2, 305CA2 and 301CU.
(j)Make sure that the control ranges are clear and pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).
(k)Do the test, Ref. para. 2.B.(5)(b), to make sure the hydraulic lines are connected correctly.

(7)Close-Up
(a)Depressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).
(b)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c)Remove the warning notices.
(d)Close the access door 675AB.
(e)Make sure that the work area is clean and clear of tools and other items of equipment.
(f)Remove the access platform.

3. SERVO-VALVE

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 3 to 6 m (10 to 20 ft)
B.	Container
C.	Circuit Breaker Safety Clips and Tags
D.	Torque Wrench 0 - 0.5 m.daN (0 - 44.2 lbf.in)
E.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
F.	Warning Notice - Prohibiting Unauthorized Use of Hydraulic Systems
G. 0U130560	Adaptor
H. 98A27207543000	Pin - Rigging
J. 98A27108014000	Pin - Rigging
K. 98A27901000000	Pin - Rigging
L.	Blanking Caps and Plugs - Electrical
M.	Brush
N.	Lint-Free Cloth
P.	Applicable Support to prevent droop of the Flight Control Surface(s)
Q. Material No. 04-004	Common Grease (Ref. 20-31-00)
R. Material No. 04-015	Common Grease (Ref. 20-31-00)
S. Material No. 05-005	Special Materials (Ref. 20-31-00)
T. Material No. 05-027	Special Materials (Ref. 20-31-00)
U. Material No. 11-026	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization)
- 51-75-10, P. Block 801	Repair Paint Coatings

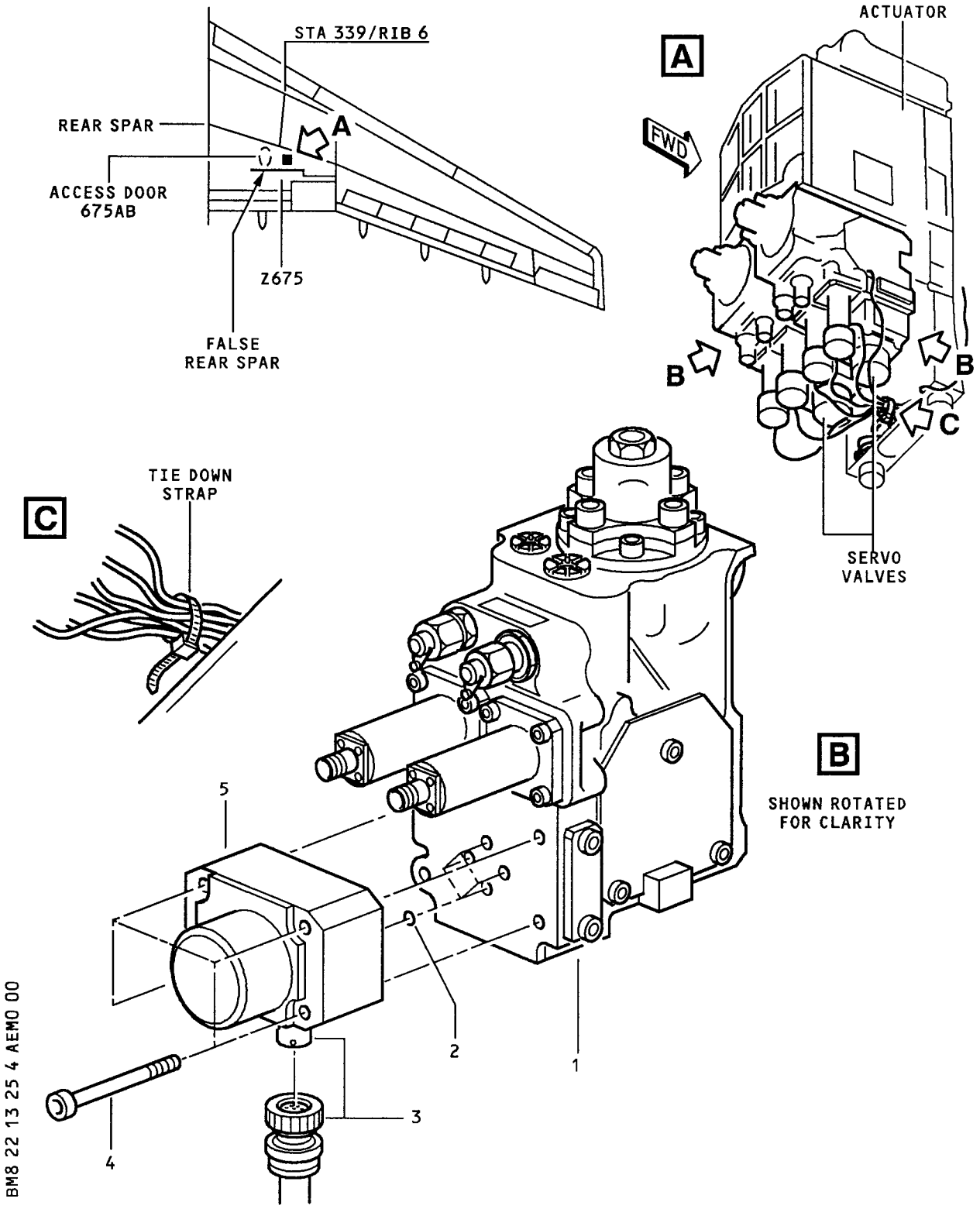
B. Procedure
(Ref. Fig. 402)

NOTE : The removal/installation procedure is the same for the LH or the RH servo-valve (5).

(1)Job Set-Up
(a)Put the access platform in position below zone 675.
(b)Pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized the flight control surfaces will not droop and put a force on the rigging pins.

(c)Make sure that the aileron trim is set at zero.
(d)Make sure that the aileron control wheel is central and install the rigging pin PN 98A27901000000.



Autopilot Roll Actuator, Servo-Valve
Figure 402

(e)Put the warning notice in the flight compartment, prohibiting unauthorized operation of the flight controls.

(f)On panel 21VU, open, safety and tag the following AUTO FLIGHT SYSTEM circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301/CU	102/H01
21VU	AFS/AP1/FD1/FCC1 26VAC	305/CA1	102/H02
21VU	AFS/AP1/FD1/FCC1 28VDC	306/CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312/CA1	102/H04
21VU	AFS/AP1/FD1/FCU	312/CA2	102/H05
21VU	AFS/AP2/FD2/FCC2 28VDC	306/CA2	102/H06
21VU	AFS/AP2/FD2/FCC2 26VAC	305/CA2	102/H07

(g)Open the access door 675AB.

CAUTION : MAKE SURE THAT YOU PRESSURIZE THE HYDRAULIC SYSTEM(S) BEFORE YOU INSTALL THE RIGGING PINS. IF YOU DO NOT PRESSURIZE THE HYDRAULIC SYSTEM(S), THE FLIGHT CONTROL SURFACES WILL NOT STAY AT ZERO, BUT WILL DROOP AND PUT A FORCE ON THE RIGGING PINS.

(h)Install the rigging pin PN 98A27207543000 into the outboard side of the actuator and the rigging pin PN 98A27108014000 into the aileron cable-drum differential-droop unit.

(j)Put a support in position below the applicable flight control surface(s), to prevent droop.

(k)De-pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

(l)Put the warning notice in the flight compartment prohibiting unauthorized use of the hydraulic system(s).

(2)Removal

CAUTION: TOOLS AND OTHER EQUIPMENTS, INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69 mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE. STORE THE REMOVED SERVO-VALVE IN A PROTECTIVE COVER AND AWAY FROM MAGNETIC MATERIALS. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

DO NOT ALLOW DUST, DIRT OR METAL FILINGS TO CONTAMINATE THE INTERIOR PASSAGE OF THE SERVO-VALVE OR THE ROLL ACTUATOR MANIFOLD. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

(a)Remove and discard the tie down strap that holds the electrical harness between the servo-valves (5).

(b)Disconnect the applicable electrical connector (3) from the servo-valve (5) and install the blanking caps and plugs onto the fixed and free ends.

(c)Put the container in position below the servo-valve (5).
(d)Remove the screws (4) from the applicable servo-valve (5).
(e)Carefully remove the servo-valve (5) from the hydraulic unit (1).
(f)Remove and discard the O-rings (2) and put the servo-valve (5) into a protective cover.
(g)Blank the open manifold face of the hydraulic unit (1).

(3)Preparation of Replacement Component
(a)Remove the protective cover from the servo-valve (5) and the blank from the manifold face of the hydraulic unit (1).
(b)Use cleaning agents (Material No. 11-026) with a clean lint-free cloth to make sure that the hydraulic unit (1) and the servo-valve (5) manifold faces are fully cleaned.
(c)Apply a coat of common grease (Material No. 04-004) to the manifold faces of the hydraulic unit (1) and the servo-valve (5) and to the threads of the screws (4).
(d)Use common grease (Material No. 04-015) and lubricate the new O-rings (2).

(4)Installation
(a)Put the new O-rings (2) in position on the mating face of the servo-valve (5).
(b)Put the servo-valve (5) in position on the hydraulic unit (1) and install the screws (4).
(c)Use the adaptor PN 0U130560 to install the screws (4). TORQUE the screws (4) to between 0.39 and 0.49 m.daN (34.5 and 43.3 lbf.in).
(d)Remove the container.
(e)Use a brush to apply special material (Material No. 05-005) to the heads of the screws (4).
(f)Remove the caps and plugs and examine the ends of the electrical connector (3) for the correct condition then connect the electrical connector (3) to the servo-valve (5).
(g)Put back the electrical harness of the actuator between the servo-valves (5) and install a new tie down strap.
(h)Make sure that the control surface ranges are clear.
(j)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(k)Pressurize aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

(l)Remove the support from below the flight control surface(s).
(m)Remove the rigging pins PN 98A27108014000, PN 98A27207543000 and PN 98A27901000000.
(n)Remove safety clips and tags and close the circuit breakers 301CU, 305CA1, 306CA1, 312CA1, 312CA2, 306CA2 and 305CA2.
(p)Make sure that the actuator and the servo-valves (5) are free from hydraulic leaks.

(5)Test
(a)Do the test to make sure that the electrical components of the actuator

were connected correctly during the removal/installation procedure (Ref. Para. 2.B.(5)(b)).

(6)Close-Up
(a)De-pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).
(b)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c)On the hydraulic unit (1) and the servo-valve (5), examine the coating of the special material (Material No. 05-027) for good condition and renew as necessary (Ref. 51-75-10, P. Block 801).
(d)Close the access door 675AB.
(e)Make sure that the work area is clean and clear of tools and miscellaneous items of equipment.
(f)Remove the access platform.
(g)Remove the warning notices.

4. SOLENOIDS

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 3 to 6 m (10 to 20 ft)
B.	Circuit Breaker Safety Clips and Tags
C.	Torque Wrench 0 - 0.5 m.daN (0 - 44.2 lbf.in)
D.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
E.	Warning Notice - Prohibiting Unauthorized Use of Hydraulic Systems
F. 0U130411	Mounting Cone
G. 0U130423	Adaptor
H. 98A27108014000	Pin - Rigging
J. 98A27207543000	Pin - Rigging
K. 98A27901000000	Pin - Rigging
L.	Blanking Caps and Plugs - Electrical
M.	Brush
N.	Lint-Free Cloth
P.	Lockwire, Corrosion-Resistant Steel 0.8 mm (0.032 in.) dia.
Q.	Applicable Support to prevent droop of the Flight Control Surface(s)
R. Material No. 04-004	Common Grease (Ref. 20-31-00)
S. Material No. 04-015	Common Grease (Ref. 20-31-00)
T. Material No. 05-005	Special Materials (Ref. 20-31-00)
U. Material No. 05-027	Special Materials (Ref. 20-31-00)
V. Material No. 11-026	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization)
- 51-75-10, P. Block 801	Repair Paint Coatings

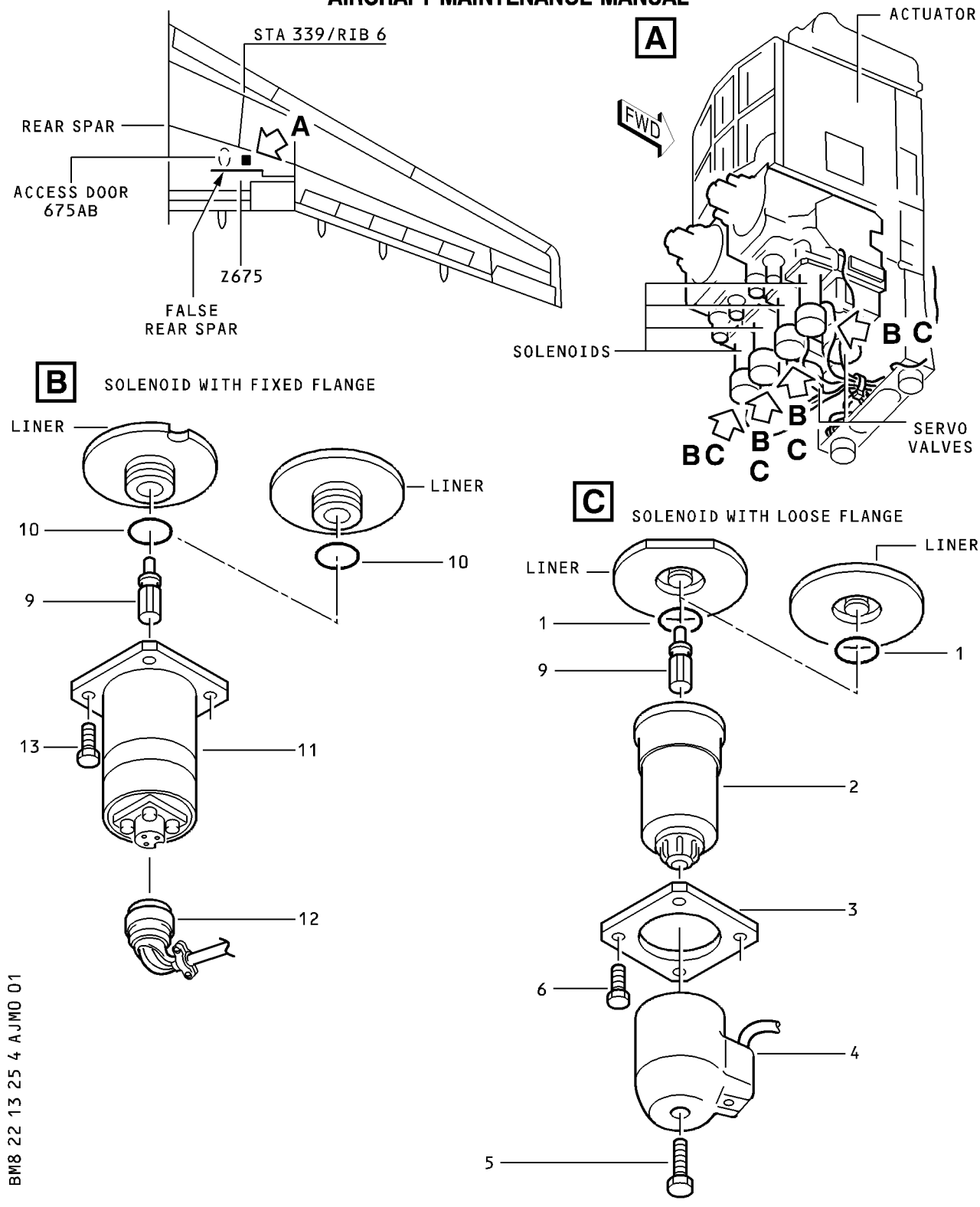
B. Procedure
(Ref. Fig. 403)

NOTE : The removal/installation procedure is the same for the LH or the RH pair of solenoids (2) and (11).

(1)Job Set-Up
(a)Put the access platform in position below zone 675.
(b)Pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

(c)Make sure that the aileron trim is set at zero.



Autopilot Roll-Actuator, Solenoids
Figure 403

R

(d)Make sure that the aileron control wheel is central and install the rigging pin PN 98A27901000000.
(e)Put the warning notice in the flight compartment, prohibiting unauthorized operation of the flight controls.
(f)On panel 21VU, open, safety and tag the AUTO FLIGHT SYSTEM circuit breakers that follow:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301/CU	102/H01
21VU	AFS/AP1/FD1/FCC1 26VAC	305/CA1	102/H02
21VU	AFS/AP1/FD1/FCC1 28VDC	306/CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312/CA1	102/H04
21VU	AFS/AP1/FD1/FCU	312/CA2	102/H05
21VU	AFS/AP2/FD2/FCC2 28VDC	306/CA2	102/H06
21VU	AFS/AP2/FD2/FCC2 26VAC	305/CA2	102/H07

(g)Open the access door 675AB.

CAUTION : MAKE SURE THAT YOU PRESSURIZE THE HYDRAULIC SYSTEM(S) BEFORE YOU INSTALL THE RIGGING PINS. IF YOU DO NOT PRESSURIZE THE HYDRAULIC SYSTEM(S), THE FLIGHT CONTROL SURFACES WILL NOT STAY AT ZERO, BUT WILL DROOP AND PUT A FORCE ON THE RIGGING PINS.

(h)Install the rigging pin PN 98A27207543000 into the outboard side of the actuator, and the rigging pin PN 98A27108014000 into the aileron cable-drum differential-droop unit.
(j)Put a support in position below the applicable flight control surface(s), to prevent droop.
(k)De-pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).
(l)Put the warning notice in the flight compartment prohibiting unauthorized use of the hydraulic system(s).

(2)Removal

CAUTION : TOOLS AND OTHER EQUIPMENT, INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

(a)On solenoids (11) with a fixed flange.
1 Put labels or marks on the solenoids to be removed and their related electrical connectors to help you identify the correct connections when you install the solenoids.

CAUTION : DO NOT REMOVE MORE THAN ONE SOLENOID FROM THE ACTUATOR OR DISCONNECT MORE THAN ONE ELECTRICAL CONNECTOR (12) AT THE SAME TIME. YOU MUST RE-INSTALL OR REPLACE THE SOLENOID BEFORE YOU CONTINUE TO REMOVE MORE SOLENOIDS FROM THE ACTUATOR. THIS IS TO PREVENT CROSS CONNECTION OF THE SOLENOIDS.

2 Cut and remove the lockwire and disconnect the electrical connector (12). Install the blanking caps and plugs onto the open ends of the electrical connector (12) and the solenoid (11). Discard the lockwire.

R CAUTION : MAKE SURE THAT THE PUSHROD (9) IS NOT PUSHED OUT BY
R INTERNAL RESIDUAL PRESSURE OF THE AUTOPILOT ACTUATOR.

3 Cut and remove the lockwire and remove the screws (13) and the solenoid (11). Discard the lockwire.
4 Remove and discard the packing (10).
5 Put a blanking cap on the liner.

(b)On solenoids (2) with a loose flange.
1 Put labels or marks on the solenoids to be removed and their related electrical connectors to help you identify the correct connections when you install the solenoids.

CAUTION : DO NOT REMOVE MORE THAN ONE SOLENOID FROM THE ACTUATOR OR DISCONNECT MORE THAN ONE ELECTRICAL CONNECTOR (4) AT THE SAME TIME. YOU MUST RE-INSTALL OR REPLACE THE SOLENOID BEFORE YOU CONTINUE TO REMOVE MORE SOLENOIDS FROM THE ACTUATOR. THIS IS TO PREVENT CROSS CONNECTION OF THE SOLENOIDS.

2 Cut and remove the lockwire and remove the screw (5). Discard the lockwire.
3 Disconnect the electrical connector (4) and install the blanking caps and plugs onto the open ends of the electrical connector (4) and the solenoid (2). Discard the lockwire.

R CAUTION : MAKE SURE THAT THE PUSHROD (9) IS NOT PUSHED OUT BY
R INTERNAL RESIDUAL PRESSURE OF THE AUTOPILOT ACTUATOR.

4 Remove and discard the lockwire and remove the screws (6), the flange (3) and the solenoid (2).
5 Remove and discard the packing (1).
6 Put a blanking cap on the liner.

(3)Preparation of Replacement Component
(a)Remove the blanking cap from the liner.
(b)Use cleaning agents (Material No. 11-026) with a clean lint-free cloth to make sure that the actuator and the solenoids (2) and (11) manifold interfaces are fully cleaned.
(c)Apply a coat of common grease (Material No. 04-004) to the manifold interfaces of the actuator and the solenoids (2) and/or (11) and to

the threads of the screws (6) and/or (13).

(d)Use common grease (Material No. 04-015) and lubricate the new packing (1) and/or (10).

(4)Installation

(a)On solenoids (11) with a fixed flange.

1 Use the mounting cone PN 0U130411 to install the new packing (10) onto the liner.

R CAUTION : MAKE SURE THAT THE PUSHROD (9) IS STILL IN PLACE.

2 Put the solenoid (11) in position on the liner and install the screws (13). Use the adaptor PN 0U130423 and TORQUE the screws (13) to between 0.22 and 0.29 m.daN (19.5 and 25.6 lbf.in).

3 Safety the screws (13) with lockwire.

4 Use a brush to apply special material (Material No. 05-005) to the heads of the screws (13).

5 Remove the blanking caps and plugs and examine the ends of the electrical connector (12) for the correct condition.

6 Identify the label or mark on the electrical connector (12) and its related solenoid (11) (Ref. para 4.B.(2)(a)1). Connect the electrical connector (12) to the correct solenoid (11).

7 Safety the electrical connector (12) with lockwire.

(b)On solenoids (2) with a loose flange.

1 Install the new packing (1) onto the liner.

R CAUTION : MAKE SURE THAT THE PUSHROD (9) IS STILL IN PLACE.

2 Put the solenoid (2) and the flange (3) in position on the liner and install the screws (6).

3 Use the adaptor PN 0U130423 and TORQUE the screws (6) to between 0.22 and 0.29 m.daN (19.5 and 25.6 lbf.in).

4 Safety the screws (6) with lockwire.

5 Remove the blanking caps and plugs and examine the ends of the electrical connector (4) for the correct condition.

6 Identify the label or mark on the electrical connector (4) and its related solenoid (2) (Ref. para 4.B.(2)(b)1). Connect the electrical connector (4) to the correct solenoid (2). Install the screw (5).

7 Safety the screw (5) with lockwire.

8 Use a brush to apply corrosion-preventative compound (Material No. 05-005) to the heads of the screws (5) and (6).

(c)Make sure that the control surface ranges are clear.

(d)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

(e)Pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

(f)Remove the support from below the flight control surface(s).
(g)Remove the rigging pins PN 98A27108014000, PN 98A27207543000 and PN 98A27901000000.
(h)Remove safety clips and tags and close circuit breakers 301CU, 305CA1, 306CA1, 312CA1, 312CA2, 306CA2 and 305CA2.
(j)Make sure that the solenoids (2) and (11) are free from hydraulic leaks.

(5)Test
(a)Do the test to make sure that the electrical components of the actuator were connected correctly during the removal/installation procedure (Ref. Para. 2.B.(5)(b)).

(6)Close-Up
(a)De-pressurize the aircraft hydraulic system (Ref. 29-10-00, P. Block 301).
(b)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c)On the actuator and the solenoids (2) and (11), examine the coating of the special material (Material No. 05-027) for good condition and renew as necessary (Ref. 51-75-10, P. Block 801).
(d)Close the access door 675AB.
(e)Make sure that the work area is clean and clear of tools and miscellaneous items of equipment.
(f)Remove the access platform.
(g)Remove the warning notices.

5. FILTER

A. Equipment and Materials

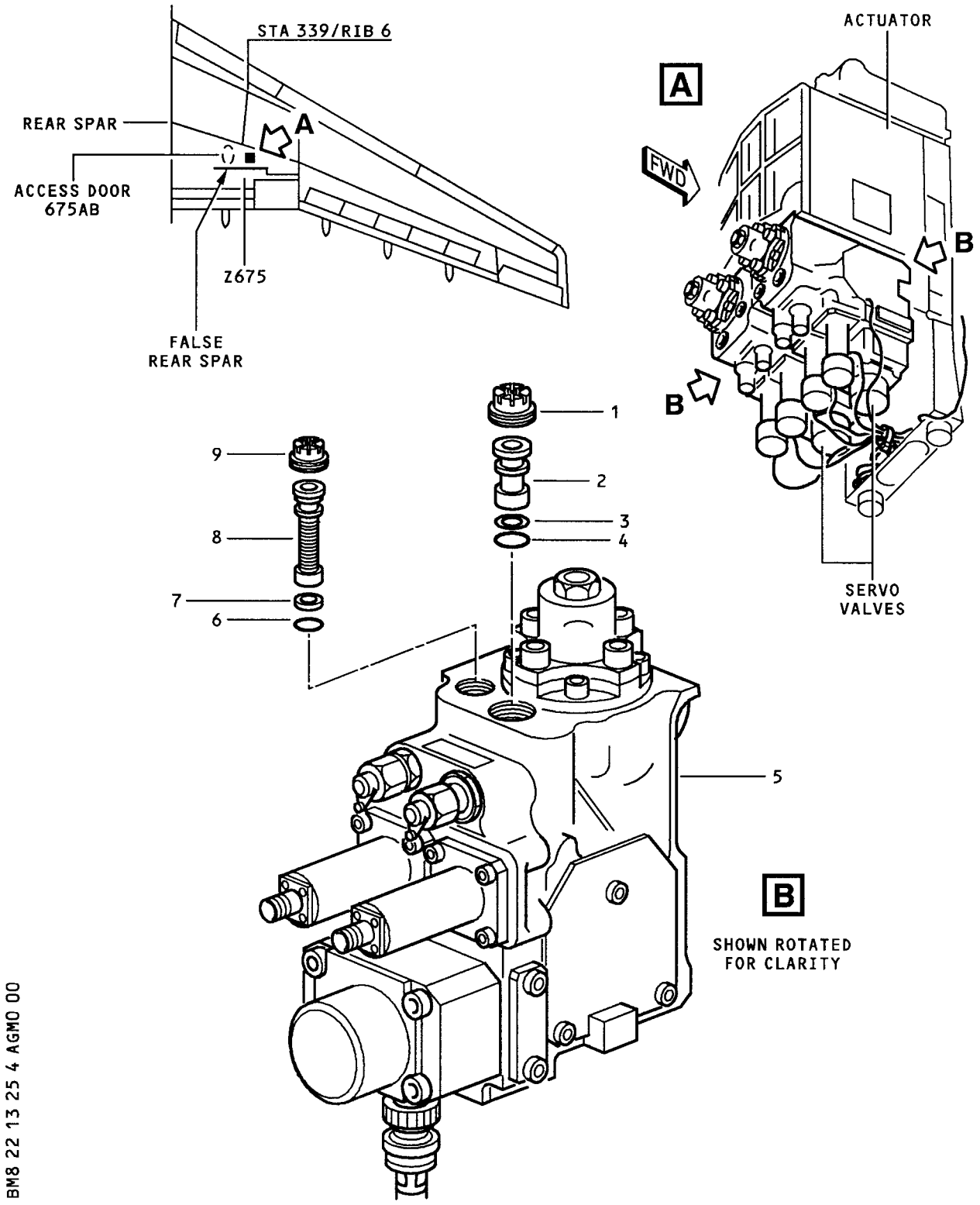
ITEM	DESIGNATION
A.	Access Platform 3 to 6 m (10 to 20 ft)
B.	Circuit Breaker Safety Clips and Tags
C.	Torque Wrench 0 - 0.5 m.daN (0 - 44.2 lbf.in)
D.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
E.	Warning Notice - Prohibiting Unauthorized Use of Hydraulic Systems
F. 0U129485	Shrinking Tool
G. 0U129493	Mounting Cone
H. 0U134344	Adaptor
J. 98A27108014000	Pin - Rigging
K. 98A27207543000	Pin - Rigging
L. 98A27901000000	Pin - Rigging
M.	Brush
N.	Lint-Free Cloth
P.	Threaded Rod, (8 - 32 UNC 3A)
Q.	Lockwire, Corrosion-Resistant Steel 0.8 mm (0.032 in.) dia.
R.	Applicable Support to prevent droop of the Flight Control Surface(s)
S. Material No. 04-004	Common Grease (Ref. 20-31-00)
T. Material No. 04-015	Common Grease (Ref. 20-31-00)
U. Material No. 05-005	Special Materials (Ref. 20-31-00)
V. Material No. 05-027	Special Materials (Ref. 20-31-00)
W. Material No. 11-026	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization)
- 51-75-10, P. Block 801	Repair Paint Coatings

B. Procedure
(Ref. Fig. 404)

NOTE : The removal/installation procedure is the same for the LH or the RH filter (8).

(1)Job Set-Up
(a)Put the access platform in position below zone 675.
(b)Pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.



Autopilot Roll-Actuator, Filter/Strainer
Figure 404

(c)Make sure that the aileron trim is set at zero.
(d)Make sure that the aileron control wheel is central and install the rigging pin PN 98A27901000000.
(e)Put warning notice in the flight compartment, prohibiting unauthorized operation of the flight controls.
(f)On panel 21VU, open, safety and tag the AUTO FLIGHT SYSTEM circuit breakers breakers that follow:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301/CU	102/H01
21VU	AFS/AP1/FD1/FCC1 26VAC	305/CA1	102/H02
21VU	AFS/AP1/FD1/FCC1 28VDC	306/CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312/CA1	102/H04
21VU	AFS/AP1/FD1/FCU	312/CA2	102/H05
21VU	AFS/AP2/FD2/FCC2 28VDC	306/CA2	102/H06
21VU	AFS/AP2/FD2/FCC2 26VAC	305/CA2	102/H07

(g)Open the access door 675AB.

CAUTION : MAKE SURE THAT YOU PRESSURIZE THE HYDRAULIC SYSTEM(S) BEFORE YOU INSTALL THE RIGGING PINS. IF YOU DO NOT PRESSURIZE THE HYDRAULIC SYSTEM(S), THE FLIGHT CONTROL SURFACES WILL NOT STAY AT ZERO, BUT WILL DROOP AND PUT A FORCE ON THE RIGGING PINS.

(h)Install the rigging pin PN 98A27207543000 into the outboard side of the actuator, and the rigging pin PN 98A27108014000 into the aileron cable-drum differential-droop unit.
(j)Remove the support from below the flight control surface(s).
(k)De-pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).
(l)Put the warning notice in the flight compartment prohibiting unauthorized use of the hydraulic system(s).

(2)Removal

CAUTION: TOOLS AND OTHER EQUIPMENTS, INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69 mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

DO NOT ALLOW DUST, DIRT OR METAL FILINGS TO CONTAMINATE THE FILTER SURFACES OR TO ENTER THE INTERIOR OF THE ROLL ACTUATOR. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

(a)Remove and discard the lockwire from the nut (9).
(b)Use the adaptor PN 0U134344 to remove the nut (9).
(c)To remove the filter (8), install a threaded rod (8 - 32 UNC 3A) into the body of the filter (8) and remove it complete from the hydraulic

unit (5). Remove the threaded rod from the filter (8).
(d)Remove and discard the packing (6) and the ring (7).

(3)Preparation of Replacement Component
(a)Use cleaning agents (Material No. 11-026) with a clean lint-free cloth to make sure that the interface threads of the hydraulic unit (5) and the filter (8) are fully cleaned.
(b)Use common grease (Material No. 04-015) and lubricate the new packing (6) and the new ring (7).
(c)Use the mounting cone PN 0U129493 and install the packing (6) onto the filter (8).

NOTE: Installation of the packing (6) must be completed before the ring (7) is installed.

(d)Use the shrinking tool PN 0U129485 and install the ring (7) on the filter (8).

(4)Installation
(a)Install the filter (8) into the hydraulic unit (5).
(b)Apply a coat of common grease (Material No. 04-004) to the threads of the nut (9).
(c)Use the adaptor PN 0U134344 to install the nut (9). TORQUE the nut (9) to between 0.4 and 0.5 m.daN (35 and 44 lbf. in.).
(d)Safety the nut (9) with lockwire.
(e)Use a brush to apply special material (Material No. 05-005) to the head of the nut (9).
(f)Make sure that the control surface ranges are clear.
(g)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(h)Pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

(j)Remove the support from below the flight control surface(s).
(k)Remove the rigging pins PN 98A27108014000, PN 98A27207543000 and PN 98A27901000000.
(l)Remove safety clips and tags and close the circuit breakers 301CU, 305CA1, 306CA1, 312CA1, 312CA2, 306CA2 and 305CA2.
(m)Make sure that the hydraulic unit (5) and the filter (8) are free from hydraulic leaks.

(5)Test
(a)Do the test to make sure that the electrical components of the actuator were connected correctly during the removal/installation procedure (Ref. Para. 2.B.(5)(b)).

(6)Close-Up
(a)De-pressurize the aircraft hydraulic system (Ref. 29-10-00,

P. Block 301).
(b)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c)On the area around the nut (9) and the hydraulic unit (5) examine the coating of the special material (Material No. 05-027) for good condition and renew as necessary (Ref. 51-75-10, P. Block 801).
(d)Make sure that the work area is clean and clear of tools and miscellaneous items of equipment.
(e)Close the access door 675AB.
(f)Remove the access platform.
(g)Remove the warning notices.

6. STRAINER

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 3 to 6 m (10 to 20 ft)
B.	Circuit Breaker Safety Clips and Tags
C.	Torque Wrench 0 - 0.5 m.daN (0 - 44.2 lbf.in)
D.	Warning Notice - Prohibiting Unauthorized Use of Flight Controls
E	
F.	Warning Notice - Prohibiting Unauthorized Use of Hydraulic Systems
G. 0U129455	Mounting Cone
H. 0U131396	Adaptor
J. 0U131397	Shrinking Tool
K. 98A27108014000	Pin - Rigging
L. 98A27207543000	Pin - Rigging
M. 98A27901000000	Pin - Rigging
N.	Brush
P.	Lint-Free Cloth
Q.	Lockwire, Corrosion Resistant Steel 0.8 mm (0.032 in.) dia.
R.	Threaded Rod, (8 - 32 UNC 3A)
S.	Applicable Support to prevent droop of the Flight Control Surface(s)
T. Material No. 04-004	Common Grease (Ref. 20-31-00)
U. Material No. 04-015	Common Grease (Ref. 20-31-00)
V. Material No. 05-005	Special Materials (Ref. 20-31-00)
W. Material No. 05-027	Special Materials (Ref. 20-31-00)
X. Material No. 11-026	Cleaning Agent (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization)
- 51-75-10, P. Block 801	Repair Paint Coatings

B. Procedure (Ref. Fig. 404)

NOTE : The removal/installation procedure is the same for the LH or the the RH strainer (2).

(1)Job Set-Up
(a)Put the access platform in position below zone 675.
(b)Pressurize the aircraft hydraulic system(s) (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

(c)Make sure that the aileron trim is set at zero.

(d)Make sure that the aileron control wheel is central and install the rigging pin PN 98A27901000000.
(e)Put the warning notice in the flight compartment, prohibiting unauthorized operation of the flight controls.
(f)On panel 21VU, open, safety and tag the AUTO FLIGHT SYSTEM circuit breakers that follow:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301/CU	102/H01
21VU	AFS/AP1/FD1/FCC1 26VAC	305/CA1	102/H02
21VU	AFS/AP1/FD1/FCC1 28VDC	306/CA1	102/H03
21VU	AFS/AP1/FD1/FCU	312/CA1	102/H04
21VU	AFS/AP1/FD1/FCU	312/CA2	102/H05
21VU	AFS/AP2/FD2/FCC2 28VDC	306/CA2	102/H06
21VU	AFS/AP2/FD2/FCC2 26VAC	305/CA2	102/H07

(g)Open the access door 675AB.

CAUTION : MAKE SURE THAT YOU PRESSURIZE THE HYDRAULIC SYSTEM(S) BEFORE YOU INSTALL THE RIGGING PINS. IF YOU DO NOT PRESSURIZE THE HYDRAULIC SYSTEM(S), THE FLIGHT CONTROL SURFACES WILL NOT STAY AT ZERO, BUT WILL DROOP AND PUT A FORCE ON THE RIGGING PINS.

(h)Install the rigging pin PN 98A27207543000 into the outboard side of the actuator, and the rigging pin PN 98A27108014000 into the aileron cable-drum differential-droop unit.
(j)Put a support in position below the applicable flight control surface(s), to prevent droop.
(k)De-pressurize the aircraft hydraulic systems (Ref. 29-10-00, P. Block 301).
(l)Put the warning notice in the flight compartment prohibiting unauthorized use of the aircraft hydraulic system(s).

(2)Removal

CAUTION : TOOLS AND OTHER EQUIPMENTS, INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69 mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

DO NOT ALLOW DUST, DIRT OR METAL FILINGS TO CONTAMINATE THE FILTER SURFACES OR TO ENTER THE INTERIOR OF THE ROLL ACTUATOR. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

(a)Remove and discard the lockwire from the nut (1).
(b)Use the adaptor PN 0U131396 to remove the nut (1).
(c)To remove the strainer (2), install a threaded rod (8 - 32 UNC 3A) into the body of the strainer (2) and remove it complete from the

hydraulic unit (5). Remove the threaded rod from the strainer (2).
(d)Remove and discard the packing (4) and the ring (3).

(3)Preparation of Replacement Component
(a)Use cleaning agent (Material No. 11-026) with a clean lint-free cloth to make sure that the interface threads of the hydraulic unit (5) and the strainer (2) are fully cleaned.
(b)Use common grease (Material No. 04-015) and lubricate the packing (4) and the ring (3).
(c)Use the mounting cone PN 0U129455 and install the new packing (4) on the strainer (2).

NOTE: Installation of the packing (4) must be completed before the ring (3) is installed.

(d)Use the shrinking tool PN 0U131397 and install the new ring (3) on the strainer (2).

(4)Installation
(a)Install the strainer (2) into the hydraulic unit (5).
(b)Apply a coat of common grease (Material No. 04-004) to the threads of the nut (1).
(c)Use the adaptor PN 0U131396 to install the nut (1). TORQUE the nut (1) to between 0.4 and 0.5 m.daN. (35 to 44 lbf. in.).
(d)Safety the nut (1) to the nut (9) with lockwire.
(e)Use a brush to apply special material (Material No. 05-005) to the head of the nut (1).
(f)Make sure that the control surface ranges are clear.
(g)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(h)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301).

NOTE : When the hydraulic system(s) are pressurized, the flight control surfaces will not droop and put a force on the rigging pins.

(j)Remove the support from below the flight control surface(s).
(k)Remove the rigging pins PN 98A27108014000, PN 98A27207543000 and PN 98A27901000000.
(l)Remove safety clips and tags and close circuit breakers 301CU, 305CA1, 306CA1, 312CA1, 312CA2, 306CA2 and 305CA2.
(m)Make sure that the hydraulic unit (5) and the strainer (2) are free from hydraulic leaks.

(5)Test
(a)Do the test to make sure that the electrical components of the actuator were connected correctly during the removal/installation procedure (Ref. Para. 2.B.(5)(b)).

(6)Close-Up
(a)De-pressurize the aircraft hydraulic system (Ref. 29-10-00,

P. Block 301).
(b) De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(c) On the area around the nut (1) and the hydraulic unit (5), examine the coating of the special material (Material No. 05-027) for good condition and renew as necessary (Ref. 51-75-10, P. Block 801).
(d) Close the access door 675AB.
(e) Make sure that the work area is clean and clear of tools and miscellaneous items of equipment.
(f) Remove the access platform.
(g) Remove the warning notices.

AUTOPILOT YAW ACTUATOR 22CA - REMOVAL/INSTALLATION

WARNING : CHECK THAT LANDING GEAR GROUND SAFETIES, INCLUDING WHEEL CHOCKS, ARE IN POSITION.
BEFORE APPLYING OR RELIEVING HYDRAULIC SYSTEM PRESSURE, MAKE CERTAIN THAT TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, CHECK THAT ALL CONTROLS ARE SET TO CORRESPOND WITH THE ACTUAL POSITION OF THE SERVICES THEY OPERATE.
BEFORE PROCEEDING WITH MAINTENANCE WORK ON OR NEAR MECHANICAL FLIGHT CONTROLS OR PRIMARY FLIGHT CONTROL SURFACES, LANDING GEARS, ASSOCIATED DOORS OR ANY MOVING COMPONENT, MAKE CERTAIN THAT GROUND SAFETIES AND/OR WARNING NOTICES ARE IN CORRECT POSITION TO PREVENT INADVERTENT OPERATION OF CONTROLS.

1. Autopilot Actuator

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 5.5 m (18 ft.)
B. 98A27207543000	Pin - Rigging
C. 98A27307546000	Rigging Pin - Main Bellcrank, Rudder Mechanical Controls
D. 98A27901006000	Pin - Rigging, Rudder Control (Qty. 2)
E.	Torque Wrench, 0.5 m.daN (3.69 lbf.ft.)
F.	Circuit Breaker Safety Clips
G.	Blanking Caps - Electrical Connector
H.	Blanking Plugs/Caps - Hydraulic Line and Port
J.	Warning Notices
K.	Corrosion-Resistant Steel Lockwire, 0.6 mm (0.024 in.) dia.
L. Material No. 05-027	Special Materials (Ref. 20-31-00)

Referenced Procedures

- 20-28-11, P. Block 1	Electrical Bonding
- 22-10-00, P. Block 501	Autopilot
- 22-13-28, P. Block 401	Yaw Torque Limiter Lever 23CA
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-22-00, P. Block 301	Blue Auxiliary Power (Power Transfer Unit)
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 51-75-10, P. Block 801	Repair of Paint Coatings

B. Procedure (Ref. Fig. 401)

(1)Job Set-Up
(a)Position access platform.
R (b)When the rudder trim is set to zero and the rigging pins P/N

R 98A27307546000 and 98A27901006000 are installed, the hydraulic
R systems should then be de-pressurized.
R (c)Make certain that rudder trim is set at zero.
R (d)Make certain that rudder pedals are set at neutral and install rigging pins P/N 98A27901006000 (Qty. 2) in rudder control bellcranks on control mechanism mounting beam in avionics compartment.
R (e)Display warning notices in flight compartment prohibiting operation of flight controls.
R (f)Open, safety and tag the following circuit breakers:

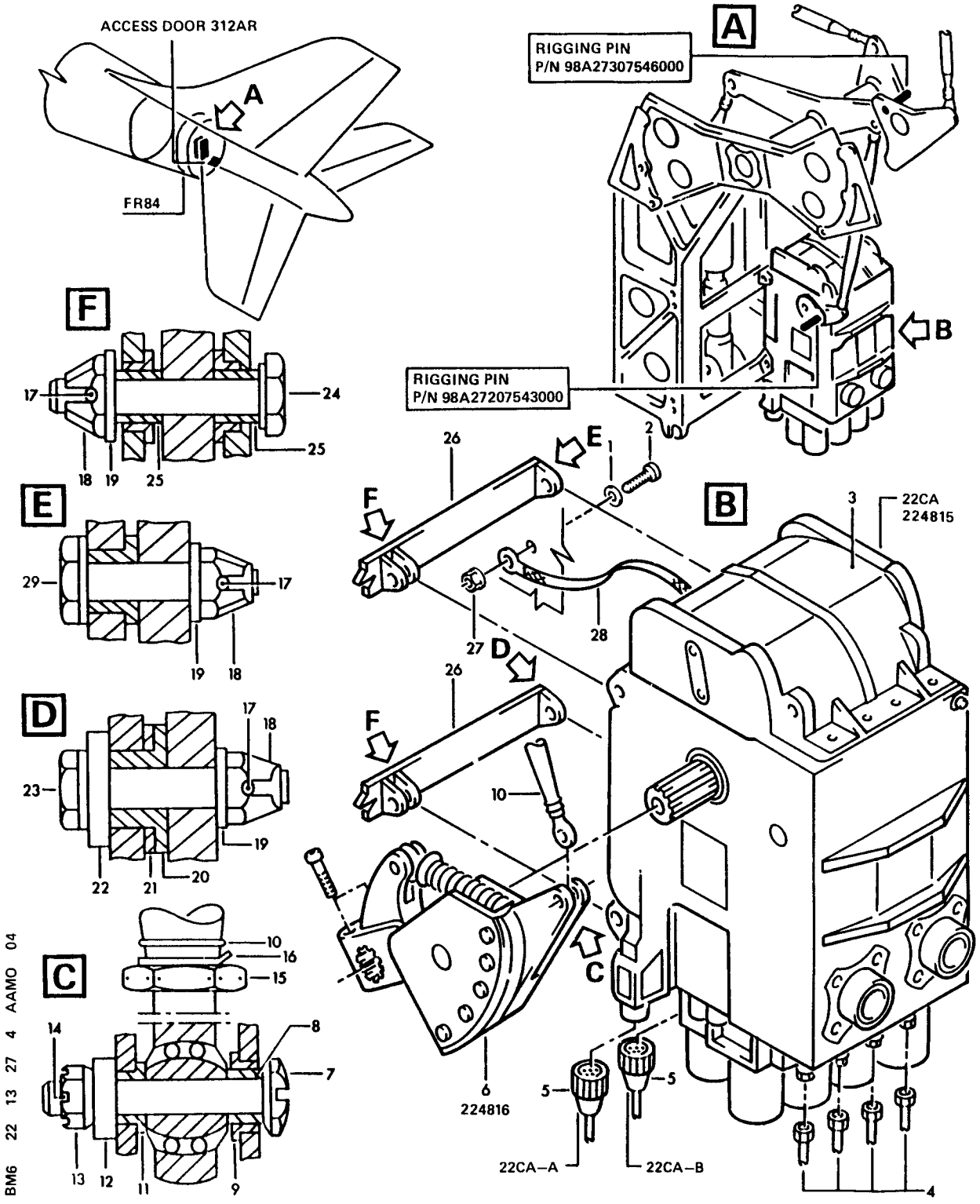
PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/26VAC	305CA1	102/H 2
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/AP1-FD1/FCU	312CA1	102/H 4
21VU	AFS/AP2-FD2/FCU	312CA2	102/H 5
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/AP2-FD2/FCC2/26VAC	305CA2	102/H 7
21VU	AFS/MTP	301CU	102/H 1

(g)Open access door 312AR.
(h)Install rigging pin P/N 98A27307546000 in rudder control main bellcrank.
R (i)Depressurize hydraulic systems (Ref. 29-10-00, P. Block 301) and
R display warning notices, prohibiting repressurization.

(2)Removal (Ref. Fig. 401)
(a)Disconnect electrical connectors (5) and install blanking caps.
(b)Disconnect hydraulic lines (4) and install blanking plugs.
(c)Remove nut (27), screw (2), washer (1) and disconnect bonding strap (28) from rudder control frame (26).
(d)Disconnection of pushrod (10).
1 Remove cotter pin (14), nut (13) washer (12) and bolt (7).
2 Disconnect pushrod (10) and secure to structure.
3 Remove spacer (8).
4 Secure bushes (9, 11) temporarily.
(e)Removal of autopilot yaw actuator (3).
1 Remove cotter pins (17), nuts (18) and washers (19).
CAUTION : AUTOPILOT YAW ACTUATOR WEIGHS APPROXIMATELY 20 kg (44 lb.).
2 Support autopilot yaw actuator (3) and remove bolts (23, 24, 29) and cupwasher (22).
3 Remove autopilot yaw actuator (3).
4 Secure spacers (25) and bush (20) including shim (21) temporarily.

(3)Preparation of Replacement Components (Ref. Fig. 401)
(a)Transfer bonding strap (28) from removed to replacement component. For electrical bonding refer to 20-28-11, P. Block 1.
(b)Transfer yaw torque limiter lever (6) from removed to replacement component (Ref. 22-13-28, P. Block 401).

(4)Installation (Ref. Fig. 401)



Autopilot Yaw Actuator
Figure 401

R

(a)Installation of autopilot yaw actuator.
1 Position autopilot yaw actuator (3).
2 Make certain that spacers (25) and bush (20), including shim (21), are installed.
3 Install cupwasher (22) and bolts (23, 24, 29).
4 Install washers (19) and nuts (18).
5 TORQUE nuts (18) to between 0.21 and 0.23 m.daN (18.6 and 20.4 lbf.in.) and safety with cotter pins (17).
CAUTION : MAKE SURE THAT YOU CONNECT THE HYDRAULIC LINES TO THE CORRECT HYDRAULIC CONNECTIONS ON THE ACTUATOR. IF YOU DO NOT CONNECT THE LINES CORRECTLY, THE TEST (AFTER YOU COMPLETED INSTALLATION) WILL FAIL.
6 Remove blanking plugs and connect and tighten hydraulic lines (4).
7 Connect bonding strap (28) with screw (2), washer (1) and nut (27). (Ref. 20-28-11, P. Block 1).
8 Remove blanking caps and connect electrical connectors (5)
LH connection - 22CA-A
RH connection - 22CA-B
(b)Make certain that control surface travel ranges are clear.
(c)Pressurize hydraulic systems (Ref. 29-10-00, P. Block 301).
NOTE : As an alternative use electrical pumps and PTU (Ref. 29-21-00, 29-22-00 and 29-23-00, P. Block 301).
(d)Check that hydraulic connections are free from leaks.
(e)Install rigging pin P/N 98A27207543000 in autopilot yaw actuator.
(f)Installation of push rod (10).
1 Make certain that bushes (9, 11) are installed and install spacer (8).
2 Connect pushrod (10) to yaw torque limiter lever (6) with bolt (7).
3 If bolt (7) cannot be easily inserted, adjust pushrod (10) as follows:
-Cut and remove lockwire and loosen nuts (15).
-Adjust length of pushrod (10) until bolt (7) can be easily inserted.
-Tighten nuts (15) and safety with lockwire to lockwashers (16).
4 Install washer (12) and nut (13).
5 TORQUE nut (13) to between 0.21 and 0.23 m.daN (18.6 and 20.4 lbf.in) and safety with cotter pin (14).
(g)Remove rigging pins P/N 98A27901006000 (Qty. 2), P/N 98A27307546000 and P/N 98A27207543000.
(h)Remove safety clips and tags, and close circuit breakers 305CA1, 306CA1,312CA1, 312CA2, 306CA2, 305CA2 and 301CU.
R (i)De-pressurize hydraulic systems (Ref. 29-10-00, P. Block 301).

(5)Test
R (a)Pressurize the green and yellow hydraulic systems (Ref. 29-10-00,
R P. Block 301).
R (b)Initial conditions
- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct
- Overhead panel circuit breakers closed, particularly those associated with AFS, NAVIGATION, ADS and ECAM.

- Aircraft in ground configuration (**LDG GEAR PROX DET & RELAYS** circuit breakers on panel **133VU** closed).
- **ADS** operating (Ref. 34-10-00, P. Block 501).
- **IRS** 1, 2 and 3 aligned (perform **ALIGN** procedure until red **HDG** flags on **NDS** disappear, present position entering is not necessary) (Ref. 34-25-00, P. Block 501)
- **EFIS** operating by adjustment on panels **412VU**, **413VU** of **PFD** and **ND** brightness.
- On center pedestal, **ECAM** display unit brightness adjusted and **FLT CTL** page selected.
- On overhead panel, **PITCH TRIM** 1 and **YAW DAMPER** 1 engaged (Ref. 22-26-00, P. Block 1).
- Extend the slats fully (Ref. 27-80-00, P. Block 301).

R (c) Do the test that follows to check correct behaviour of servomotor and to make sure that the electrical subparts of servomotor are correctly connected following servomotor or subpart replacement.

NOTE : If you open or close an **FCC C/B** during the subsequent work steps, you must wait a minimum of one minute to permit the **FCC** to power-up or power-down, as necessary.

ACTION	RESULT
1. On **SERVO CTL** section of panel **428VU**: - release Y p/bsw.	On **428VU**: - corresponding **OFF** legend comes on. - corresponding **SERVO CTL/LO PR/Y** warning light comes on. On the left **ECAM** display unit: - **FLT CTL YELLOW SERVO LO PR** comes on.
2. On panel **21VU**: - open the **AP2-FD2/FCC2/28DCD** circuit breaker **306CA2 (102/H6)**.	The **FCC2** is de-energized.
3. On the flight control unit (**FCU**): - put the **AP1** lever in the **ON** position. - push (in) the **CWS/CMD** p/bsw.	The **AP1** lever is engaged. The **AP1** lever stays engaged. In the **CWS/CMD** p/bsw, the green **CMD** light comes on.
- if the **AP1** lever does not stay engaged.	- stop the test and check electrical connections.
4. On the flight control unit (**FCU**): - with **HDG SEL** selector knob,	On the **NDS**: - the heading index moves in

ACTION	RESULT
select heading deviation of 10 deg. to the right.	relation.
- pull **HDG SEL** selector knob.	On the **FCU**: - **HDG SEL P/BSW** comes on green. On the **PFDs**: - the green **HDG** goes off. - the green **HDG/S** comes on. - the control wheel rotates to the right. **RH** rudder pedals move forward. On right **ECAM** display unit, **RUD** scale: - bug moves to the right.
- select a heading deviation equal to the aircraft heading.	- the control wheel and rudder returns towards the neutral position. On right **ECAM** display unit, **RUD** scale: - bug returns towards zero.
NOTE : The pedal deflection will not be fully cancelled except if a left **HDG SEL** selection is temporarily applied.	
- using **HDG SEL** selector knob, select heading deviation to the left, then select a heading deviation equal to the heading to center the control wheel.	
5. - disengage the **AP1** lever.	The **AP1** lever is disengaged. - check that rudder does not move.
6. On **SERVO CTL** section of panel **428VU**: - press Y p/bsw.	On **428VU**: - corresponding **OFF** legend goes off. - corresponding **SERVO CTL/LO PR/Y** warning light goes off. On the left **ECAM** display unit: - **FLT CTL YELLOW SERVO LO PR** goes off.
- release G p/bsw.	- corresponding **OFF** legend comes on. - corresponding **SERVO CTL/LO PR/G**

ACTION	RESULT
	warning light comes on. On the left ECAM display unit: - FLT CTL GREEN SERVO LO PR comes on.
7. On panel 21VU: - close the AP2-FD2/FCC2/28VDC circuit breaker. 306CA2 (102/H6).	The FCC2 is energized.
- open the AP1-FD1/FCC1/28VDC circuit breaker. 306CA1 (102/H3).	The FCC1 is de-energized.
8. On the flight control unit (FCU): - put the AP2 lever in the ON position.	The AP2 lever is engaged.
- push (in) the CWS/CMD p/bsw.	The AP2 lever stays engaged. In the CWS/CMD p/bsw, the green CMD light comes on.
- if the AP2 lever does not stay engaged.	- stop the test and check electrical connections.
9. On the flight control unit (FCU): - with HDG SEL selector knob, select heading deviation of 10 deg. to the right.	On the NDS: - the heading index moves in relation.
- pull HDG SEL selector knob	On the FCU: - HDG SEL P/BSW comes on green. On the PFDs: - the green HDG goes off. - the green HDG/S comes on. - the control wheel rotates to the right. RH rudder pedals move forward. On right ECAM display unit, RUD scale: - bug moves to the right.
- select a heading deviation equal to the aircraft heading.	- the control wheel and rudder returns towards the neutral position. On right ECAM display unit,

ACTION	RESULT
	RUD scale: - bug returns towards zero.
NOTE : The pedal deflection will not be fully cancelled except if a left HDG SEL selection is temporarily applied.	
- using HDG SEL selector knob, select heading deviation to the left, then select a heading deviation equal to the heading to center the control wheel.	
10. - disengage the AP2 lever.	The AP2 lever is disengaged. - check that rudder does not move.
11. On panel 21VU: - close the AP1-FD1/FCC1/28VDC circuit breaker 306CA1 (102/H3).	The FCC1 is energized.
12. On SERVO CTL section of panel 428VU: - press G pushbutton switch.	On panel 428VU: - SERVO CTL LO PR/G warning goes off. - On SERVO CTL/G pushbutton switch OFF legend goes off. On the left ECAM display unit: - FLT CTL GREEN SERVO LO PR goes off.

R (d) Do the test that follows, to make sure that the hydraulic lines to the actuator were connected correctly during the removal/installation procedure.
NOTE : Failure to correctly engage the autopilot system 1 and 2 in the command mode during this test will show that the hydraulic lines are not connected correctly.

ACTION	RESULT
1. Initial condition: - Blue system available for yaw damper 1 engagement (Ref. 29-22-00, P. Block 301). - Green system available for AP1 engagement (Ref. 29-22-00, P. Block 301). - Yellow system must be isolated to check hydraulic tubing connection. - Pitch Trim I must be engaged	

ACTION	RESULT
(Ref. 22-26-00, P. Block 1).	
2. On the flight control unit (FCU): - put the AP1 lever in the ON position. - push (in) the CWS/CMD p/bsw. - if the AP lever does not stay engaged, stop the test and correct the connections of the hydraulic lines. (Ref. 1. B. (5) (f)).	 - the AP1 lever is engaged. - the AP1 lever stays engaged.
3. Disengage the AP1 lever.	- the AP1 lever is disengaged.
4. Change hydraulic pressure configuration - Yellow system available for yaw damper 2 and AP 2 engagement (Ref. 29-23-00, P. Block 301). - Green system must be isolated to check correct hydraulic tubing connection. NOTE : Green system can be isolated at servo control panel if yellow system is pressurized through PTU.	
5. On the flight control unit (FCU): - put the AP2 lever in the on position. - push (in) the CWS/CMD p/bsw. - if the AP1 lever does not stay engaged, stop the test and correct the connections of the hydraulic lines (Ref. 1. B. (5) (f)).	 - the AP2 lever is engaged. - the AP2 lever stays engaged.
6. Disengage the AP2 lever.	- the AP2 lever is disengaged.

R (e)Do the test that follows, to make sure that the rudder does not move when the autopilot is disengaged.
NOTE : Before doing this test:
- Make sure that the rudder pedals are set at neutral

ACTION	RESULT
1. On SERVO CONTROL section of panel 428VU: - release Y pushbutton switch.	- Corresponding OFF legend comes on. - Corresponding SERVO CTL/ LO PR warning light comes on.
2. On flight control unit (FCU): - engage AP1 in ON position.	Lever remains engaged. On FCU: - green CWS comes on. On PFDs: - amber CWS1 indication appears.
- press CWS/CMD pushbutton switch.	On this pushbutton switch: - green CWS goes off - green CMD comes on. On PFDs: - white CMD1 indication appears - amber CWS1 indication disappears - green HDG is displayed.
3. On flight control unit (FCU): - with HDG SEL selector knob select heading deviation of 10- to right.	On NDs: - Heading index moves accordingly.
- pull HDG SEL selector knob.	On FCU: - HDG SEL pushbutton switch comes on green. On PFDs: - green HDG goes off - green HDG V/S comes on Control wheel rotates to the right. RH rudder pedals move forwards. On right ECAM display unit, RUD scale: - bug moves to the right.
- using HDG SEL selector knob, select heading deviation to left to center control wheel then select a heading deviation equal to heading.	- Control wheel returns to neutral position. On right ECAM display unit, RUD scale: - bug returns towards zero.
4. Disengage AP1 and cancel warning.	- Check that rudder does not move.

ACTION	RESULT
5. On SERVO CONTROL section of panel 428VU: - press Y pushbutton switch.	On panel 428VU: - SERVO CTL/LO PR/Y warning light goes off. - On SERVO CTL/Y pushbutton switch OFF legend goes off.
- release G pushbutton switch.	- Corresponding OFF legend comes on. - Corresponding SERVO CTL/ LO PR/G warning light comes on.
6. On flight control unit (FCU): - engage AP2 in CMD.	On PFDs: - white CMD2 is displayed - green HDG is displayed.
7. Carry out same operation as in step 3.	Same results as step 3.
8. Disengage AP2 and cancel warning.	- Check that rudder does not move
9. On SERVO CONTROL section of panel 428VU: - press G pushbutton switch.	On panel 428VU: - SERVO CTL LO PR/G warning goes off. - On SERVO CTL/G pushbutton switch OFF legend goes off.
10. On overhead panel: - place PITCH TRIM 1 lever in OFF position.	

R (f)Carry out AP yaw mechanical override test (Ref. 22-10-00, P. Block 501).

R (g)Correction of hydraulic line connections.

NOTE : Use this procedure to correct the hydraulic line connections to the yaw actuator. Do the procedure when the test to engage the autopilot systems 1 and 2 in the command mode fails.

1 Depressurize the green and yellow hydraulic systems (Ref. 29-10-00 P. Block 301).
2 Open, safety and tag the circuit breakers 305CA1, 306CA1, 312CA1, 312CA2, 306CA2, 305CA2 and 301CU (Ref. 1. B. (1) (e)).
3 Place a container under the hydraulic connections.
4 Identify and mark lines and connections for correct installation. Disconnect the hydraulic delivery and return lines (4) from the

actuator.

CAUTION : MAKE SURE THAT YOU CONNECT THE HYDRAULIC LINES TO THE CORRECT HYDRAULIC CONNECTIONS ON THE ACTUATOR. IF YOU DO NOT CONNECT THE LINES CORRECTLY, THE TEST (AFTER YOU HAVE COMPLETED THIS PROCEDURE) WILL FAIL AGAIN.

5 Connect the hydraulic delivery and return lines (4) to the correct positions on the actuator.
6 Tighten the hydraulic lines (Ref. 20-23-12, P. Block 1).
7 Remove the container from under the hydraulic connections.
8 Remove safety clips and tags and close circuit breakers 305CA1, 306CA1, 312CA1, 312CA2, 306CA2, 305CA2 and 301CU (Ref. 1. B. (1) (e)).
9 Make sure that the control ranges are clear and pressurize the green and yellow hydraulic systems (Ref. 29-10-00 P. Block 301).
10 Do the test, Ref. 1. B. (5) (b), to make sure the hydraulic lines are connected correctly.
11 Test completion:
- Shut down and disconnect hydraulic ground power cart (Ref. 29-10-00, P. Block 301).
- De-energize the aircraft electrical network and disconnect electrical ground power unit.
- Restore aircraft and system to normal operating condition.

(6)Close-Up
(a)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(b)On autopilot yaw actuator, check coating of moisture-removing oil (Mat. No. 05-027) for good condition and renew as necessary (Ref. 51-75-10, P. Block 801).
(c)Close access door 312AR.
(d)Remove warning notices.
(e)Remove access platform.

2. Servo-Valve

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 5.5 m (18 ft.)
B. 98A27307546000	Rigging Pin - Main Bellcrank, Rudder Mechanical Controls
C. 98A27901006000	Pin - Rigging, Rudder Control (Qty. 2)
D. 0U130560	Adaptor
E.	Torque Wrench, 0.5 m.daN (3.69 lbf.ft.)
F.	Circuit Breaker Safety Clips
G.	Blanking Caps and Plugs
H.	O-rings
J.	Tie down strap

ITEM	DESIGNATION
K.	Corrosion-Resistant Steel Lockwire, 0.8 mm (0.032 in.) dia.
L.	Warning Notices
M. Material No. 04-015	Common Greases (Ref. 20-31-00)
N. Material No. 04-004	Common Greases (Ref. 20-31-00)
P. Material No. 05-005	Special Materials (Ref. 20-31-00)
Q. Material No. 05-027	Special Materials (Ref. 20-31-00)
R. Material No. 11-004	Cleaning Agents (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 51-75-10, P. Block 801	Repair of Paint Coatings

B. Procedure

NOTE : The removal/installation procedure is the same for the LH or the RH servo-valve (5).

(1)Job Set-Up

(a)Position access platform.

(b)Make certain that rudder trim is set at zero.

(c)Make certain that rudder pedals are set at neutral and install rigging pins P/N 98A27901006000 (Qty. 2) in rudder control bellcranks on control mechanism mounting beam in avionics compartment.

(d)Display warning notices in flight compartment prohibiting operation of flight controls and hydraulic systems.

(e)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/26VAC	305CA1	102/H 2
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/AP1-FD1/FCU	312CA1	102/H 4
21VU	AFS/AP2-FD2/FCU	312CA2	102/H 5
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/AP2-FD2/FCC2/26VAC	305CA2	102/H 7
21VU	AFS/MTP	301CU	102/H 1

(f)Depressurize hydraulic systems (Ref. 29-10-00, P. Block 301) and display warning notices, prohibiting repressurization.

(g)Open access door 312AR.

(h)Install rigging pin P/N 98A27307546000 in rudder control main bellcrank.

(2)Removal (Ref. Fig. 402)

CAUTION : TOOLS AND OTHER EQUIPMENTS, INCLUDING CONTAINERS, MADE OF MAGNETIC MATERIALS, MUST NOT BE USED WITHIN 0.5 in. (12.69 mm) OF THE TORQUE MOTOR LOCATED WITHIN THE SERVO-VALVE. STORE THE REMOVED SERVO-VALVE IN A PROTECTIVE COVER AND AWAY FROM MAGNETIC MATERIALS. FAILURE TO OBEY, COULD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

CAUTION : DO NOT ALLOW DUST, DIRT OR METAL FILINGS TO CONTAMINATE THE INTERIOR PASSAGE OF THE SERVO-VALVE OR YAW ACTUATOR, MANIFOLD. FAILURE TO OBEY, COUD CAUSE A MALFUNCTION OR FAILURE OF THE AUTOPILOT SYSTEM.

(a)Remove and discard the tie down strap that holds the electrical harness between the servo-valves (5).

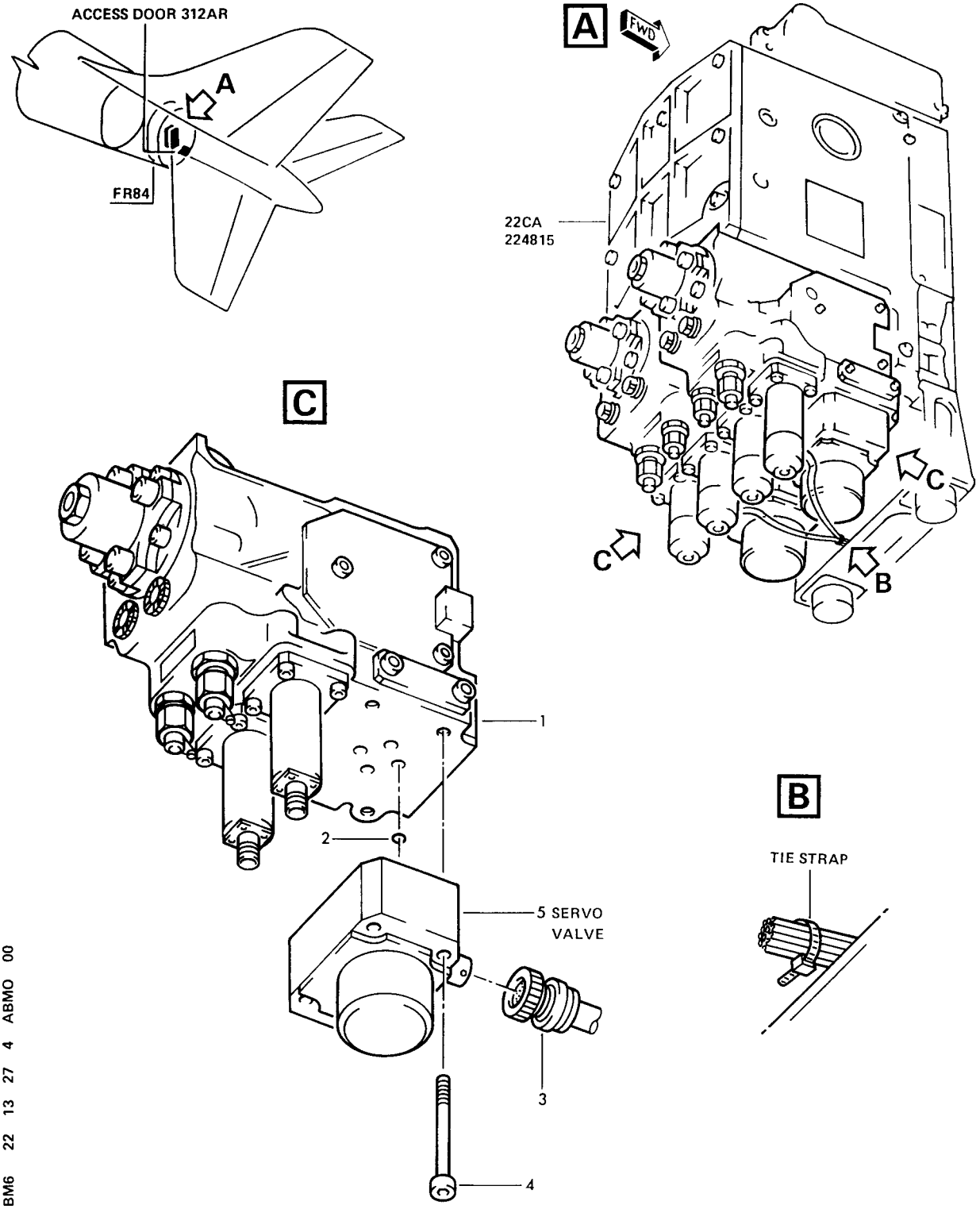
(b)Disconnect the applicable electrical connector (3) from the servo-valve (5).

(c)Cap the open end of the electrical connector (3).

(d)Cut and remove lockwire.

(e)Remove the screws (4) from the applicable servo-valve (5).

(f)Carefully remove the servo-valve (5) from the actuator (1).



Autopilot Actuator Servo-Valves
Figure 402

(g)Remove and discard the O-rings (2) and immediately blank the open manifold face of the actuator (1).
(h)Put the removed servo-valve (5) into a protective cover.

(3)Preparation of Replacement Component (Ref. Fig. 402)
(a)Remove the protective cover from the servo-valve (5) and the blank from the manifold face of the actuator (1).
(b)Use cleaning agent (Mat. No. 11-004) to make sure that the actuator (1) and the servo-valve (5) manifold faces are fully cleand.
(c)Apply a coat of common grease (Mat. No. 04-004) to the manifold faces of the actuator (1) and the servo-valve (5) and to the threads of the screws (4).
(d)Use common grease (Mat. No. 04-015) and lubricate the new O-rings (2).

(4)Installation (Ref. Fig. 402)
(a)Put the new O-rings (2) in position on the mating face of the servo-valve (5).
(b)Put the servo-valve (5) in position on the actuator (1) and install the screws (4)
(c)Use the adaptor P/N 0U130560, and torque the screws (4) to between 0.39 and 0.49 m.daN (34.5 and 43.3 lbf.in).
(d)Safety screws (4) with lockwire.
(e)Use a brush to apply special material (Mat. No. 05-005) to the heads of the screws (4).
(f)Remove the caps, examine the ends of the electrical connector (3) for correct condition then connect the electrcal connector (3) to the servo-valve (5).
(g)Put back the electrical harnes of the actuator (1) between the servo-valves (5) and install a new tie down strap.
(h)Remove the rigging pins P/N 98A27307546000 and P/N 98A27901006000 (Qty. 2).
(j)Remove safety clips and tags and close circuit breakers 301CU, 305CA1, 306CA1, 312CA1, 312CA2, 306CA2 and 305CA2.
(k)Make sure that the control ranges are clear.
(l)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(m)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301).
(n)Make sure that the actuator (1) and the servo-valves (5) are free from hydraulic leaks.

(5)Test
(a)Carry out test discribed in 1. B. (5).

(6)Close-Up
(a)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(b)Close access door 312AR.
(c)Remove warning notices.
(d)Remove access platform.

3. Solenoids

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 5.5 m (18 ft.)
B. 0U130411	Mounting Cone
C. 0U130423	Adaptor
D. 98A27307546000	Rigging Pin - Main Bellcrank, Rudder Mechanical Controls
E. 98A27901006000	Pin - Rigging, Rudder Control (Qty. 2)
F.	Torque Wrench, 0.5 m.daN (3.69 lbf.ft.)
G.	Circuit Breaker Safety Clips
H.	Blanking Caps - Electrical Connector
J.	Corrosion-Resistant Steel Lockwire, 0.6 mm (0.024 in.) dia.
K.	Corrosion-Resistant Steel Lockwire, 0.8 mm (0.032 in.) dia.
L. Material No. 04-004	Common Greases (Ref. 20-31-00)
M. Material No. 04-015	Common Greases (Ref. 20-31-00)
N. Material No. 05-005	Special Materials (Ref. 20-31-00)
P. Material No. 05-027	Special Materials (Ref. 20-31-00)
Q. Material No. 11-004	Cleaning Agents (Ref. 20-31-00)
Referenced Procedures	
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-22-00, P. Block 301	Blue Auxiliary Power (Power Transfer Unit)
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 51-75-10, P. Block 801	Repair of Paint Coatings

B. Procedure
(1)Job Set-Up
(a)Position access platform.
(b)Make certain that rudder trim is set at zero.
(c)Make certain that rudder pedals are set at neutral and install rigging pins P/N 98A27901006000 (Qty. 2) in rudder control bellcranks on control mechanism mounting beam in avionics compartment.
(d)Display warning notices in flight compartment prohibiting operation of flight controls.
(e)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/26VAC	305CA1	102/H 2
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/AP1-FD1/FCU	312CA1	102/H 4
21VU	AFS/AP2-FD2/FCU	312CA2	102/H 5
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/AP2-FD2/FCC2/26VAC	305CA2	102/H 7
21VU	AFS/MTP	301CU	102/H 1

(f)Depressurize hydraulic systems (Ref. 29-10-00, P. Block 301) and display warning notices, prohibiting repressurization.
(g)Open access door 312AR.
(h)Install rigging pin P/N 98A27307546000 in rudder control main bellcrank.

**On solenoids with fixed flange

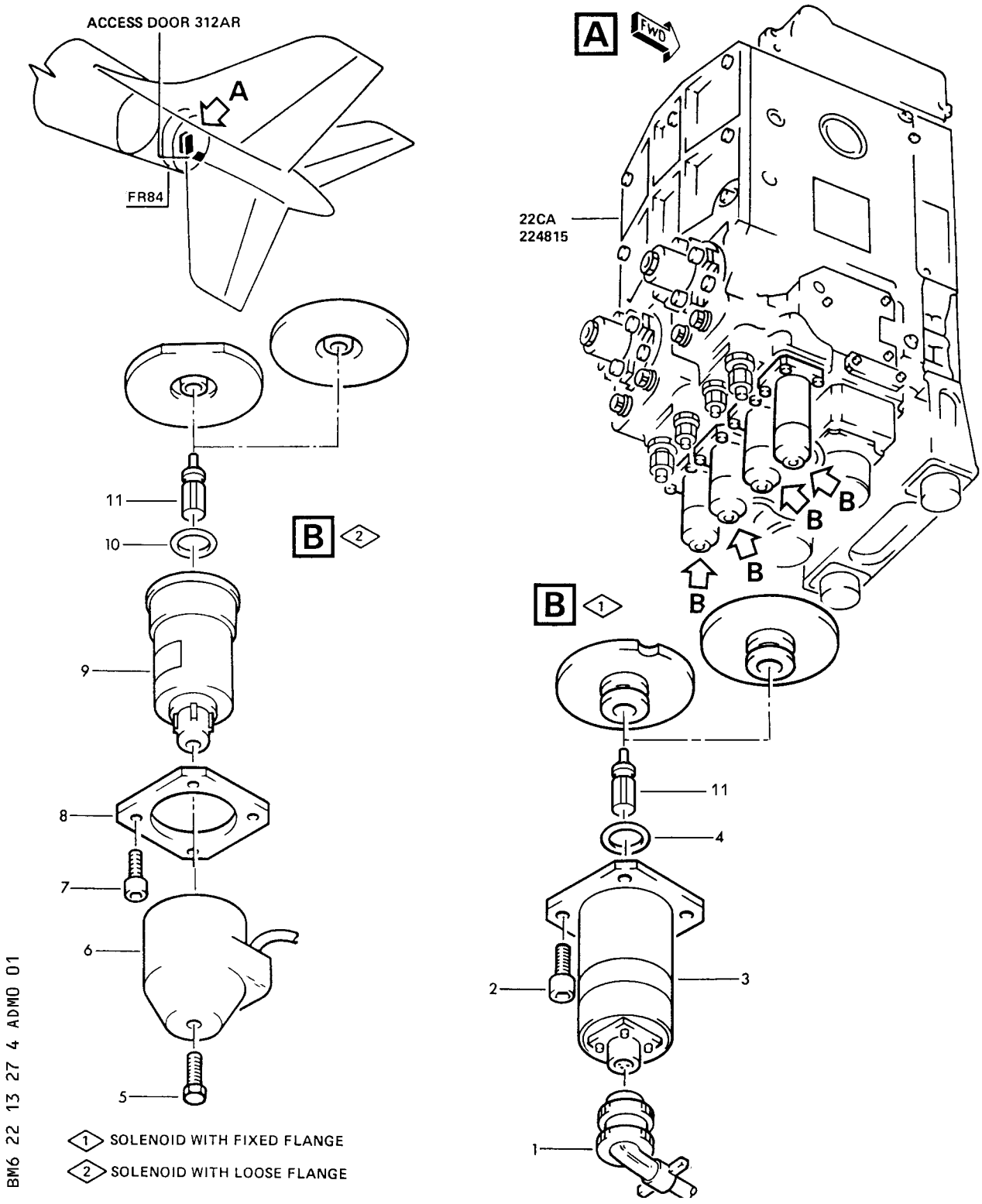
(2)Removal (Ref. Fig. 403)
(a)Put labels or marks on the solenoid to be removed and their related electrical connectors to help you identify the correct connections when you install the solenoids.

CAUTION : DO NOT REMOVE MORE THAN ONE SOLENOID FROM THE ACTUATOR OR DISCONNECT MORE THAN ONE ELECTRICAL CONNECTOR AT THE SAME TIME. YOU MUST RE-INSTALL OR REPLACE THE SOLENOID BEFORE YOU CONTINUE TO REMOVE MORE SOLENOIDS FROM THE ACTUATOR. THIS IS TO PREVENT CROSS CONNECTION OF THE SOLENOIDS.

(b)Cut and remove lockwire and disconnect electrical connector (1) and install blanking caps.
CAUTION : MAKE CERTAIN THAT PUSHROD (11) IS NOT PUSHED OUT BY INTERNAL RESIDUAL PRESSURE OF THE AUTOPILOT ACTUATOR.
(c)Cut and remove lockwire and remove screws (2) and solenoid (3).
(d)Remove and discard packing (4).

**On solenoids with loose flange

(2)Removal (Ref. Fig. 403)



Autopilot Actuator Solenoids
Figure 403

(a)Put labels or marks on the solenoid to be removed and their related electrical connectors to help you identify the correct connections when you install the solenoids.

CAUTION : DO NOT REMOVE MORE THAN ONE SOLENOID FROM THE ACTUATOR OR DISCONNECT MORE THAN ONE ELECTRICAL CONNECTOR AT THE SAME TIME. YOU MUST RE-INSTALL OR REPLACE THE SOLENOID BEFORE YOU CONTINUE TO REMOVE MORE SOLENOIDS FROM THE ACTUATOR. THIS IS TO PREVENT CROSS CONNECTION OF THE SOLENOIDS.

(b)Cut and remove lockwire and remove screw (5).
(c)Disconnect electrical connector (6) and install blanking caps.
CAUTION : MAKE CERTAIN THAT PUSHROD (11) IS NOT PUSHED OUT BY INTERNAL RESIDUAL PRESSURE OF THE AUTOPILOT ACTUATOR.
(d)Cut and remove lockwire and remove screws (7), support flange (8) and solenoid (9).
(e)Remove and discard O-ring (10).

**On solenoids with fixed flange

(3)Installation (Ref. Fig. 403)
(a)Lubricate new packing (4) with grease (Mat. No. 04-015).
(b)Apply a coat of common grease (Mat. No. 04-004) to screws (2).
(c)Install new packing (4) on valve using mounting cone P/N 0U130411.
CAUTION : MAKE CERTAIN THAT PUSHROD (11) IS STILL IN PLACE.
(d)Position solenoid (3) over the valve and install screws (2).
(e)Torque screws (2) to between 0.22 and 0.29 m.daN (19.5 and 25.6 lbf.in) using adaptor P/N 0U130423.
(f)Safety screws (2) with lockwire 0.8 mm (0.032 in.) dia.
(g)Use a brush to apply special material (Mat. No. 05-005) to the heads of the screws (2).
(h)Remove blanking caps and examine the ends of the electrical connector (1) for the correct condition.
(j)Identify the label or mark on the electrical connector (1) and its related solenoid (Ref. para 3. B. (2) (a)). Connect the electrical connector (1) to the correct solenoid.
(k)Safety electrical connector (1) with lockwire 0.6 mm (0.024 in.) dia.

**On solenoids with loose flange

(3)Installation (Ref. Fig. 403)
(a)Lubricate new O-ring (10) with grease (Mat. No. 04-015).
(b)Apply a coat of common grease (Mat. No. 04-004) to screws (7) and (5).
(c)Install new O-ring (10) to valve.
CAUTION : MAKE CERTAIN THAT PUSHROD (11) IS STILL IN PLACE.
(d)Position solenoid (9) over the valve and install support flange (8) with screws (7).
(e)Torque screws (7) to between 0.22 and 0.29 m.daN (19.5 and 25.6 lbf.in) using adaptor P/N 0U130423.
(f)Safety screws (7) with lockwire 0.8 mm (0.032 in.) dia.
(g)Remove blanking caps and examine the ends of the electrical

connector (6) for the correct condition.
(h)Identify the label or mark on the electrical connector (6) and its related solenoid (Ref. para 3. B. (2) (a)). Connect the electrical connector (6) to the correct solenoid. Install the screw (5).
(j)Safety screw (5) with lockwire 0.8 mm (0.032 in.) dia.
(k)Use a brush to apply special material (Mat. No. 05-005) to the heads of the screws (7) and (5).

**On A/C ALL

(l)Remove rigging pins P/N 98A27901006000 (Qty. 2) and P/N 98A27307546000.
(m)Remove safety clips and tags, and close circuit breakers 305CA1, 306CA1,312CA1, 312CA2, 306CA2, 305CA2 and 301CU.
(n)Make certain that control surface travel ranges are clear.
(p)Pressurize hydraulic systems (Ref. 29-10-00, P. Block 301).
(q)Check that hydraulic connections are free from leaks.

(4)Test
(a)Carry out test discribed in 1. B. (5).

(5)Close-Up
(a)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(b)On autopilot yaw actuator, check coating of moisture-removing oil (Mat. No. 05-027) for good condition and renew as necessary (Ref. 51-75-10, P. Block 801).
(c)Close access door 312AR.
(d)Remove warning notices.
(e)Remove access platform.

4. Filters

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 5.5 m (18 ft.)
B. OU129485	Shrinking Tool
C. OU129493	Mounting Cone
D. OU134344	Adaptor
E. 98A27307546000	Rigging Pin - Main Bellcrank, Rudder Mechanical Controls
F. 98A27901006000	Pin - Rigging, Rudder Control (Qty. 2)
G.	Torque Wrench, 0.5 m.daN (3.69 lbf.ft.)
H.	Threaded Rod 8-32 UNC 3A
J.	Circuit Breaker Safety Clips
K.	Corrosion-Resistant Steel Lockwire, 0.8 mm (0.032 in.) dia.
L.	O-ring

ITEM	DESIGNATION
M.	Packing
N.	Warning Notices
P. Material No. 04-015	Common Greases (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization

B. Procedure

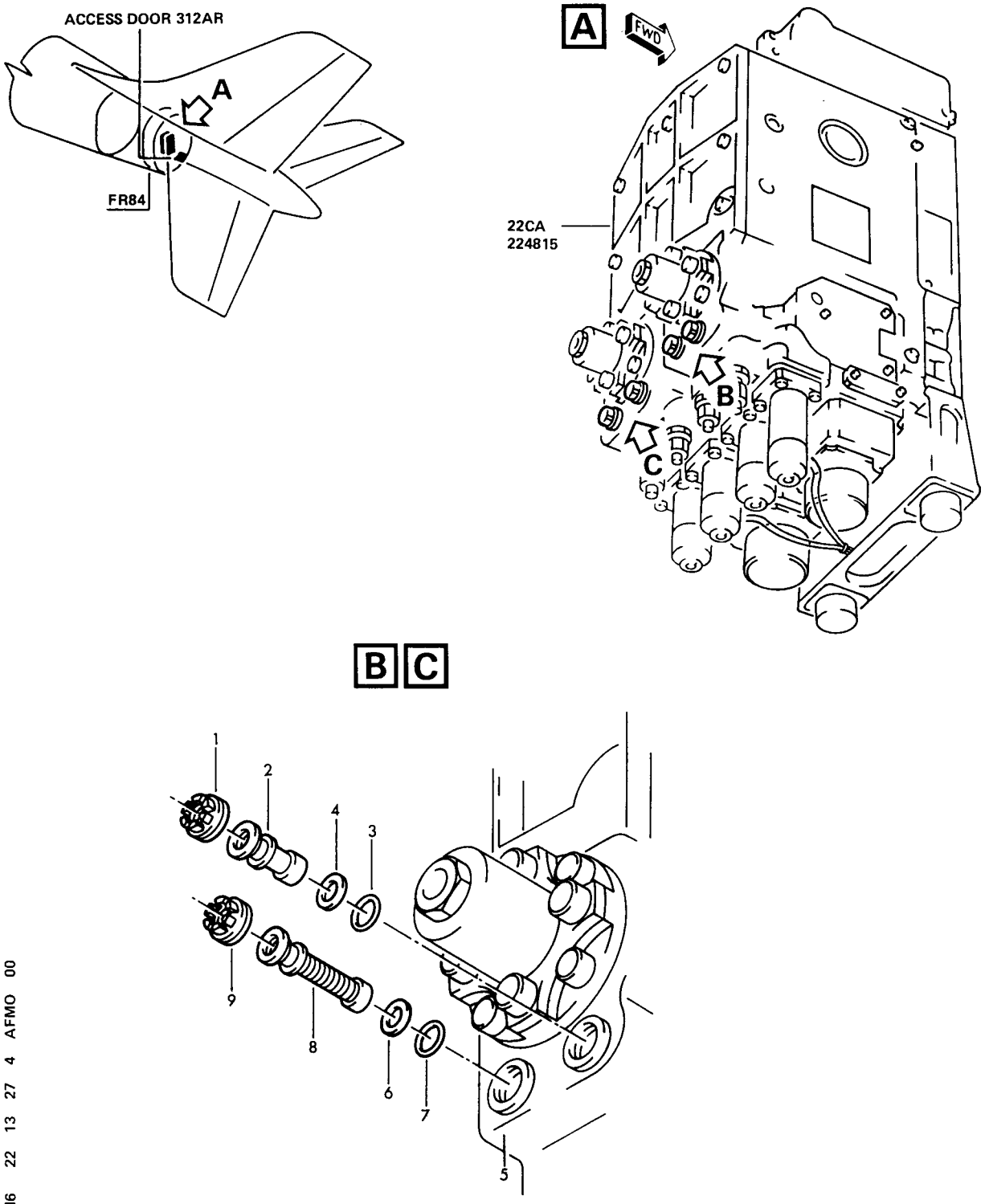
(1)Job Set-Up
(a)Position access platform.
(b)Make certain that rudder trim is set at zero.
(c)Make certain that rudder pedals are set at neutral and install rigging pins P/N 98A27901006000 (Qty. 2) in rudder control bellcranks on control mechanism mounting beam in avionics compartment.
(d)Display warning notices in flight compartment prohibiting operation of flight controls and hydraulic systems.
(e)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/26VAC	305CA1	102/H 2
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/AP1-FD1/FCU	312CA1	102/H 4
21VU	AFS/AP2-FD2/FCU	312CA2	102/H 5
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/AP2-FD2/FCC2/26VAC	305CA2	102/H 7
21VU	AFS/MTP	301CU	102/H 1

(f)Depressurize hydraulic systems (Ref. 29-10-00, P. Block 301) and display warning notices, prohibiting repressurization.
(g)Open access door 312AR.
(h)Install rigging pin P/N 98A27307546000 in rudder control main bellcrank.

(2)Removal (Ref. Fig. 404)
(a)Cut and remove lockwire.
(b)Remove the nut (9) using adaptor 0U134344.
(c)To remove the filter (8), use a threaded rod 8-32 UNC 3A screwed into the filter (8) body and pull out of the housing (5).
(d)Remove and discard the O-ring (7) and packing (6).

(4)Installation (Ref. Fig. 404)
(a)Lubricate the packing (6) and the O-ring (7) with grease (Mat. No. 04-015).
(b)Install the packing (6) on the filter (8) with mounting cone P/N 0U129493.
(c)Shrink packing (6) using shrinking tool P/N 0U129485.
(d)Install the O-ring (7) on the filter (8) with mounting cone P/N 0U129493.
(e)Install the filter (8) into the housing.
(f)Install the nut (9), torque to between 0.4 and 0.5 m.daN (35 and 44 lbf.in) using adaptor P/N 0U134344.
(g)Safety filter (8) to adjacent strainer (2) with lockwire.
(h)Remove the rigging pins P/N 98A27307546000 and P/N 98A27901006000 (Qty. 2).
(j)Remove safety clips and tags and close circuit breakers 301CU, 305CA1, 306CA1, 312CA1, 312CA2, 306CA2 and 305CA2.



Autopilot Actuator Filters and Strainers
Figure 404

(k)Make sure that the control ranges are clear.
(l)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(m)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301).
(n)Make sure that the actuator (1) and the filters (8) are free from hydraulic leaks.

(5)Close-Up
(a)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(b)Close access door 312AR.
(c)Remove warning notices.
(d)Remove access platform.

5. Strainers

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 5.5 m (18 ft.)
B. 0U129455	Mounting Cone
C. 0U131396	Adaptor
D. 0U131397	Precasting Bush
E. 98A27307546000	Rigging Pin - Main Bellcrank, Rudder Mechanical Controls
F. 98A27901006000	Pin - Rigging, Rudder Control (Qty. 2)
G.	Torque Wrench, 0.5 m.daN (3.69 lbf.ft.)
H.	Threaded Rod 8-32 UNC 3A
J.	Circuit Breaker Safety Clips
K.	Corrosion-Resistant Steel Lockwire, 0.8 mm (0.032 in.) dia
L.	O-ring
M.	Packing
N.	Warning Notices
P. Material No. 04-015	Common Greases (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization

B. Procedure
(1)Job Set-Up
(a)Position access platform.
(b)Make certain that rudder trim is set at zero.
(c)Make certain that rudder pedals are set at neutral and install rigging pins P/N 98A27901006000 (Qty. 2) in rudder control bellcranks on control mechanism mounting beam in avionics compartment.
(d)Display warning notices in flight compartment prohibiting operation of flight controls and hydraulic systems.
(e)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/26VAC	305CA1	102/H 2
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/AP1-FD1/FCU	312CA1	102/H 4
21VU	AFS/AP2-FD2/FCU	312CA2	102/H 5
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/AP2-FD2/FCC2/26VAC	305CA2	102/H 7
21VU	AFS/MTP	301CU	102/H 1

(f)Depressurize hydraulic systems (Ref. 29-10-00, P. Block 301) and display warning notices, prohibiting repressurization.
(g)Open access door 312AR.
(h)Install rigging pin P/N 98A27307546000 in rudder control main bellcrank.

(2)Removal (Ref. Fig. 404)
(a)Cut and remove lockwire.
(b)Remove the nut (1) using adaptor P/N 0U131396.
(c)To remove the strainer (2), use a threaded rod 8-32 UNC 3A screwed into the strainer (2) body and pull out of the housing (5).
(d)Remove and discard the O-ring (3) and packing (4).

(4)Installation (Ref. Fig. 404)
(a)Lubricate the O-ring (3) and packing (4) with grease (Mat. No. 04-015).
(b)Install the packing (4) on the strainer (2) with mounting cone P/N 0U129455.
(c)Shrink packing (4) using precasting bush P/N 0U131397.
(d)Install the O-ring (3) on the strainer (2) with mounting cone P/N 0U129455.
(e)Install the strainer (2) into the housing.
(f)Install and tighten the nut (1) with adaptor P/N 0U131396.
(g)Safety strainer (2) to adjacent filter (8) with lockwire.
(h)Remove the rigging pins P/N 98A27307546000 and P/N 98A27901006000 (Qty. 2).
(j)Remove safety clips and tags and close circuit breakers 301CU, 305CA1, 306CA1, 312CA1, 312CA2, 306CA2 and 305CA2.
(k)Make sure that the control ranges are clear.
(l)Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

(m)Pressurize the hydraulic systems (Ref. 29-10-00, P. Block 301).
(n)Make sure that the actuator (1) and strainers (2) are free from hydraulic leaks.

(5)Close-Up
(a)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(b)Close access door 312AR.
(c)Remove warning notices.
(d)Remove access platform.

YAW TORQUE LIMITER LEVER 23CA - REMOVAL/INSTALLATION

WARNING : CHECK THAT LANDING GEAR GROUND SAFETIES, INCLUDING WHEEL CHOCKS, ARE IN POSITION.
BEFORE APPLYING OR RELIEVING HYDRAULIC SYSTEM PRESSURE, MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, CHECK THAT ALL CONTROLS ARE SET TO CORRESPOND WITH THE ACTUAL POSITION OF THE SERVICES THEY OPERATE.
BEFORE PROCEEDING WITH MAINTENANCE WORK ON OR NEAR MECHANICAL FLIGHT CONTROLS OR PRIMARY FLIGHT CONTROL SURFACES, LANDING GEARS, ASSOCIATED DOORS OR ANY MOVING COMPONENT, MAKE CERTAIN THAT GROUND SAFETIES AND/OR WARNING NOTICES ARE IN CORRECT POSITION TO PREVENT INADVERTENT OPERATION OF CONTROLS.

1. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 5.5 m (18 ft.)
B. 98A27207543000	Pin - Rigging
C. 98A27307546000	Rigging Pin - Main Bellcrank, Rudder Mechanical Controls
D. 98A27901006000	Pin - Rigging, Rudder Control (Qty. 2)
E.	Torque Wrench, 0.5 m.daN (3.69 lbf.ft.)
F.	Circuit Breaker Safety Clips
G.	Blanking Caps - Electrical Connector
H.	Blanking Plugs/Caps - Hydraulic Line and Port
J.	Warning Notices
K.	Corrosion-Resistant Steel Lockwire, 0.6 mm (0.024 in.) dia.
L. Material No. 05-027	Special Materials (Ref. 20-31-00)

Referenced Procedures

- 22-10-00, P. Block 501	Autopilot
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 51-75-10, P. Block 801	Repair of Paint Coatings

2. Procedure (Ref. Fig. 401)

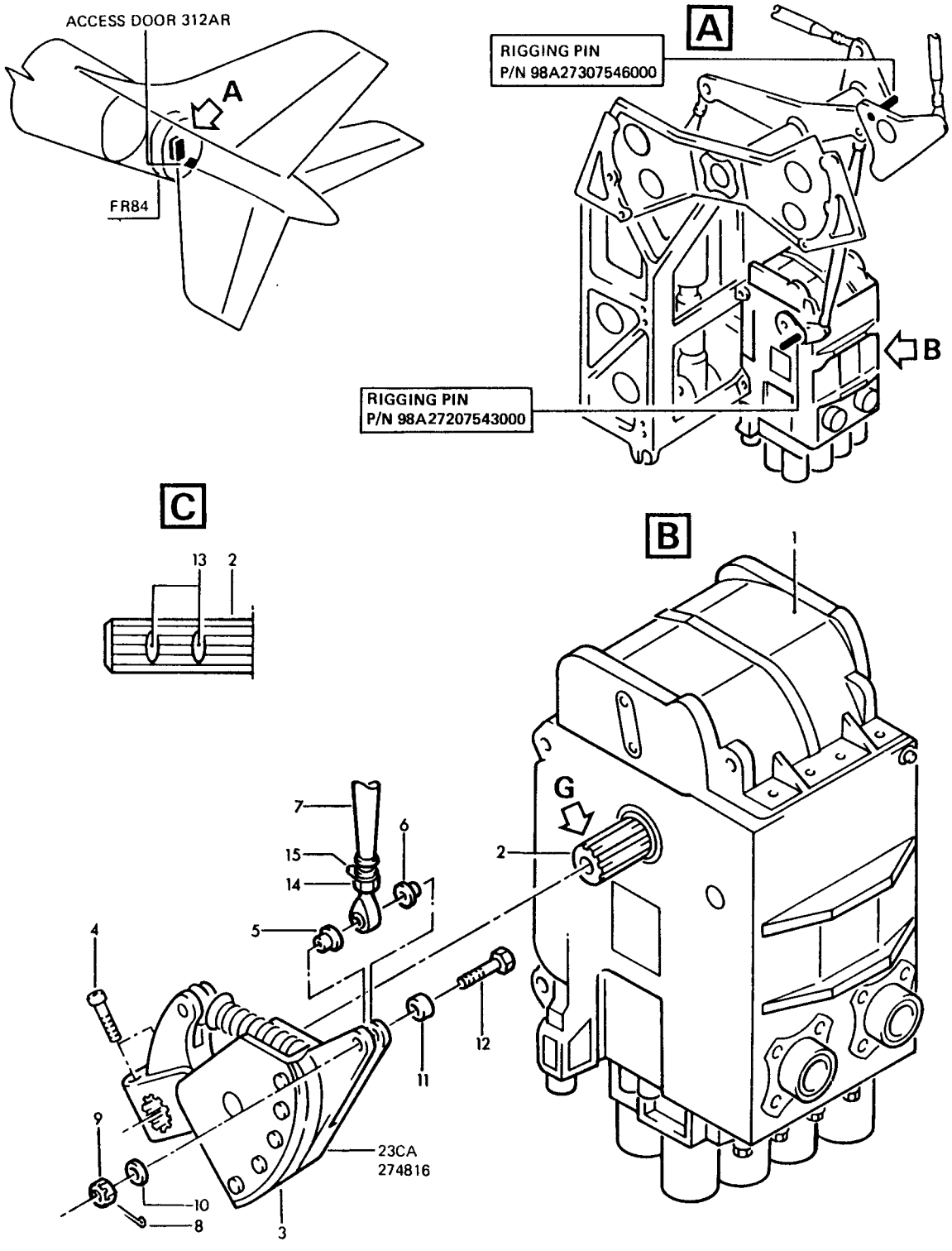
A. Job Set-Up
(1)Position access platform.
(2)Make certain that rudder trim is set at zero.
(3)Make certain that rudder pedals are set at neutral and install rigging pins P/N 98A27901006000 (Qty. 2) in rudder control bellcranks on control mechanism mounting beam in avionics compartment.
(4)Display warning notices in flight compartment prohibiting operation of flight controls.
(5)Open, safety and tag the following circuit breakers:

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/26VAC	305CA1	102/H 2
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/AP1-FD1/FCU	312CA1	102/H 4
21VU	AFS/AP2-FD2/FCU	312CA2	102/H 5
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/AP2-FD2/FCC2/26VAC	305CA2	102/H 7
21VU	AFS/MTP	301CU	102/H 1

(6)Display warning notices, prohibiting repressurization of hydraulic systems.
(7)Open access door 312AR.
(8)Install rigging pin P/N 98A27307546000 in rudder control main bellcrank.

B. Removal (Ref. Fig. 401)
(1)Disconnect pushrod (7) as follows:
(a)Remove cotter pin (8), nut (9), washer (10) spacer (11) and bolt (12).
(b)Disconnect pushrod (7) and secure to structure.
(c)Secure bushes (5, 6) temporarily.
(2)Remove yaw torque limiter lever (3) as follows:
(a)Cut and remove lockwire and remove screws (4).
(b)Remove yaw torque limiter lever (3) from shaft (2).

D. Installation (Ref. Fig. 401)
WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED. BEFORE APPLYING OR RELIEVING HYDRAULIC SYSTEM PRESSURE, MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR. BEFORE PRESSURIZING HYDRAULIC SYSTEMS, CHECK THAT ALL CONTROLS ARE SET TO CORRESPOND WITH THE ACTUAL POSITION OF THE SERVICES THEY OPERATE.
(1)Installation of yaw torque limiter lever.
(a)Make certain the screws (4) are removed from yaw torque limiter lever (3).
(b)Slide yaw torque limiter lever (3) onto shaft (2) so that grooves (13) for screws (4) align.
(c)Install screws (4).
(d)TORQUE screws (4) to between 0.39 and 0.48 m.daN (2.87 and 3.54 lbf.ft.) and safety with lockwire.
(e)Energize the aircraft electrical network (Ref. 24-41-00. P. Block 301).
(f)Make certain that electronics racks ventilation is correct.
(g)Apply hydraulic power to the aircraft systems (Ref. 29-10-00 P. Block 301).
(2)Install rigging pin P/N 98A27207543000 in autopilot yaw actuator (1).
(3)Installation of pushrod.
(a)Make certain that bushes (5, 6) are positioned and install spacer (11).
(b)Connect pushrod (7) to yaw torque limiter lever (3) with bolt (12).
(c)If bolt (12) cannot be easily inserted, adjust pushrod (7) as follows:
1 Cut and remove lockwires and loosen nuts (14).



Yaw Torque Limiter Lever
Figure 401

R

2 Adjust length of pushrod (7) until bolt (12) can be easily inserted.
3 Tighten nuts (14) and safety with lockwire to lockwasher (15).

(d) Install washer (10) and nut (9).
(e) TORQUE nut (9) to between 0.21 and 0.23 m.daN (18.6 and 20.4 lbf.in.) and safety with cotter pin (8).

(4) Remove rigging pins P/N 98A27901006000 (Qty. 2), P/N 98A27307546000 and P/N 98A27207543000.
(5) Remove safety clips and tags, and close circuit breakers 305CA1, 306CA1, 312CA1, 312CA2, 306CA2, 305CA2 and 301CU.
(6) Make certain that rudder control travel ranges are clear.

E. Test

(1) Carry out AP yaw mechanical override test (Ref. 22-10-00, P. Block 501).

F. Close-Up

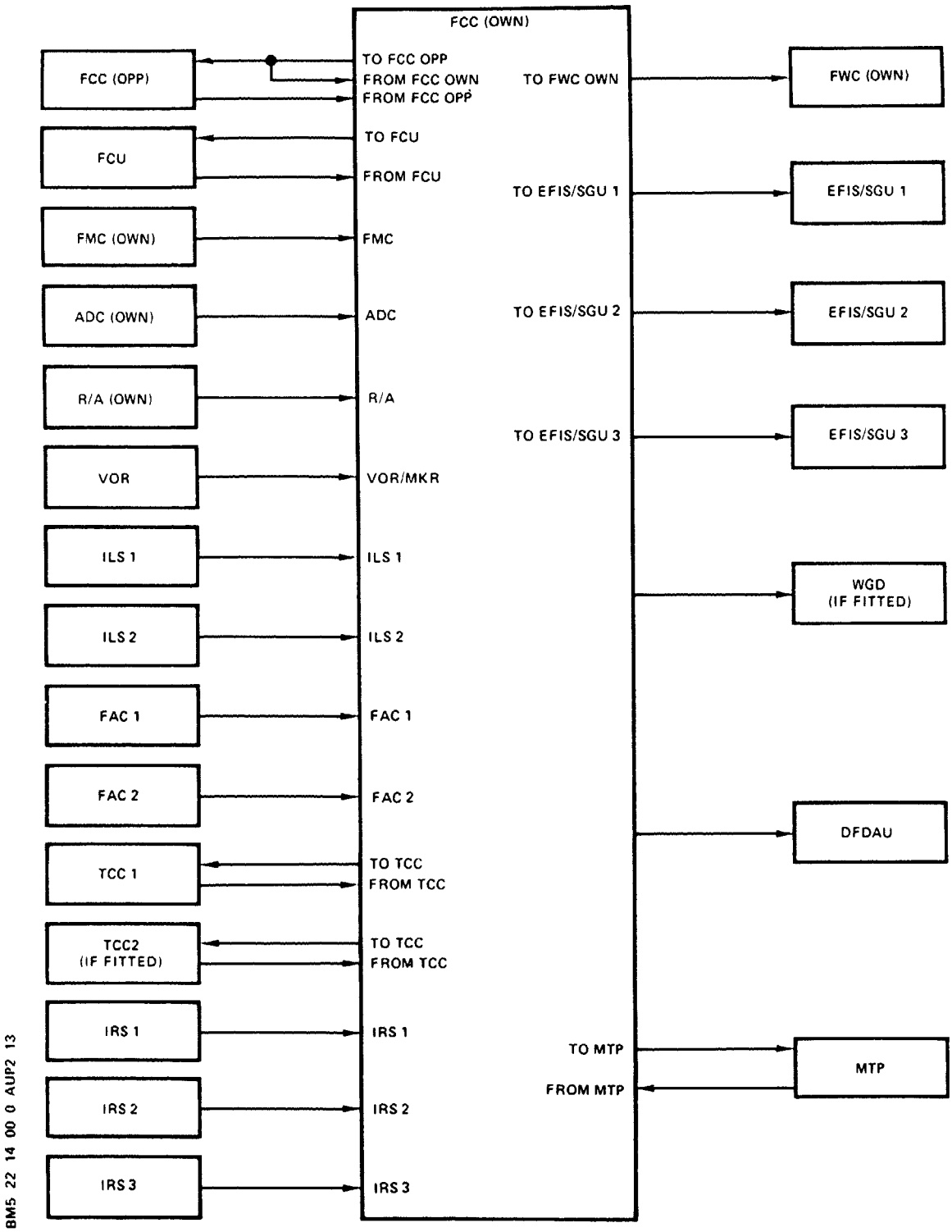
(1) Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2) On autopilot yaw actuator and yaw torque limiter lever, check coating of moisture-removing oil (Mat. No. 05-027) for good condition and renew as necessary (Ref. 51-75-10, P. Block 801).
(3) Close access door 312AR.
(4) Remove warning notices.
(5) Remove access platform.

INTERFACE WITH PERIPHERAL SYSTEMS - DESCRIPTION AND OPERATION

1. Introduction

In operation each FCC receives signals of the following type :
- digital
- analog
- discrete

from interface with various peripherals, sensors and other AFS computers.
In addition, it transmits signals of the same type to the systems with which it is associated :
The digital information is transmitted via ARINC 429 bus lines.
The FCC is provided with one ARINC 600, SIZE 3 rear connector and two front test connectors.
(Ref. Fig. 001)



FCC1(2) - Bus Interconnection Schematic
Figure 001

2. Signals Exchanged with Peripheral Systems

A. ADS (Air Data System)

The following table gives the characteristics of the ADC ARINC 706 signals used by the FCC.

These signals are used for computing cruise guidance modes (speed/Mach, altitude, vertical speed, altitude acquisition).

The Mach signal is used for adjustment of FCC flight control law gains thoughout the flight envelope.

Each FCC is connected to the ADC of the same side.

ADC data bus is connected to command and monitoring channels, thus enabling monitoring of ARINC reception for the parameters used by the control and monitoring channels (case of the Mach data used for the variable gain function of the Mach flight control laws).

The angle-of-attack data available on the ADC bus is not used. The FAC transmits this information to the various AFS systems after monitoring and validation of the information.

ARINC 429 WORD CHARACTERISTICS - ADS INPUTS

PARAMETER NAME	LABEL octal	UNITS	RANGE	RESOLUTION	ACCURACY (minimum significant bits)	TRANSMIT. INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
MACH number	205	mMach (milli MACH)	4096	0.0625	16	125		X		X	Gain adaptation and Mach hold
Computed airspeed	206	Knots	1024	0.0625	14	125		X		X	Speed hold
Baro corrected Alt 1	204	Feet	131072	1.0	17	62.5		X		X	Altitude capture and hold V/S hold
Altitude STD	203	Feet	131072	1.0	17	62.5		X		X	Altitude hold Altitude acquisition V/S hold

B. IRS (Inertial Reference System)

The following table gives the characteristics of the signals from the associated IRS and used by the FCC.

These signals (attitudes, attitude variation speed, acceleration etc.) are used in the various cruise and land modes.

Each FCC is connected to the three IRS's. The associated IRS (OWN) signals are directed on control and monitoring channels ; the opposite IRS (OPPOSITE) and the 3rd IRS signals are used on the monitoring channel where a comparison is performed. Detail of this comparison is described in 22-18.

Signals from opposite and No.3 IRS, used in the FCC are given in the second table.

ARINC 429 WORD CHARACTERISTICS - IRS OWN PARAMETERS USED IN FCC

	PARAMETER NAME	LABEL octal	UNITS	RANGE	RESOLUTION	ACCURACY (minimum significant bits)	TRANSMIT. INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
	PITCH ANGLE	324	°/180	± 90	0.01	14	20	X			X	Pitch basic loop CWS
	ROLL ANGLE	325	°/180	± 180	0.01	14	20	X			X	Roll basic loop CWS
	INERTIAL PITCH RATE	336	°/s	128	0.015	13	20	X			X	Pitch basic loop
	INERTIAL ROLL RATE	337	°/s	128	0.015	13	20	X			X	Roll basic loop
	BODY YAW RATE	330	°/s	128	0.015	13	20	X			X	Yaw basic loop TO - ROLL OUT
	BODY LONG. ACCELERATION	331	g	4	0.001	12	20	X			X	LVL/CH TO/GA laws
	BODY LAT. ACCELERATION	332	g	4	0.001	12	20	X			X	LOC - GA laws
	VERTICAL ACCELERATION	364	g	4	0.001	12	20	X			X	LAND, TO/GA, CWS laws
R	MAG HEADING	320	°/180	± 180°	0.05	12	40	X			X	Heading hold LAND TO mode
R	BARO/ INERTIAL V/S	365	ft/mn	16384	16	10	40	X			X	V/S - ALT - ALT ACQUIRE TO/GA laws
	GROUND SPEED	312	Kts	4096	0.125	15	50	X			X	LOC, LAND laws
R R R	MAGNETIC TRACK ANGLE	317	°/180°	±180°	0.005	15	50	X			X	LOC,LAND laws

	PARAMETER NAME	LABEL octal	UNITS	RANGE	RESOLUTION	ACCURACY (minimum significant bits)	TRANSMIT. INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
R	INERTIAL ALTITUDE	361	ft	131072		18	62.5				X	IRS comparison

ARINC 429 WORDS CHARACTERISTICS - OPPOSITE AND No.3 PARAMETERS USED IN FCC

Origin	Parameters used	Label (Octal)	Utilization in equipment	Comments
IRS OPP	PITCH ANGLE	324	IRS Comparison	- 1 Monitor Receiver
	ROLL ANGLE	325	IRS Comparison	
	VERTICAL ACCELER.	364	IRS Comparison	
	MAG HEADING	320	IRS Comparison	
	INERTIAL PITCH RATE	336	IRS Comparison	
	INERTIAL ROLL RATE	337	IRS Comparison	
	BODY YAW RATE	330	IRS Comparison	
	GROUND SPEED	312	IRS Comparison	
	MAGNETIC TRACK ANGLE	317	IRS Comparison	
IRS 3	PITCH ANGLE	324	IRS Comparison	- 1 Monitor Receiver
	ROLL ANGLE	325	IRS Comparison	
	VERTICAL ACCELER.	364	IRS Comparison	
	MAG HEADING	320	IRS Comparison	
	INERTIAL PITCH RATE	336	IRS Comparison	
	INERTIAL ROLL RATE	337	IRS Comparison	
	BODY YAW RATE	330	IRS Comparison	
	GROUND SPEED	312	IRS Comparison	
	MAGNETIC TRACK ANGLE	317	IRS Comparison	

C. Radio Navigation Equipment

(1)Radio - Altimeters (R/A) (ARINC 707)

The table gives the characteristics of these signals.

ARINC 429 WORD CHARACTERISTICS - RADIO ALTIMETER INPUTS

PARAMETER NAME	LABEL octal	UNITS	RANGE	RESOLUTION	ACCURACY (minimum significant bits)	TRANSMIT. INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
RADIO-HEIGHT	164	Feet	8 192	0.125	16	50	X			X	LAND mode laws

(2)ILS receivers (ARINC 710)

The table gives the characteristics of the digital LOC, GLIDE and heading of the runway selected signals.
(The heading of runway information selected at 1/10 of degree required for the inertial LOC filtering function is delivered by the FAC).
Each FCC receives signals from both ILS for the landing capability computation and the detection of the transmitter failure.

ARINC 429 WORD CHARACTERISTICS - ILS

PARAMETER NAME	LABEL octal	UNITS	RANGE	RESOLUTION	ACCURACY (minimum significant bits)	TRANSMIT. INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
LOCALIZER DEVIATION	173	DDM	± 0.4 387μ A	0.0001	12	50	X			X	Landing laws
GLIDE SLOPE DEVIATION	174	DDM	± 0.8 686μ A	0.0002	12	50	X			X	Landing laws
Selected Runway HDG	105	°/180	± 180	0.1	11	50	X			X	Landing and TO laws

D. Signals Exchanged with the FMC

Two types of signals are exchanged between the FMC and the AFS :

- The FMC transmits guidance orders in NAV and PROFILE modes
- The FMC receives from the AFS, signals from the FCU (mode selection data), FAC (configuration data), TCC (limit thrust data).

The table gives the characteristics of signals transmitted by the FMC to the FCC.

Signals transmitted by the FCU to the FMC are given in 22-15.

ARINC WORD CHARACTERISTICS - FMC INPUTS USED BY FCC

	PARAMETER NAME	LABEL octal	UNITS	RANGE	RESOLUTION	ACCURACY (minimum significant bits)	TRANSMIT. INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
	HORIZONTAL STEERING	121	Degree	±180	0.01	14	100				X	NAV Mode
	VERTICAL STEERING	122	Degree	± 180	0.05	12	100				X	PROFILE Mode
R R	FMC DISCRETES	270	-	-								MODE LOGIC
R												

3. Signals Exchanged with Other AFS Sub-Systems

A. Analog Signals

The FCCS receives analog input signals from the synchro detectors detecting :

- either actuator feedback signals
- or loads applied to the dynamometric rods

and generates the following analog signals :

- actuator commands (control surface deflection commands)
- autotrim commands.

(1) Control surface deflection and feedback commands

Tables, in following pages give the characteristics of actuator control signals and control surface feedback signals.

(Ref. Fig. 002)

(Ref. Fig. 003)

(2) Autotrim command

For safety reasons, the autotrim command is analog.

It is generated directly from the pitch actuator deflection order.

(Analog signal taken after the voter).

The command voter signal is directed through buffer amplifiers to FAC1 and FAC2 command channels.

The monitoring voter signal is directed to FAC1 and FAC2 monitoring channels.

Auto Trim Signal Characteristics

SIGNAL	TYPE	GRADIENT	RANGE	PARAMETER POSITIVE SENSE	POLARITY/PHASE SIGNAL FOR POSITIVE FUNCTION	COMMENTS
TRIM ERROR (AUTO-TRIM	DC	500 mV/° elevator	±20° ele-vator	Request for nose-up speed increase	Positive Voltage across PIN 13J/ in relation to 13K	4 differen-tial outputs

BM5 22 14 00 0 BDM2 13 08
DM5

CONTROL SURFACE POSITION SIGNAL CHARACTERISTICS

SIGNAL	TYPE	GRADIENT	RANGE	PARAMETER POSITIVE SENSE	REFERENCE	POLARITY/PHASE SIGNAL FOR POSITIVE FUNCTION	PHASE SHIFT	FUNCTION	COMMENTS
PITCH SERVO FU1	AC	412 mV/° AP ACTR	± 20.5° AP ACTR	CONTROL SURFACE POSITION GIVING A NOSE-UP ROTATION SPEED	PITCH RATE NULL	IN PHASE SIGNAL ACROSS PIN AB/10H IN RELATION TO AB/10J	11 ± 2°	PITCH POWER LOOP	1° AP ACTR = 1° ELEVATOR
PITCH SERVO FU2	AC	412 mV/° AP ACTR	± 20.5° AP ACTR			IN PHASE SIGNAL ACROSS PIN AE/10H IN RELATION TO AE/10J	11 ± 2°		
PITCH SERVO DIFF FU	AC	0.824 V/° AP ACTR	± 13° AP ACTR	COMMAND REQUESTING A NOSE-UP ROTATION SPEED LOWER THAN CONTROL SURFACE POSITION	CONTROL POSITION EQUAL TO CONTROL SURFACE POSITION	IN PHASE SIGNAL ACROSS PIN AE/12H IN RELATION TO AE/12J	11 ± 2°		
ROLL SERVO FU1	AC	412 mV/° AP ACTR	± 20.5° AP ACTR	CONTROL SURFACE POSITION GIVING A ROLL RATE, RH WING DOWN	ROLL RATE NULL	IN PHASE SIGNAL ACROSS PIN AB/10A IN RELATION TO AB/10B	11 ± 2°	ROLL POWER LOOP	30° AP ACTR = 20° AILERONS
ROLL SERVO FU2	AC	412 mV/° AP ACTR	± 20.5° AP ACTR			IN PHASE SIGNAL ACROSS PIN AE/10A IN RELATION TO AE/10B	11 ± 2°		
ROLL SERVO DIFF FU	AC	824 mV/° AP ACTR	± 22.5° AP ACTR	COMMAND REQUESTING A ROLL RATE, RH WING DOWN, LOWER THAN CONTROL SURFACE POSITION	CONTROL POSITION EQUAL TO CONTROL SURFACE POSITION	IN PHASE SIGNAL ACROSS PIN AE/12A IN RELATION TO AE/12B	11 ± 2°		
YAW SERVO FU1	AC	412 mV/° AP ACTR	± 20.5° AP ACTR	CONTROL SURFACE TO RIGHT WHEN LOOKING FORWARD	YAW RATE NULL	IN PHASE SIGNAL ACROSS PIN AB/10E IN RELATION TO AB/10F	11 ± 2°	YAW POWER LOOP	18.5° AP ACTR = 20° DIRECT
YAW SERVO FU2	AC	412 mV/° AP ACTR	± 20.5° AP ACTR			IN PHASE SIGNAL ACROSS PIN AE/10E IN RELATION TO AE/10F	11 ± 2°		
YAW SERVO DIFF FU	AC	824 mV/° AP ACTR	± 22.5° AP ACTR	COMMAND REQUESTING A POSITION LESS TO THE RIGHT THAN CONTROL SURFACE POSITION	CONTROL POSITION EQUAL TO CONTROL SURFACE POSITION	IN PHASE SIGNAL ACROSS PIN AE/12E IN RELATION TO AE/12F	phi = 11 ± 2°	YAW POWER LOOP	

Control Surface Position Signal Characteristics
Figure 002

SIGNAL	TYPE	GRADIENT	RANGE	PARAMETER POSITIVE SLOPE	REFERENCE	POLARITY/PHASE SIGNAL FOR POSITIVE FUNCTION	COMMENS
PITCH SERVO DIV	DC	0.20 mA/°/s CONTROL SURFACE		REQUEST FOR NOSE UP PITCH RATE INCREASE		POSITIVE VOLTAGE ACROSS PIN AB/12G IN RELATION TO AB/12H	2000Ω DIFFERENTIAL OUTPUT
ROLL SERVO DRIVE	DC	0.10 mA/°/s CONTROL HANDWHEEL		REQUEST FOR ROLL RATE INCREASE RH WING DOWN		POSITIVE VOLTAGE ACROSS PIN AB/12A IN RELATION TO AB/12B	2000Ω DIFFERENTIAL OUTPUT
YAW SERVO DRIVE	DC	0.20 mA/°/s LS AILERON		REQUEST FOR RH YAW RATE INCREASE		POSITIVE VOLTAGE ACROSS PIN AB/12D IN RELATION TO AB/12E	2000Ω DIFFERENTIAL OUTPUT

BM5 22 14 00 0 BDN2 13 08
DM5

Actuator Control Signal Characteristics
Figure 003

(3)Dynamometric rod signals
Each pitch and roll dynamometric rod on Captain's and First Officer's side include two synchros ; one associated to the FCC1, the other to the FCC2.
In this way each FCC receives load signal information from each of the four rods.

Dynamometric Rod Signal Characteristics

SIGNAL	TYPE	GRADIENT	RANGE	PARAMETER POSITIVE SENSE	REFERENCE	POLARITY/ PHASE SIGNAL FOR POSITIVE FUNCTION	PHASE SHIFT	FUNCTION
PITCH CWS, CAPTAIN	AC	408mV/kg	7.65kg	Nose-up request	Null load	In-phase signal across pin AB/11F in relation to AB/11G	phi = 14.5 ±3°	Pitch control wheel steering
PITCH CWS, FIRST OFFICER	AC	408mV/kg	7.65kg			In-phase signal across pin AA/11F in relation to AB/11G	phi = 14.5 ±3°	
ROLL CWS, CAPTAIN	AC	408mV/kg	7.65kg	Roll speed request, RH wing down	Null load	In-phase signal across pin AB/11C in relation to AB/11D	phi = 14.5 ±3°	Roll control wheel steering
ROLL SCWS, FIRST OFFi-CER	AC	408mV/kg	7.65kg			In-phase signal across pin AA/11C in relatiom to AA/11D	phi = 14.5 ±3°	

B. Discrete Signals

The acquisition and transmission principle of discrete signals is described in 22-00, general.

The following tables give the list of the discrete signals received and transmitted by the FCC.

Table of FCC Discrete Input Signal Characteristics

SIGNAL	LEVEL		FUNCTION	COMMENTS
	ACTIVE	NOT ACTIVE		
YAW DAMPER ENG OWN	+28 V	0 C	AP engagement, capabilities	2 signals
TRIM ENG OWN	+28 V	0 C	AP engagement, capabilities	2 signals
YAW CLUTCH SW	+28 V	0 C	AP engagement	2 switches
ROLL CLUTCH SW	+28 V	0 C	AP engagement	2 switches
PITCH CLUTCH SW	+28 V	0 C	AP engagement	2 switches
INST DISC NC SW	0 C	+28 V	AP engagement	
INST DISC NO SW	+28 V	0 C	AP engagement	
CWS SENS CNCTD	Ground	0 C	AP engagement	
AP SW	+28 V	0 C	AP engagement	2 switches
NOSE GEAR SW	+28 V	0 C	Deviation, TO/GA, test	
ENGINE STOP	+28 V	0 C	Ground supply cutoff, AP engagement	
CAPT F/D BAR CLEAR	Ground	0 C	FD bars cleared, FD engagement	
F/O F/D BAR CLEAR	Ground	0 C		

Table of FCC Discrete Input Signal Characteristics

SIGNAL	LEVEL		FUNCTION	COMMENTS
	ACTIVE	NOT ACTIVE		
TEST POWER ON	+28 V	0 C	Supply circuits	
TO/GA NC SW	0 C	+28 V	TO/GA mode	
TO/GA NO SW	+28 V	0 C	TO/GA mode	
SIDE 1	+28 V	0 C	AP engagement	2 signals
HYDRAULIC CIRCUIT HLTY	Ground	0 C	AP engagement	2 signals
A/P ENG. OPP	+28 V	0 C	AP engagement - capabilities - mode logic	2 signals
F/D ENG OPP	+28 V	0 C	AP engagement - capabilities - mode logic	2 signals
YAW DAMPER ENG OPP	+28 V	0 C	AP engagement - capabilities	2 signals
TRIM ENG OPP	+28 V	0 C	AP engagement - capabilities	2 signals
A/P OWN LEVER ON CWS	+28 V	0 C	AP engagement	
A/P OPP LEVER ON CWS	+28 V	0 C	AP engagement	
A/P OPP LEVER ON CMD	+28 V	0 C	AP engagement, mode logic	2 signals
A/P OWN LEVER ON CMD	+28 V	0 C	AP engagement, mode logic	2 signals
A/C 28 BARS SPLIT	+28 V	0 C	Capabilities	
TCC SELECT OWN	+28 V	0 C	ARINC TCC line selection	
TCC SELECT OPP	+28 V	0 C	ARINC TCC line Selection	
PFD SWITCHING OWN PFD	+28 V	0 C	FD interface PFD - FD interface	Signals: PFD SWOWN, PFD SWOPP

SIGNAL	LEVEL		FUNCTION		COMMENTS	
	ACTIVE	NOT ACTIVE				
SWITCHING OPP			PFD	both on 1	+28 V	0 C
				both on 2	0 C	+28 V
				split	0 C	0 C
FAC P/S DEMAND OPP	+28 V	0 C	Ground supply cut off			
FAC P/S DEMAND OWN	+28 V	0 C	Ground supply cut off			

Table of FCC Discrete Input Signal Characteristics

SIGNAL	LEVEL		FUNCTION	COMMENTS
	ACTIVE	NOT ACTIVE		
RIGHT WHEEL SPEED > 70 Kt	Ground	0 C	Flight Leg Increment	
LEFT WHEEL SPEED > 70 Kt	Ground	0 C		
A/P CLUTCH RETURN	Ground	0 C	AP engagement	
A/Th in SPEED MODE OWN	+28 V	0 C	Capabilities	2 signals
PRESS/STD ALT SEL	Ground	0 C	ALTITUDE select	Ground = STD
FCU HEALTHY	+28 V	0 C	AP engagement	
TCC LAND 3 F/W OWN	0 C	+28 V	Capabilities	
EFIS FD OWN F/W	0 C	+28 V	Capabilities	
FWC OWN ACQ FAIL	0 C	Ground	Capabilities	
A/Th IN SPEED MODE OPP	+28 V	0 C	Capabilities	2 signals
TCC LAND 3 F/W OPP	0 C	+28 V	Capabilities	
EFIS FD OPP F/W	0 C	+28 V	Capabilities	
FWC OPP ACQ FAIL	Ground	0 C	Capabilities	
EFIS ATT F/W OWN	0 C	+28 V	Capabilities	

SIGNAL	LEVEL		FUNCTION	COMMENTS
	ACTIVE	NOT ACTIVE		
R **ON A/C 001-100,				
EFIS ATT F/W OPP	O C	+28 V	Capabilities	
R **ON A/C 101-200,				
EFIS ATT F/W OPP	O C	+28 V	Capabilities	
G/S PIN PROG	Ground	O C	GLIDE capture before LOC capture	

R EFFECTIVITY: 001-200,

PIA

R **ON A/C ALL

Table of FCC Discrete Output Signal Characteristics

SIGNAL	LEVEL		FUNCTION	COMMENTS
	ACTIVE	NOT ACTIVE		
BUS BARS SPLIT	+28 V	O C	28 V Generation Splitting side 1 side 2	
SRS MODE ENGAGED	Ground	O C	FAC	2 signals
ROLL CLUTCH COIL	C: +28 V M:Ground	C : O C M : O C	-	
PITCH CLUTCH COIL	C: +28 V M:Ground	C : O C M : O C	- Control of AP clutch coils	0.4A per clutch
YAW CLUTCH COIL	C: +28 V M:Ground	C : O C M : O C	-	
A/P TOGGLE HOLD	Ground	O C	AP lever control	2 signals
A/P ENG OWN	+28 V	O C	-	4 signals
A/P OWN LEVER ON CWS	+28 V	O C	- FCC sys status to all AFS systems	4 signals
A/P OWN LEVER ON CMD	+28 V	O C	-	4 signals
HYDRAULIC PRESS ON	+28 V	O C	Servo Hydraulic Pressure Control	3 signals : (0.4A per AP Actuator)
R/A TEST INHIBIT	Ground	O C	- Receiver Test - Inhibition	
F/D ENG OWN	+28 V	O C	All AFS systems	4 signals
CWS LEV/DET	+28 V	O C	FAC	4 signals
EFIS ATT F/W OWN	+28 V	O C	Capabilities	
EFIS ATT F/W OPP	+28 V	O C	Capabilities	
ILS TEST INHIBIT	Ground	O C	Receiver Test Inhibition	4 signals

C. ARINC Signals

(1)ARINC data transmitted by the FCC

Most of the data exchanged between the FCC and the other AFS systems are transmitted via the FCC output ARINC 429 data bus.

The acquisition and transmission principle of the ARINC buses is described in 22-00, general.

Figures show the FCC data bus generation and its users. It can be noted that the bus from the command channel is sent to the monitoring channel input (exchange of cruise mode guidance orders, monitoring of FD orders, synchronization, maintenance, etc...).

Exchange from the monitoring channel to the command channel is via an FCC internal data bus.

Tables show the characteristics of the signals on the output data bus and their functions.

Table of ARINC Output Signal (12 KHZ) Characteristics (Command Channel)

OCTAL LABEL	PARAMETER	TRANSMIT INTERVAL (ms)	UNSIGNED SIGNIFICANT BITS	UNIT	RANGE	COMMENTS
274	FCC DISCRETES 1	50	-	-	-	
275	FCC DISCRETES 2	50	-	-	-	
273	FCC DISCRETES 3	50	-	-	-	
146	FCC DISCRETES 4	50	-	-	-	
270	FCC DISCRETES 5	50	-	-	-	
271	FCC DISCRETES 6	50	-	-	-	
155	θ C MODE SUP	150	15	degs	±20°	
157	Phi C MODE SUP	150	15	degs	±45°	
144	ACCURACY INPUT	150	15	degs	±45°	Accuracy Integration
173	ESTIMATION LOC	150	12	micro A	400 micro A	
141	FD BAR PITCH	50	12	degs	±180°	
140	FD BAR ROLL	50	12	degs	±180°	
143	FD BAR YAW	50	12	degs	±180°	
204	ALTITUDE BARO	300	17	Feet	50 000ft.	
320	HEADING	300	12	degs	±180°	
206	COMPUTER AIRSPEED	300	14	Knots	400 Kt	
212	ALTITUDE RATE	300	11	Ft/mn	9000ft/mn	
205	MACH	300	13	mMach	1 Mach.	
103	SELECTED AIRSPEED	300	11	Knots	400 Kt	Computed by FCC if FCU in MACH mode recopy if FCU in SPEED mode
151	LOC BEARING TRUE FMC	300	11	degs	±180°	
350	MAINTENANCE DATA 1	Async	-	-	-	
351	MAINTENANCE DATA 2	Async	-	-	-	
353	MAINTENANCE DATA 3	Async	-	-	-	

ARINC OUTPUT SIGNALS (Continued)

BIT NUMBER	Functions	Coding 0	Coding 1	Comments
1 - 8	Label 274			Coded Information
9	- - DES			ALT ACQ 13 and 29
				ALT 13 and 22
10	- - CLIMB			GS CPT 19 and 29
				GS TRK 19 and 22
11	SPD			
12	PROFILE			
13	ALT			SPD CLB $\overline{12}$ and 10 and 11
				SPD DES $\overline{12}$ and 9 and 11
14	V/S			MACH CLB $\overline{12}$ and 10 and 18
				MACH DES $\overline{12}$ and 9 and 18
15	FLARE			PROFILE ALT 12 and $\overline{9}$ and $\overline{10}$
				PROFILE CLB 12 and 10
16	PITCH GA			PROFILE DES 12 and 9
				PSPD PHASE 11 and 12
17	PITCH TO			PMACH PHASE 18 and 12
18	MACH			
19	GS			
		NOT		
20		REQUESTED	REQUESTED	
21				
22	- - TRK			
23	IRS 3 ATT			
24	FD PITCH BAR FLASH			
25				
26				
27				
28				
29	- - CPT			
30	-|			
	|- MATRIX			
31	-|			
32	Parity			

ARINC OUTPUT SIGNALS (Continued)

BIT NUMBER	Functions	Coding 0	Coding 1	Comments
1 - 8	Label 275			Coded Information
9	Spare			
10	Spare			NAV 11 and 23
				LOC CPT 15 and 21
11	NAV			LOC TRK 15 and 23
				VOR CPT 22 and 21
12	HDG HOLD			VOR TRK 22 and 23
13	HDG SEL			
14				
15	LOC			
16	ROLL OUT			
17	YAW TO			
18	ROLL GA			
19	Spare			
		NOT		
20		REQUESTED	REQUESTED	
21	- - CPT			
22	VOR			
23	- - TRK			
24	FD ROLL BAR FLASH			
25	TO GA LEVER			
26				
27				
28	ALIGN			
29				
30	-| |- MATRIX			
31	-|			
32	Parity			

ARINC OUTPUT SIGNALS (Continued)

BIT NUMBER	Functions	Coding 0	Coding 1	Comments
1 - 8	Label 273			Coded Information
9	Spare			
				PROFILE CLB ARM 15 and 24
10	Spare			PROFILE DES ARM 15 and 25
11				
12	ALT ACQ ARM			
13				
14	NAV ARM			
15	PROFILE ARM			
16	LOC ARM			
17				
18				
19				
		NOT		
20	LAND ARM	REQUESTED	REQUESTED	
21				
22	GLIDE ARM			
23	VOR ARM			
24	- - CLIMB			
25	- - DES			
26	Spare			
27	Spare			
28	Spare			
29	V/S LIMIT			
30	-| |- MATRIX			
31	-|			
32	Parity			

ARINC OUTPUT SIGNALS (Continued)

BIT NUMBER	Functions	Coding		Comments
		0	1	
1 - 8	Label 146			
9	INSTINCTIVE DIS-CONNECT			
10	ALT ALERT INHIBITION			MW
11	A/P CWS ENGD			
12	A/P CMD ENGD			
13	F/D ENGD			
14	LAND TRK			
15				
16	APPR 2			EFIS
17	LAND 2			EFIS
18	LAND 3			EFIS
19	RA F/W			
20		NOT REQUESTED	REQUESTED	
21	BPLDC 3			
22				
23				
24				
26	LAND 3 F/W			EFIS
27	SURVIVAL LOC FAILURE			MW
28	DRIFT LOC BEAM			MW
29				
30	-| |- MATRIX			
31	-|			
32	Parity			

ARINC OUTPUT SIGNALS (Continued)

BIT NUMBER	Functions	Coding		Comments
		0	1	
1 - 8	Label 270			
9	IRS OPP VALIDITY			
10				
11	GS CPT LD COM			
12	LOC CPT COND			
13	LOC TRK LD COM			
14	LOC TRK LD MON			
15				
16				
17	FLARE COND			
18	LD 12			
19	PITCH CWS LD	NOT REQUESTED	REQUESTED	
20	ROLL CWS LD			
21	APPR 2 CAPABILITY			
22	LAND 2 CAPABILITY			
23	LAND 3 CAPABILITY			
24	FCC F/W			
25	IRS OWN VALIDITY			
26	SET ALT ACQ ARM POSSIBLE			
27				
28				
29	GS CPT LD MON			
30	-| |- MATRIX			
31	-|			
32	Parity			

ARINC OUTPUT SIGNALS (Continued)

BIT NUMBER	FUNCTION	CODING		
1 to 8	LABEL 350			
9	TYPE OF INFORMATION	0 : COMPLEMENTARY INFORMATION 1 : EVENT REPORT		
10 to 13	TYPE OF EVENT OR CODE FOR COMPLEMENTARY INFORMATION	EVENT REPORT 0 : AP DISCONNECTION 1 : FD LOSS 2 : LAND3 LOSS 3 : LAND2 LOSS 4 : APP2 LOSS 5 : CAPACITY NOT AT MAXIMUM LEVEL 6 to 13 : SPARE 14 : COMPLEMENTARY TEST RESULT 15 : AFS TEST RESULT	COMPLEMENT INFORMATION CODE 1 : END OF THE CONTEXTS (2) TABLE 2 : CLEAR EXECUTED 3 : CLEAR NOT EXECUTED 4 : END OF CONTEXT (2) 0 : OTHERWISE (1)	
14 15	RESULT OF FAILURE LOCALIZATION ANALYSIS	0 : DISCONNECTION BY LEVER 1 : VOLUNTARY DISCONNECT BY INST DISC 2 : NORMAL DISCONNECTION 3 : ABNORMAL DISCONNECTION	AFS TESTS COMPL. TESTS BIT 14 0 : ACCEPTED 1 : REFUSED	RETURNED INFORMATION BIT : 14 0 : NOT CONNECTED 1 : CONNECTED
16 17	NUMBER OF LRU	IN MAINTENANCE DATA (LABEL 351)		
18	COMPLEMENTARY TESTS ON GROUND REQUESTED	0 : NO 1 : YES		BIT 22 : 0 : NOT RECEIVED 1 : RECEIVED
19 to 29	SPARE			
30 31	STATUS MATRIX			

BIT NUMBER	FUNCTION	CODING
32	PARITY	

(1) : in case of response to a need of information to improve MTP diagnostics these bits are filled with zero.
(2) : to print and decode failure information recorded inside the computers : complementary information code = 1 : end of printing
4 : end of a record.

ARINC OUTPUT SIGNALS (Continued)

BIT NUMBER	Functions	Coding 0	Coding 1	Comments
1 - 8	LABEL 351			
9	-|			
10	|			
11	|			
12	|- LRU 1 CODE			
13	|			
14	|			
15	-|			
16	-|			
17	|			
18	|			
19	|- LRU 2 CODE			
20	|			
21	|			
22	-|			
23	-|			
24	|			
25	|			
26	|- LRU 3 CODE			
27	|			
28	|			
29	-|			
30	-|			
	|- MATRIX			
31	-|			
32	Parity			

LRU CODES

0 : LRU OWN
1 : LRU OPP
Example : ILS OWN : 08H
ILS OPP : 09H

LRU CODE	LRU	LRU CODE	LRU	LRU CODE	LRU
00	NOT USED	36H	EEC SENSORS	60H	L ATS COUPLING UNIT
02	ADS	38H	PMC SENSORS	62H	ENGINE SENSOR
04	IRS	3AH	NOSE GEAR RELAY	64H	HMC
06	IRS3	3CH	R MAIN GEAR RELAY	66H	AP DYN ROD NOT CONNECTED
08	ILS	3EH	L MAIN GEAR RELAY	68H	PITCH DYN ROD CAPT/FO
0AH	Radio Altimeter	40H	GROUND SPOILER CONTROL BOX	6AH	AT DYN ROD NOT CONNECTED
0CH	VOR	42H		6CH	AT COUPLING UNIT NOT CONNECTED
0EH	EFIS	44H	AP DISCONNECT PB SW	6EH	115VAC POWER SPLY
10H	ALPHA Probe	46H	AT INST DISCONNECT SW	70H	AP PITCH ACTUATOR
12H	STBY ALPHA Probe	48H	AT ACTUATOR NOT CONNECTED	72H	AP ROLL ACTUATOR
14H		4AH		74H	AP YAW ACTUATOR
16H	FMC	4CH	R THROTTLE POSITION DETECTOR	76H	AT ACTUATOR
18H	EEC	4EH	L THROTTLE POSITION DETECTOR	78H	PITCH TRIM ACTUATOR
1AH	EFCU	50H	SPLIT BARS SWITCH	7AH	YAW DAMPER ACTUATOR
1CH	FWC	52H		7CH	28VDC POWER SPLY
1EH	FCC	54H		7EH	26VAC POWER SPLY
20H	TCC	56H	PITCH TRIM CONTROL SWITCH		
22H	FAC	58H	TO/GA LEVERS		
24H	FCU	5AH			
26H	TRP	5CH	HYDRAULIC SYSTEM		
28H	PMC	5EH	R ATS COUPLING UNIT		
2AH	MTP				
2CH	FAC/ATS Engage Unit				
2EH	VARIABLE LEVER				
30H	FD BAR REMOVAL SWITCH				
32H	SFCC				
34H	28 V GEAR				

(2)ARINC data received by the FCC

(a)FAC data bus

Each FCC receives the FAC No.1 and No.2 output data bus.

The following table gives data used by the FCC. The characteristics of signals conveyed on the FAC data bus are given in 22-24.

ORIGIN	PARAMETERS USED	OCTAL LABEL	FUNCTION IN COMPONENT	COMMENTS
FAC OWN OR OPP	MAXIMUM SPEED	207	LEVEL CHANGE TO/GA LAWS	1 Command receiver 1 Monitor receiver
	AFS DISCRETES	274	ENGAGE LOGIC	
	ANGLE OF ATTACK AVERAGE	241	TO/GA LAWS	
	MINIMUM AIR SPEED	245	TO/GA, LVL/CH LAWS	

(b)MTP

Each FCC receives the output data bus from the MTP.
The characteristics of signals conveyed on the MTP data bus are given in 22-40.

PARAMETERS USED	OCTAL LABEL	FUNCTION IN COMPONENT	COMMENTS
MAINTENANCE DATA	350	TEST	1 Command receiver

(c)FCU

Each FCC receives the FCU output data bus.
The following table gives data used by the FCC.
The characteristics of signals conveyed on the FCU data bus are given in 22-15.

ORIGIN	PARAMETERS USED	OCTAL LABEL	FUNCTION IN FCC	COMMENTS
FCU, OWN (ARINC 701)	SELECTED HDG	101	HDG MODE	
	SELECTED ALT	102	ALT MODE	
	SELECTED AIR SPEED	103	SPEED MODE	1 Command Receiver
	SELECTED VERT. SPEED	104	V/SPEED MODE	1 Monitor Receiver
	SELECTED MACH	106	MACH MODE	
	AFS DISCRETES 1	270	MODE LOGICS	
	AFS DISCRETES 2	271	MODE LOGICS	

(d)FCC (associated) - FCC (opposite)

The FCC output data bus (control channel) is sent :
- to the opposite FCC for problems of mode synchronization or mode phases.
- to the monitoring channel of the associated FCC for :
  . transmission of cruise mode guidance orders
  . mode synchronization.

Tables, in following page, give the characteristics of the signals used by the opposite FCC and the monitoring channel of the associated FCC.

ORIGIN	PARAMETERS USED	OCTAL LABEL	FUNCTION IN COMPONENT	COMMENTS
FCC OPP (ARINC 701)	FCC DISCRETES 6	271		
	FCC DISCRETES 1	274	DIALOG FCC1 - FCC2	
	FCC DISCRETES 2	275	SYNCHRONIZATION OF MODES OR MODE PHASES	1 Monitor Receiver 1 Command Receiver
	FCC DISCRETES 4	146		
	FCC DISCRETES 5	270		
	FCC DISCRETES 3	273		
FCC MONITORING	FCC DISCRETES 6	271	Mode Synchronization	
	FCC DISCRETES 3	273		
	θ C MODES SUP	155	Basic Loops	
	Phi C MODES SUP	157		
	ACCURACY INPUT	144		
	ESTIMATION LOC	173	Inertial LOC	
	F/D BAR PITCH	141	FD bar command monitoring	1 Monitor Receiver (Feedback of ARINC command line)
	F/D BAR YAW	143		
	FCC DISCRETES 1	274	Mode Synchronization	
	FCC DISCRETES 2	275		
	FCC DISCRETES 4	146		
	FCC DISCRETES 5	270		
	MAINTENANCE DATA 3	353	Processor Synchronization	

(e)TCC

Each FCC receives TCC 1 and 2 (if fitted) output data bus.
The following table gives data used by the FCC.
The characteristics of signals conveyed on the TCC data bus are given in 22-34.

ORIGIN	PARAMETERS USED	OCTAL LABEL	FUNCTION IN COMPONENT	COMMENTS
TCC OWN or OPP	EPR ACTUAL	340	Engine Failure Compensation	
	N1 ACTUAL	346		
	DISCRETE WORD 5	275		

NOTE : The ENG1 and ENG2 actual thrust signals are also transmitted under labels 340 or 346 with the following encoding :

BIT 10	BIT 9	ENGINE
0	1	Left
1	0	Right

FLIGHT CONTROL UNIT - DISPLAYS, WARNINGS, LIGHTING
DESCRIPTION AND OPERATION

1. General

This section describes the controls, displays and warnings associated with the FCCS.

A. Most of the controls are grouped on the Flight Control Unit (FCU) and also in various locations in the flight compartment for the following controls :
- AP disconnect pushbutton switches
- Take-off/go around levers
- FD/FPV selector switches
- FD data switching pushbutton switches

B. This displays (mode annunciations and approach capabilities, etc...) are grouped on the Flight Mode Annunciators (FMA) on the upper section of the PFDs located on Captain's and First Officer's instrument panels.

C. The warnings associated with the FCC are described in 22.16 for the AP OFF warning and in 22.17 for the other warnings.

2. Flight Control Unit (FCU)

A. Description

The FCU features :
- AP1 and AP2 engage levers
- AP/FD mode selection pushbutton switches
- Autothrottle engagement pushbutton switches
- Reference speed, altitude, heading, vertical speed displays.

The FCU is located on glareshield 13VU.
Permanent consumption is 50 W from the 28VDC network.

(1)Face

The face features :
- AP1 and AP2 two-position (OFF-ON) engage levers
- CMD/CWS selection pushbutton switch
- Selector knobs and display windows for following reference values :
  - Speed/Mach (used by the AP/FD or the autothrottle)
  - Altitude
  - Heading
  - Vertical Speed

  (Altitude, Heading, Vertical Speed: used by the AP/FD)
- Mode selection pushbutton switches.

(a)AP engage levers

These two levers are of the two-position (ON-OFF) electro-mechanically latched type.
Self-latching of a lever is ensured by a coil supplied by the FCC control and monitoring logics self-latching cut off is such that lever returns to OFF positions and the pilot can always override the AP lever self-latching load.

(b)CWS/CMD selection pushbutton switch

This pushbutton switch is of the same type as the mode selection pushbutton switches but it is equipped with a special cap and with four 28V bulbs instead of two.
In flight AP engagement is initially always performed in CMD mode.
Action on CWS/CMD pushbutton switch results in AP engagement in CWS mode. Return to CMD mode is obtained by a further action on pushbutton switch. Engagement in CWS or CMD mode causes illumination of pushbutton switch corresponding legend.
AP engagement on ground is always performed initially in CWS.
Change from CWS to CMD is authorized on the ground with engines shut down.

(c)AP/FD/ATS mode selection pushbutton switches
- ATS ; A/THR pushbutton switch ; enables ATS to be engaged
- AP/FD : These mode pushbutton switches enable the following modes to be engaged
  - ALT HLD (ALTITUDE HOLD) pushbutton switch :
    Selection of Altitude Hold mode for the AP/FD
  - LVL/CH (LEVEL CHANGE) pushbutton switch
    Selection of the AP/FD Speed or Mach hold mode and engagement of the thrust mode for the autothrottle (climb) or automatic autothrottle retraction (descent)
  - PROF (PROFILE) pushbutton switch
    Selection of Profile mode (coupling of the AP/FD pitch axis, and

the ATS for the FMC)

- HDG SEL (HEADING SELECT) pushbutton switch
  Selection of AP/FD HDG SEL mode acquisition and hold of the displayed heading.
- NAV (NAVIGATION) pushbutton switch
  Selection of AP/FD NAV mode (coupling of the AP/FD, roll axis, for the FMC)
- V/L (VOR/LOC) pushbutton switch (combined mode) VOR or LOC mode selection is made by means of the VOR/NAV/ILS selector switch
- LAND pushbutton switch
  Selection of AP/FD LAND (landing) mode

In addition, a pushbutton switch located to the left of the SPD/MACH display enables switching from SPD to MACH function (at display and engaged mode level)

- PRESET light located to the right of SPD/MACH display informs the crew that a speed or mach preselection has been carried out.

The mode selection pushbutton switches are 3/4 in. pushbutton switches with a mechanical return to the rest position. These pushbutton switches are equipped with two 28V bulbs, which illuminate 3 Green horizontal bars in the switch thus indicating that mode selection is taken into account by the FCCs.

(d)Reference displays

SPD/MACH, ALT, HDG, V/S displays are of the LCD type (liquid Cristal Display). These display windows are integral with modules incorporating the display with its electronic control components. These modules can be removed from the FCU front face.

(e)Selection of reference values

Selection of reference values is performed by means of selector knobs located under display windows. Each selector knob is equipped with optical incremental encoders. The selector knobs also include a push-pull position. A spring returns the selector knob to the intermediate position (spring loaded).

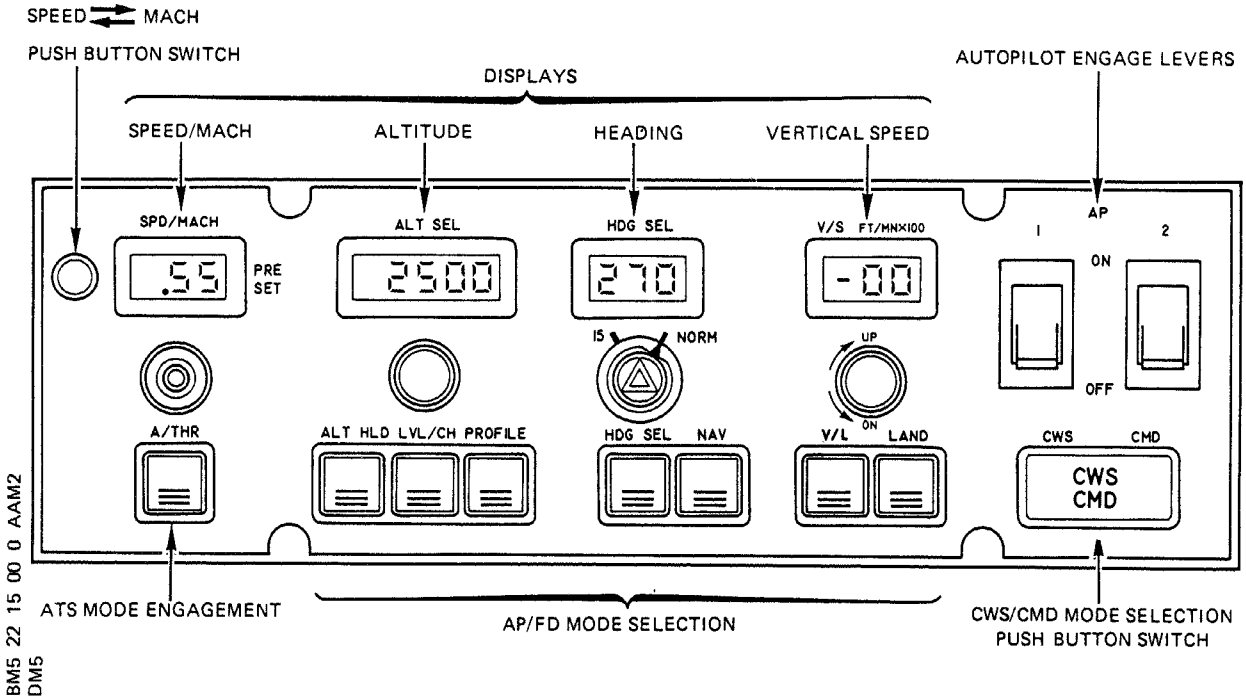
Push, pull or intermediate position is decoded by optical switches. Each selector knob is equipped with a mechanical notching corresponding to 32 clicks per turn. The knobs are shaped differently to avoid any confusion by the crew : their markings recall the indexes they drive on the instruments.

The knobs have different diameters to achieve correspondence between reference change and pilot's thumb stroke. The HDG SEL control incorporates a second outer crown enabling selection of maximum bank (15° or normal) in HDG SEL mode.

(f)Night reading

The various functions : pushbutton switches, selector knobs, display windows are equipped with integral lighting, for night reading of various markings and values, adjustable from a control located on the glareshield under the FCU.

(Ref. Fig. 001)



Flight Control Unit - Face
Figure 001

(2)FCU inputs-outputs
The following tables give the nature of the FCU inputs/outputs.

ARINC Inputs

ORIGIN	PARAMETERS USED	OCTAL LABEL	FUNCTION IN COMPONENT	COMMENTS
	ALTITUDE	204	SYNCHRONIZATION	Initiated at power rise
	MACH	205	SYNCHRONIZATION	
	COMPUTED AIRSPEED	206	SYNCHRONIZATION	
	ALTITUDE RATE	212	SYNCHRONIZATION	
	MAGNETIC HDG	320	SYNCHRONIZATION	
FCC	SELECTED AIRSPEED	103	Used in Mach mode	SPD value computed from the Sel Mach by the FCC
	FCC DISCRETE WORD 1	274	INFORMATION MODE	
	FCC DISCRETE WORD 2	275	INFORMATION MODE	
	FCC DISCRETE WORD 3	273	INFORMATION MODE	
	FCC DISCRETE WORD 4	146	INFORMATION MODE	
TCC	TCC DISCRETE WORD 3	274	INFORMATION MODE	

ANALOG INPUTS

SIGNAL	NATURE	GRADIENT	RANGE	POSITIVE SENSOR	FUNCTION	COMMENTS
5V PANEL DIM CONNECTOR J1	A.C.		0V.5V.RMS		Night lighting (Markings and Displays)	

INPUT DISCRETE SIGNALS

	SIGNAL	LEVEL		FUNCTION	COMMENTS
		ACTIVE	NOT ACTIVE		
	CONNECTOR J1				
	AP1 CLUTCH SUP 28 V	+28 V	O.C.	AP engagement,	Segregation 1
	AP1 TOGGLE HOLD C.	GND	O.C.	AP engagement	
	AP1 ON	+28 V	O.C.	AP engagement	
	CMD1 ENGD	+28 V	O.C.	FCC line selection	
	CMD2 ENGD	+28 V	O.C.		
	FD1 ENGD	+28 V	O.C.		
	FD2 ENGD	+28 V	O.C.		
	CWS1 ENGD	+28 V	O.C.		
	CWS2 ENGD	+28 V	O.C.		
	LAMP TEST	+28 V	O.C.	Test of lamps and displays	
	DIM COMMAND 1	GND	O.C.	Pushbutton Dimming	
	TCC SEL	+28 V	O.C.	TCC line selection	
	TCC1 SEL	+28 V	O.C.		
	TCC2 SEL	+28 V	O.C.		
R	TEST PIN A1				
	FCC1-FCU Spare	+28 V	O.C.	Spare	
	CP POWER INHIBIT 1	GND	O.C.	FCU supply inhibition	
	CONNECTOR J2				
	AP2 CLUTCH SUP 28 V	+28 V	O.C.	AP engagement,	Segregation 2
	AP2 TOGGLE HOLD C	GND	O.C.	AP engagement	
	AP2 ON	+28 V	O.C.	AP engagement	
	CMD1 ENGD	+28 V	O.C.	FCC line selection	
	CMD2 ENGD	+28 V	O.C.		
	FD1 ENGD	+28 V	O.C.		
	FD2 ENGD	+28 V	O.C		
	CWS1 ENGD	+28 V	O.C.		
	CWS2 ENGD	+28 V	O.C.		

	SIGNAL	LEVEL		FUNCTION	COMMENTS
		ACTIVE	NOT ACTIVE		
	LAMP TEST	+28 V	O.C.	Lamp and display test	
	DIM COMMAND 2	GND	O.C.	Pushbutton switch dimming	
	TCC1 SEL	+28 V	O.C.	TCC Line selection	
	TCC2 SEL	+28 V	O.C.		
R	TEST PIN A2				
	FCC2-FCU	+28 V	O.C.		
	Spare			Spare	
	CP POWER INHIBIT 2	GND	O.C.	FCU supply inhibition	

ARINC OUTPUTS

	OCTAL LABEL	PARAMETER MAME	TRANSMIT INTERVAL (ms)	UNSIGNED SIGNIFICANT BITS	UNIT	RANGE	COMMENTS
R	103	SELECTED AIRSPEED	56	11	knots	100-399	
R	106	SELECTED MACH	56	12	mMACH	100-990	
R	102	SELECTED ALT	56	16	Feet	0-65536	
R	101	SELECTED HDG	56	12	degs	± 180	
R	104	SELECTED VSPEED	56	10	ft/min x100	-50 +40	
	026	SEL AIRSPEED BCD	60	11	knots	100-399	
	023	SEL HEADING BCD	60	11	degs	±180°	
	020	SEL VSPEED BCD	60	15	ft/min x100	-50 +40	
	025	SEL ALT BCD	60	19	feet	0-40000	
	270	DISCRETE WORD 1	60	21	ft/min x100	-50 +40	
	271	DISCRETE WORD 2	60	21	ft/min x100	-50 +40	
	273	DISCRETE WORD 3	60	21	ft/min x100	-50 +40	
	274	DISCRETE WORD 4	60	21	ft/min x100	-50 +40	
	275	DISCRETE WORD 5	60	21	ft/min x100	-50 +40	
	350	SOFTWARE IDENT	at power on	8			Transmitted once at power on

OUTPUT DISCRETE SIGNALS

SIGNAL	LEVEL		FUNCTION	COMMENTS
	ACTIVE	NOT ACTIVE		
CONNECTOR J1				
AP1 TOGGLE SW M	+28 V	O.C.	AP1 engagement	Power for clutch
AP1 TOGGLE SW C	+28 V	O.C.	AP1 engagement	
PROFILE	+28 V	O.C.		
TEST PIN C OUT	+28 V	O.C.		
FCU-FCC SPARE	+28 V	O.C.	Spare	
FCU1-HEALTHY	+28 V	O.C.		
CONNECTOR J2				
AP2 TOGGLE SW M	+28 V	O.C.	AP2 engagement	Segregation 2
AP2 TOGGLE SW C	+28 V	O.C.	AP2 engagement	
PROFILE	+28 V	O.C.		
TEST PIN C OUT	+28 V	O.C.		
FCU-FCC SPARE	+28 V	O.C.	Spare	
FCU2-HEALTHY	+28 V	O.C.		

ARINC OUTPUT SIGNALS

BIT NUMBER	FUNCTIONS	SOURCE	FUNCTIONS	SOURCE
1-8	LABEL 270 (WORD 1)		LABEL 271 (WORD 2)	
9			ALT HLD P/BSW	FCU
10				
11				
12	FCC1 ORDERS SELECT	FCU		
13	FCC2 ORDERS SELECT	FCU	NAV P/BSW	FCU
14				
15			LAND P/BSW	FCU
16			V/L P/BSW	FCU
17				
18			CMD/CWS P/BSW	FCU
19				
20				
21	BANK ANGLE NORM 15°			
22	LIMIT SELECT 0 0			
23	0 1	FCU		
	0 1			
24	HDG SEL P/BSW	FCU	LVL/CH P/BSW	FCU
25	A/THR P/BSW	FCU	PULL V/S KNOB	FCU
26			PROFILE P/BSW	FCU
27			PULL ALT KNOB	
28	PULL SPD/MACH KNOB	FCU	ALT SEL KNOB	FCU
29	DFA P/BSW (If fitted)	FCU	PULL HDG SEL KNOB	FCU
30	SSM		SSM	
31				
32	PARITY		PARITY	

BIT NUMBER	FUNCTIONS	SOURCE	FUNCTIONS	SOURCE
1-8	LABEL 273 (WORD 3)		LABEL 274 (WORD 4)	
9				
10				
11			SPD MODE	FCC
12	ALT ACQ ARM	FCC	PROFILE MODE	FCU
13			ALT	FCC
14	NAV MODE ARM	FCC	V/S	FCC
15	PROFILE MODE ARM	FCC	FLARE	FCC
16	LOC ARM	FCC	PITCH GA	FCC
17				
18			PITCH TO	FCC
19			MACH	FCC
20	LAND ARM	FCC	GS	FCC
21				
22	GLIDE ARM	FCC	- - TRK	FCC
23	VOR ARM	FCC		
24	- - CLIMB	FCC		
25	- - DES	FCC		
26				
27				
28				
29			- - CPT	FCC
30	SSM		SSM	
31				
32	PARITY		PARITY	

BIT NUMBER	FUNCTIONS	SOURCE	COMMENTS
1-8	LABEL 275 (WORD 5)		CODED INFORMATION
9			
10			LOC CPT 15 and 21
11	NAV MODE	FCC	
12	HDG HLD	FCC	
13	HDG SEL	FCC	
14			
15	LOC CPT MODE	FCC	
16	ROLL OUT	FCC	
17	YAW TO	FCC	
18	ROLL GA	FCC	
19			
20			
21	LOC CPT MODE	FCC	
22	VOR	FCC	
23	- - TRK	FCC	
24			
25			
26			
27			
28	ALIGN	FCC	
29			
30	SSM		
31			
32	PARITY		

B. FCU Operation

(1)Hardware description

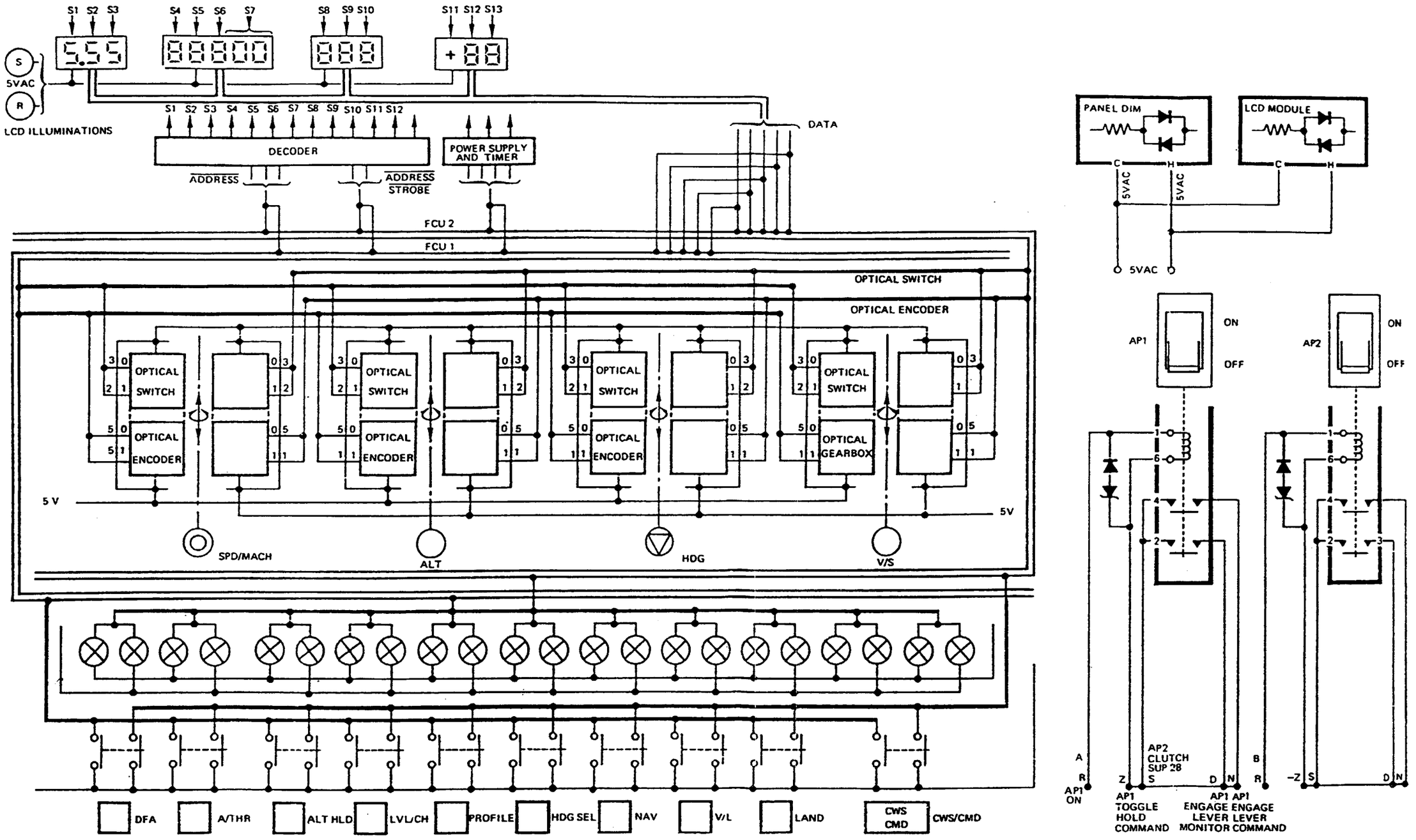
The FCU is composed of two independent sides (FCU No.1 and FCU No.2)
R (power supplies independent) built around microprocessors.
Each processor is associated with both sides of the AFS and, except for self-test function, processor No.1 only operates ; the other processor constitutes a cold redundancy which is used only in the event of failure of processor No.1. This principle applies irrespective of AP and FD engagement status.
R The following figure (Ref. Fig. 002) gives the FCU operating principle
R and shows how pilots' selections are processed :
R - the mode selection pushbutton switches are equipped with two contacts
R (one contact is connected to FCU No. 1, the other one to FCU No. 2)
R - the optical switches and encoders associated with the display controls
R are duplicated (one part is connected to FCU No. 1, the other part is
R connected to FCU No. 2).
R - each luminous indication (engaged mode confirmation, PRESET function,
R CWS/CMD selection) is achieved by two 28V bulbs connected in parallel
R and controlled by the FCU operating side.
R Displays of LCD type and associated controls are unique but controlled
R by either FCU depending on their supply conditions.
AP engage lever circuits are independent from the FCU microprocessors.
R Figure (Ref. Fig. 003) shows the organization principle of an FCU
R channel : it can be seen clearly that FCU No. 1 switching over to the
R FCU No. 2 occurs in the event of failure detected by the watchdog or
R in the event of FCU No. 1 power supply failure.

(2)Software description (Ref. Fig. 004)

The FCU software modules are written in assembler or in PLM 85 language according to the type of software to be processed.
The FCU computation time is 60 ms.
Each FCU receives the output data buses from FCC1 and FCC2 and from TCC1 (TCC2 if fitted).
The FCU uses the data bus on side 1 or side 2 according to AP/FD/ATS engagement.
The FCC data bus provides the FCU with the following data :

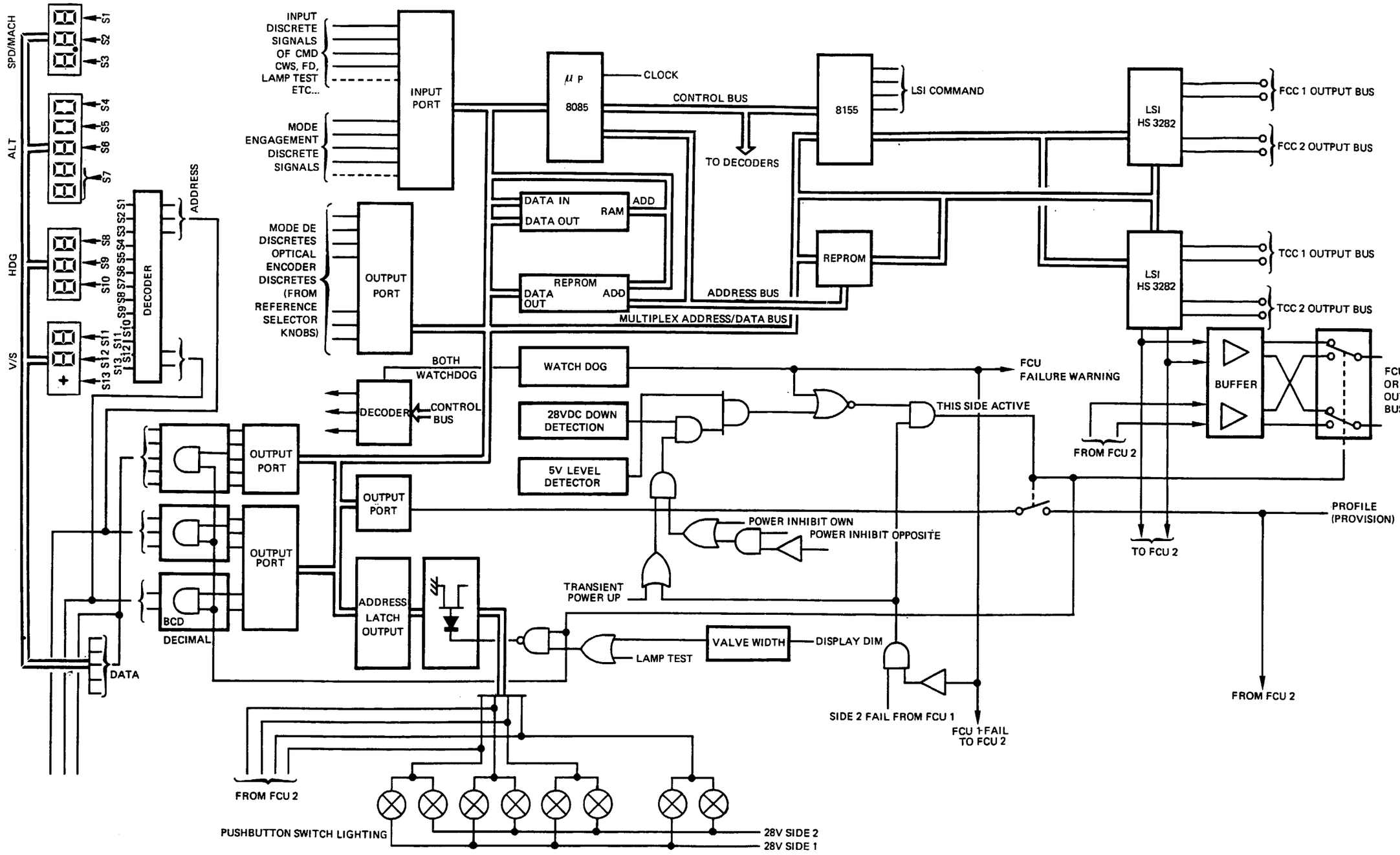
DATA	FUNCTION
Aircraft speed	SPD/MACH counter synchronization - initialization
A/C Mach number	SPD/MACH counter synchronization
A/C heading	HDG SEL counter synchronization - initialization
A/C vertical speed	V/S counter synchronization - initialization
A/C altitude	ALT SEL counter synchronization
Airspeed computed from selected Mach number	Repetition on PFD
FCC discrete words	Mode selection pushbutton switch illumination PROFILE boolean generation

DATA	FUNCTION
	Repetition of FCC modes



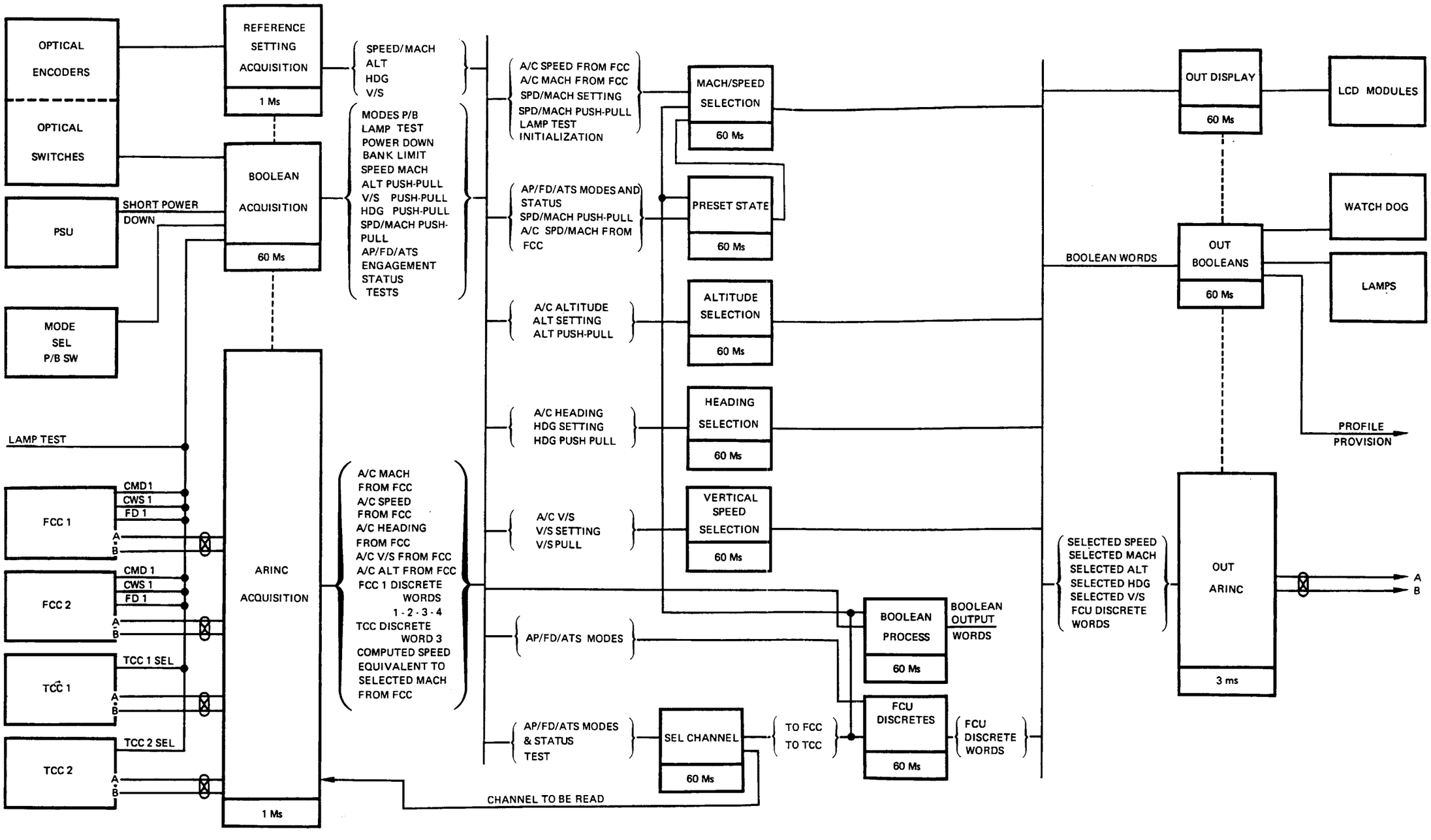
FCU Operating Principle
Figure 002

BM5 22 15 00 0 APM2 00



BM5 22 15 00 0 ARM2 00

FCU Organization Principle of One Channel
Figure 003



BM5 22 15 00 0 ATM2 00

FCU - Software Organization
Figure 004

The TCC provides the FCU with the following data :

DATA	FUNCTION
Discrete words	Mode selection pushbutton switch illumination

When the electrical network is energized the reference displays are initialized with the aircraft parameters (Vc, ALT, HDG, V/S if V/S mode is engaged and transmitted by the FCC).
In the absence or in the event of invalidity of the FCC data, the FCU initializes on set values :
SPD/MACH = 100 kt
ALT = 5000 ft
HDG = 000
V/S = ---
When the electrical network is energized, an automatic test of FCU 1 and 2 is carried out followed by FCU No. 2 power supply cut off.
FCU No. 2 will be supplied again only if FCU No. 1 is reported faulty.
FCU No. 1 change-over to FCU No.2 results in disconnection of systems engaged (AP, FD and autothrottle) ; the FCU displays initialize on the aircraft values.
After a loss of both FCCs and with a Mach indication displayed on the FCU, the SPD/MACH display is initialized again with 100 kt resulting in the autothrottle disconnection (if the SPD mode was engaged).
Engagement of the SPD mode will not be possible as long as the display (100 kt) value is not modified.
It is to be noted that the PRESET function and MACH mode are lost if neither FCC is available.

C. FCU Utilization

(1)AP engagement

The AP engage levers have two positions : ON and OFF. The AP engage lever is self-held in ON position only if the logic conditions in the command and monitoring channels are met (hard-wired logic).

When the AP lever is placed in ON position and if all the conditions are satisfied, the command logic sends a ground signal to one end of the self-holding relay coil and the monitoring logic closes the contacts of relays (aircraft wiring) which results in sending a 28VDC signal to the other end of the self-holding relay.

All the conditions for AP engagement are given in 22-16.

(2)Mode selection

The modes are selected by :

- pressing the mode selection pushbutton switches. Selection of a mode results in the illumination of the corresponding pushbutton switch (green) if selection has been taken into account by the TCC or the FCC.
- pulling the reference selector knobs (Ref. 22-17 for definition of conditions).

Action on an illuminated pushbutton switch results either in :
the return to a basic mode if mode was active (except for LAND mode in LAND track phase)
or in :
the arming loss in the event of an armed mode.

It is to be noted that no pushbutton switch is assigned to the basic modes (HDG, V/S and ALT ACQ mode).

HDG and V/S selection is automatic at FD engagement or AP engagement in CMD mode if FD was not previously engaged.

Return to a basic mode is achieved by pressing the illuminated pushbutton switch corresponding to the longitudinal or lateral mode.

The ALT ACQ mode is always armed (except in some cases described in 22-17) and becomes automatically active at capture of the flight level displayed on the FCU.

Takeoff (TO) and go around (GA) modes are engaged by operating the TO/GA levers on the throttle control levers.

(3)Display of reference values

At first energization of the FCU the reference displays are initialized with the aircraft parameters (Vc, ALT, HDG, V/S if V/S mode is engaged and transmitted by the FCC) and an automatic test of FCU 1 and 2 is carried out followed by FCU 2 power supply cutoff.

In the absence or in the event of invalidity of the FCC data, the FCU initializes on set values :

SPD/MACH = 100 kt
ALT = 5000 ft
HDG = 000
V/S = ---

FCU No.2 will be supplied again only if FCU No.1 is reported faulty. FCU No.1 change-over to FCU No.2 results in disconnection of systems engaged (AP, FD and autothrottle) ; the FCU displays initialize on the aircraft values.

After a loss of both FCCs and with a Mach indication displayed on the FCU, the SPD/MACH display is initialized again with 100 kt resulting in

the autothrottle disconnection (if the SPD mode was engaged).
Engagement of the SPD mode will not be possible as long as the display (100 kt) value is not modified.
It is to be noted that the PRESET function and MACH mode are lost if neither FCC is available.

(a)Speed/Mach

The selector knob enables the reference value to be modified.
The pushbutton switch located on the left side of the counter enables to switch from the SPEED to the MACH function and vice versa.
An automatic switching from speed to mach in climb and from mach to speed in descent at 25,400 ft. is performed.
Display value varies between :
100 and 399 kt
.01 and .99 Mach number
Sensitivity is :
1 kt per click (32 clicks per turn)
.01 Mach per click (32 clicks per turn)
Speed to Mach change-over and vice versa enables aircraft speed or Mach value to be displayed or gives a Mach value of .01 corresponding to 100 kt and vice versa.
The speed displayed on the counter is also indicated by the speed scale on the PFD.
When a Mach value is displayed in the counter, the FCU provides the PFD with a computed selected speed corresponding to the displayed Mach number (this speed data is computed by the FCC and then transmitted to the FCU).
The SPD/MACH selector knob has three positions :
- in rest position (normal) it enables the desired value to be selected.
  If the PROFILE mode is engaged, the SPD/MACH display shows dashes.
  When turning the knob (normal position) the pilot can select an aircraft speed or Mach value and then a reference speed value for the PROFILE mode loss if the SPD/MACH mode is engaged. If the SPD/MACH mode is not engaged, display goes off after 10 s.
- by pushing the selector knob, possibility is given to activate the PRESET function.
  This function is cancelled by pushing the knob a second time.
- pulling the selector knob has no effect except if the PROFILE mode was engaged.
  If the PROFILE mode is engaged, pulling the knob disengages the PROFILE mode and engages the SPD/MACH mode for the AP/FD or the ATS depending on the flight phase.

The reference speed or Mach value is the aircraft (transmitted by FCC) value except if the pilot had previously selected a value.
The pushed-pulled positions are unloaded positions which return to normal (rest) through action of a spring.
NOTE : When SRS mode is disengaged, SPD/MACH window is synchronised on aircraft airspeed, if no speed or mach value is preset and if aircraft is higher than selected value.

(b)PRESET function utilization

A possibility is given to preset a speed or Mach value by pushing the

SPD/MACH selector knob. This action results in :
- the illumination of the PRESET light on the right of the SPD/MACH counter.
- the memorization of the reference value displayed before action on the knob.
  This reference value is still transmitted by the FCU to the users (PFD, FCC and TCC).

When the PRESET light is on, a speed or Mach value can be preset by turning the selector knob in rest position. As long as the PRESET light is on, the preset value is not transmitted to the FCU by the users.
When the preset value is taken into account the PRESET light goes off and the FCU sends the value displayed on the counter and used as a new reference value, through ARINC 429 output data bus.
In SPEED mode, it is possible to preset :
- a speed value
- a Mach number by pushing the SPD/MACH selector knob and then pressing the SPD/MACH pushbutton switch on the left side of the display.
  The display then shows a Mach value which can be modified by the pilot.

In MACH mode it is possible to preset :
- a Mach number
- a speed value as indicated above.

The following table gives the six configurations which can be found on the FCU.
The preset reference value is taken into account when :
- the aircraft reaches the preset speed value (AP/FD or autothrottle system holding the MACH)
- the aircraft reaches the preset Mach number (AP/FD or autothrottle system holding the speed)
- the ALT ACQUIRE conditions are met (the preset value is taken into account by the autothrottle system which changes to SPD/MACH mode)
- the LVL/CH mode is engaged (preset value is taken into account by the AP/FD)

CONFIGURATION	DISPLAY FCU	DISPLAY PFD	VALUE MODIFIED USING SELECTOR KNOB	PRESET INDICATION	FCU ARINC OUTPUTS
SPEED/NO PRESET	Selected speed	Selected speed	Selected speed	NO	Selected speed
MACH/NO PRESET	Selected Mach	Computed airspeed corresponding to selected Mach	Selected Mach	NO	Selected Mach and corresponding computed airspeed
SPEED/ PRESET SPEED	Preset speed	Selected speed	Preset speed	YES	Selected speed
MACH/PRESET MACH	Preset Mach	Computed airspeed corresponding to selected Mach	Preset Mach	YES YES	Selected Mach and corresponding computed airspeed
SPEED/ PRESET MACH	Preset Mach	Selected speed	Preset Mach	YES	Selected speed
MACH/ PRESET SPEED	Preset speed	Computed airspeed corresponding to selected Mach	Preset speed	YES	Selected Mach and corresponding computed airspeed

NOTE : - The PRESET function being inhibited in PROFILE mode, pushing the SPD/MACH selector knob has no effect if the PROFILE mode is active. Existing preset is cancelled when the PROFILE mode is selected (the PRESET light goes off and the SPD/MACH display shows dashes).

- When a preset value is displayed, pressing the SPD/MACH selector knob results in :
  . extinguishing of the PRESET light
  . display of the speed or Mach number memorized value.

Figures (Ref. Fig. 005, 006, 007) show examples of the PRESET function operational use.

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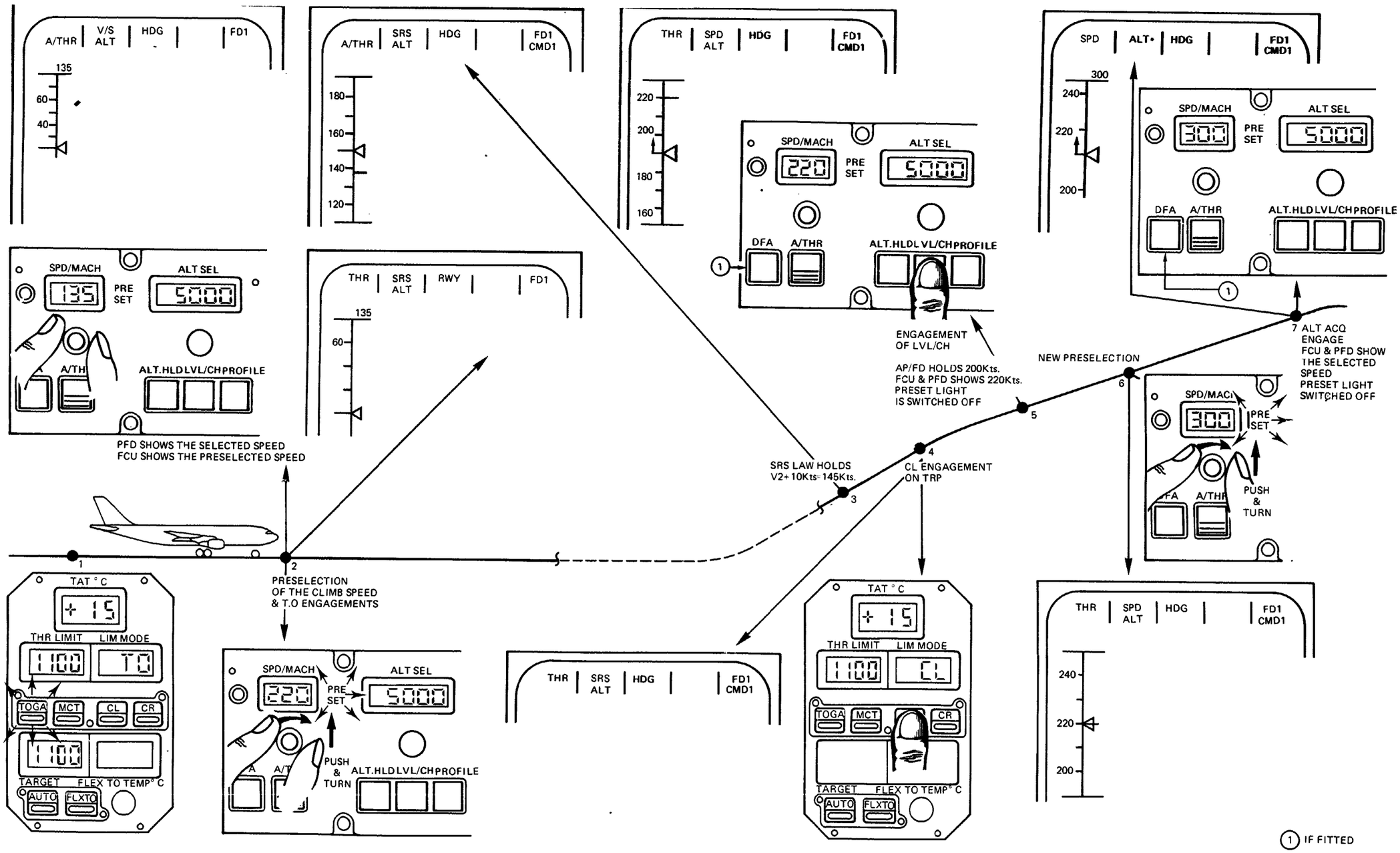
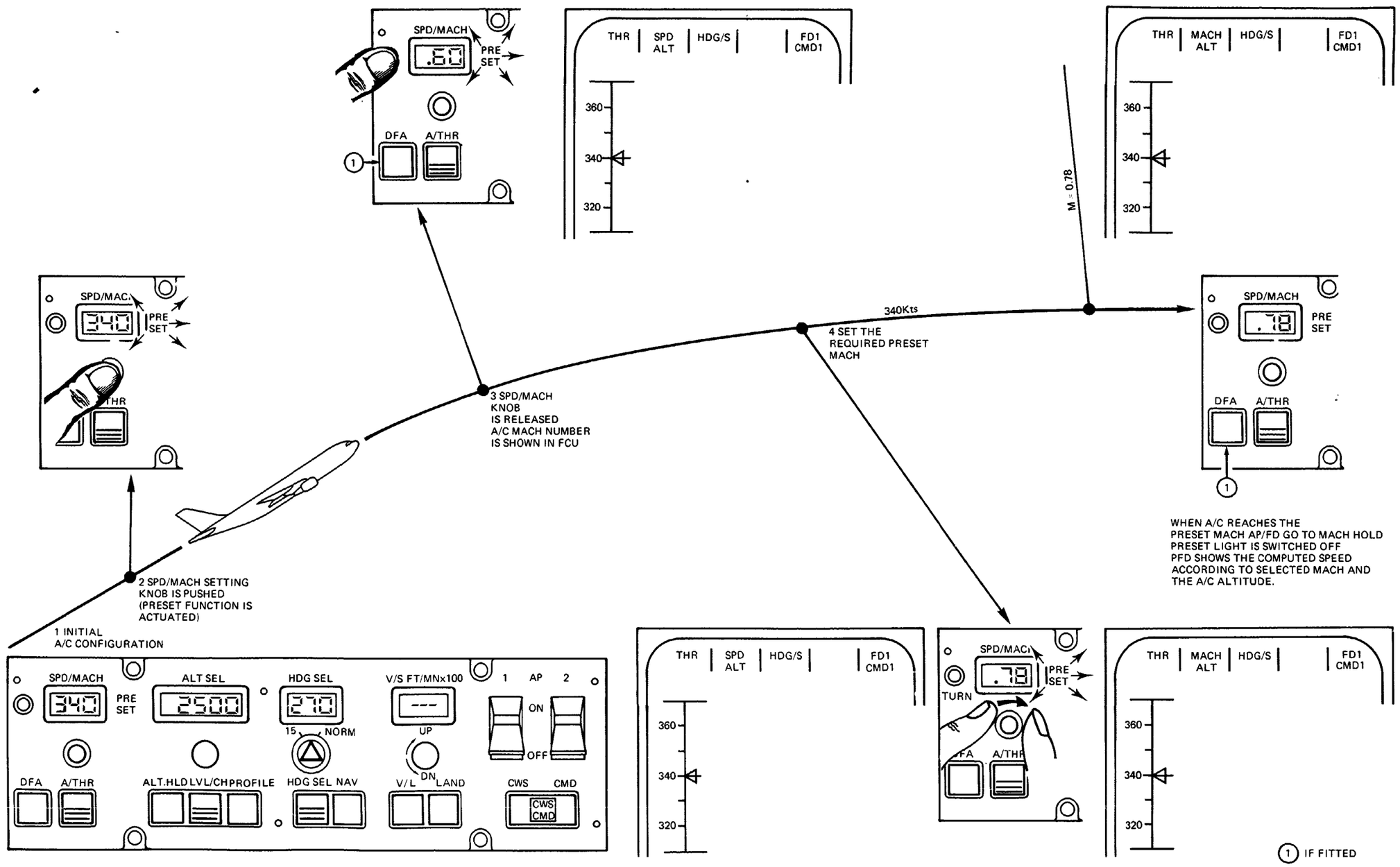


BM5 22 15 00 0 AXM2 – 10
DM5

Preset Function at TAKE OFF
Figure 005



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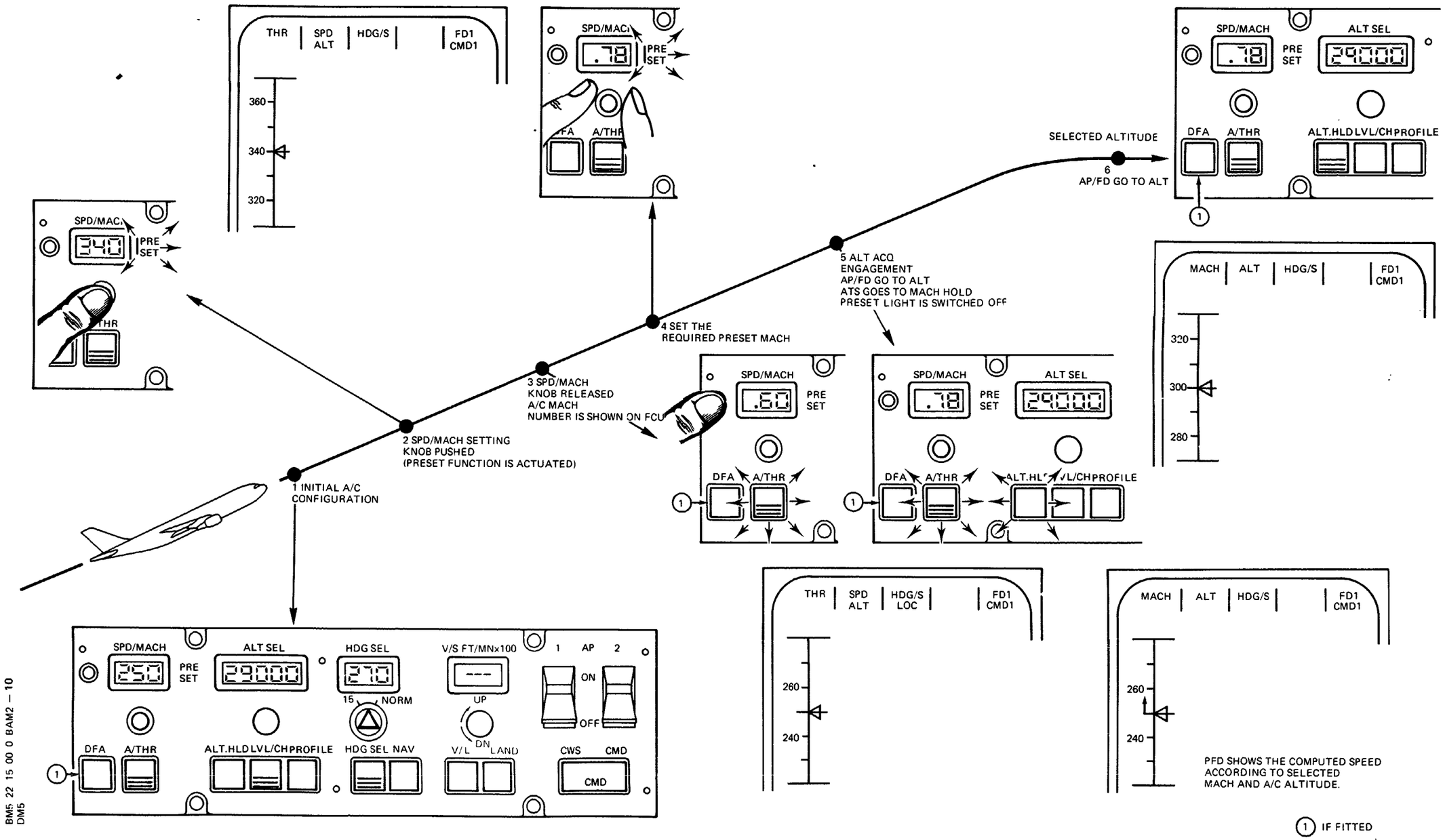
DM5

Preset Function **SPD/SEL/PRESET MACH** (Sheet 1/2)
Figure 006



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DM5

Preset Function **SPD/SEL/PRESET MACH** (Sheet 2/2)
Figure 007

(c)Altitude (ALT SEL)

When the electrical network is energized, the FCU shows the aircraft altitude value transmitted by the FCC or 5000 ft.

The notched selector knob under the ALT SEL display provides a means
R of modifying the altitude value between 100 and 45,000 ft.

The display sensitivity is :

- knob in rest position : 1000 ft.or 100 ft. per click
  32 clicks per turn

By pushing the knob the display sensitivity changes from 1000 ft. to 100 ft. per click and vice versa.

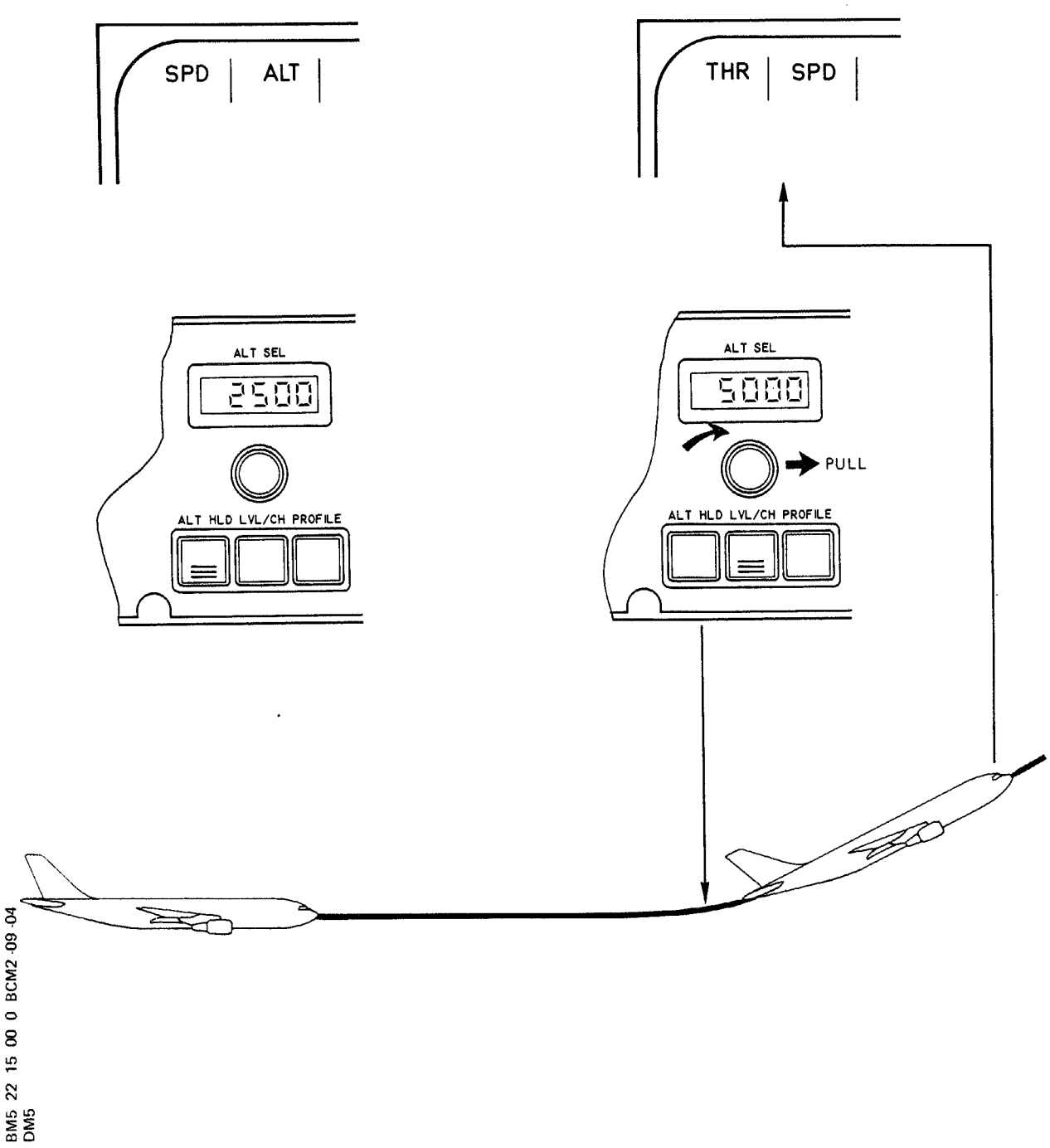
The display selector knob has three positions :

- pushed
- normal (rest)
- pulled

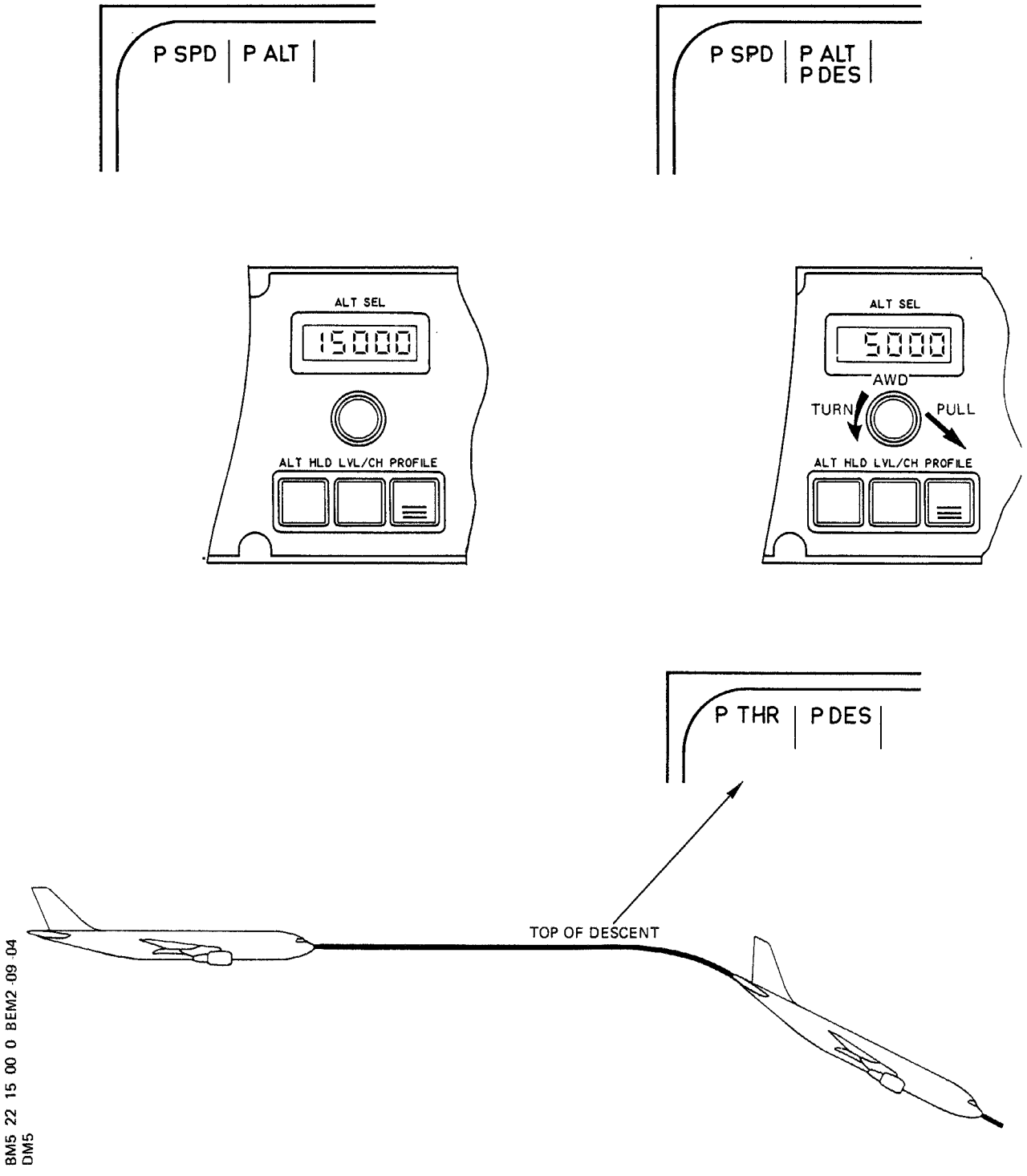
The two first positions are associated with the altitude display described above.

The pulled position enables to :

- engage the LVL/CH mode (Ref. Fig. 008).
- authorize a level change in PROFILE mode (Ref. Fig. 009)



Operational Use of FCU - Engagement of LVL/CH by Pulling Altitude Selector Knob
Figure 008



Operational Use of FCU - Altitude Release with PROFILE Mode Engaged
Figure 009

(d)Heading (HDG SEL)

When the electrical network is energized, the FCU shows the aircraft heading value transmitted by the FCC or OOO in the absence of this signal.

The selector knob under the HDG SEL display fulfills the following functions :

- outer ring : max roll angle selection in HDG SEL mode (15° or normal).

	POSITION OF MAX BANK SEL SW ON HDG SEL KNOB	
	NORMAL	15°
VOR (CAPTURE OR TRACK)	25°	15°
ROLL GA	30°	15°
HDG SEL	25° If HGD error > 15° gradually reduced to 15° for smaller HDG errors	15°
HDG	30°	15°

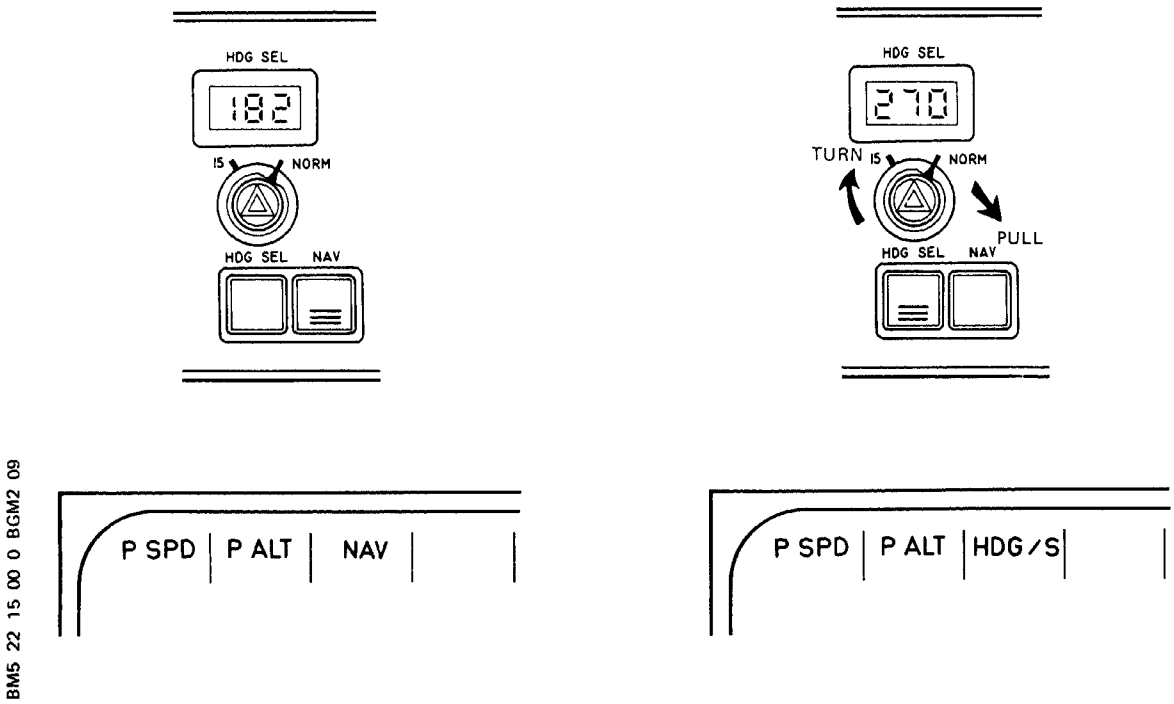
NOTE : - No limitation if CWS active.
- Bank limitation may be exceeded in case of engine failure by a maximum of 10° above 200 ft. gradually reduced to 5° between 200 and 100ft.
- With MAX BANK SELECTOR switch in 15° position, all FD orders are limited to 15° except for LOC or LAND mode.

- three-position selector knob (inner ring) :
  . pushed position : synchronization with the aircraft heading (function inhibited in HDG SEL mode)
  . rest position : modification of the heading selected reference value (slow display : 1° per click ; fast display : 2° per click and 32 clicks per turn).
  . pulled position : HDG SEL mode engagement (the HDG SEL mode can also be engaged by pressing the HDG SEL pushbutton switch)

The heading selected on the FCU is repeated on the Navigation Displays (ND)

(Ref. Fig. 010)

(Ref. Fig. 011)



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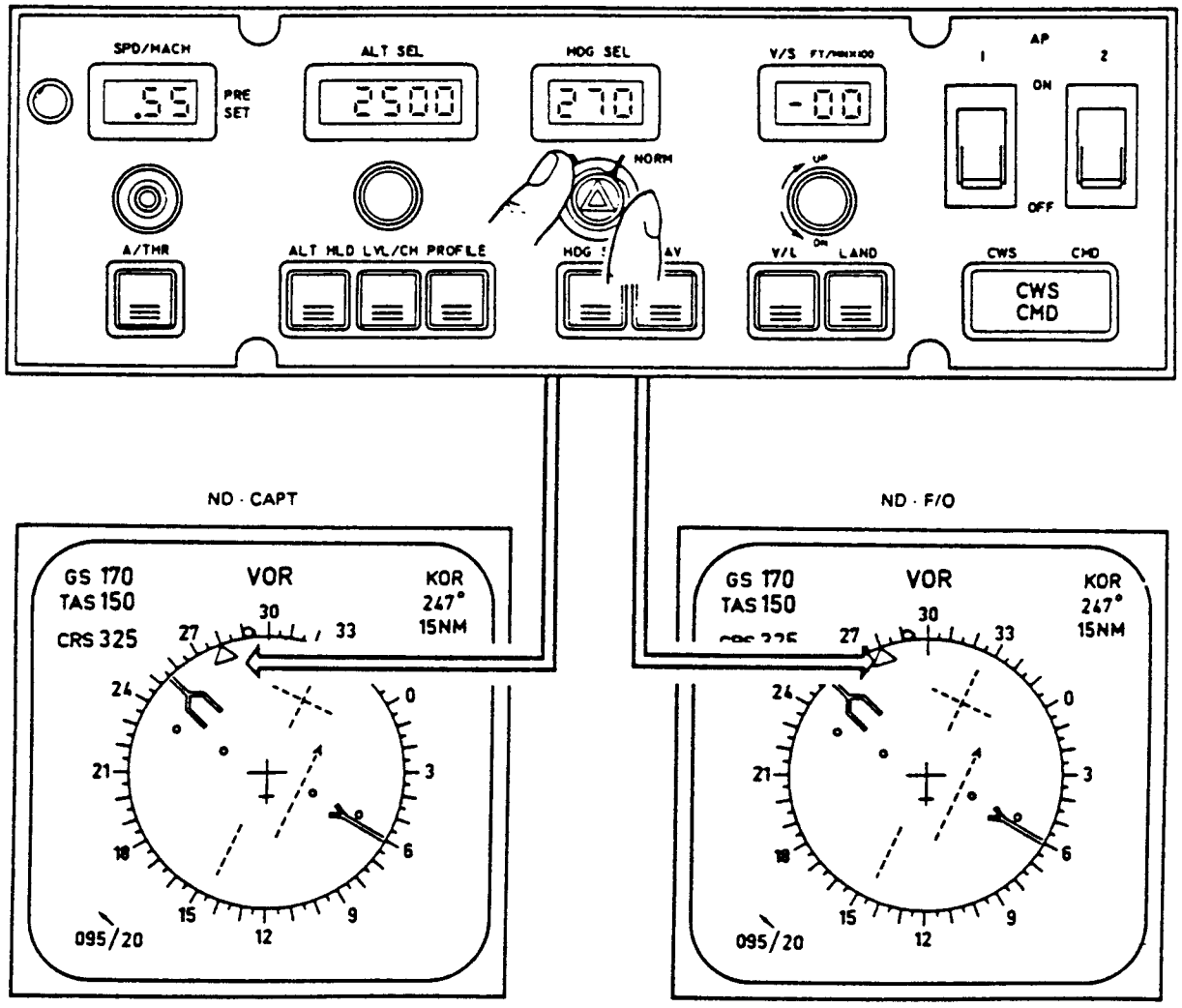
Operational Use of FCU - HDG SEL Mode Engagement
Using HDG SEL Selector Knob
Figure 010



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HDG Display
Figure 011

(e)Vertical speed (V/S)

Indication of the vertical speed is dependent on the V/S mode engagement.

When the V/S mode is disengaged, the display is off.

At V/S mode engagement the counter synchronizes with the aircraft vertical speed (transmitted by the FCC to the FCU).

After V/S mode engagement, it is possible to modify the reference value using the notched selector knob.

Selection is possible between -4000 ft./mn and + 3000 ft./mn

Sensitivity is :

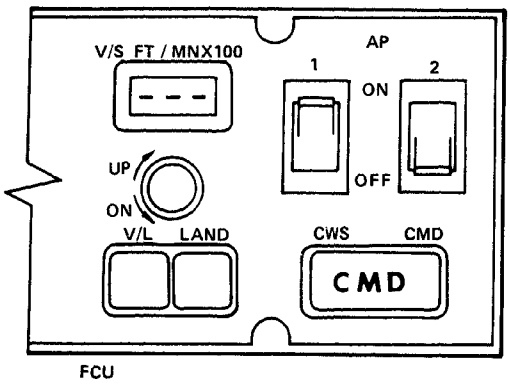
100 ft./mn per click and 32 clicks per turn.

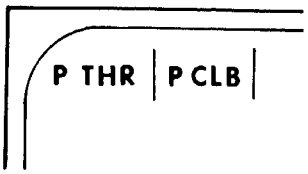
If the V/S mode is disengaged, turning the selector knob leads to the synchronization of the counter with the aircraft vertical speed. A value used as a reference if the V/S mode is engaged can then be selected.

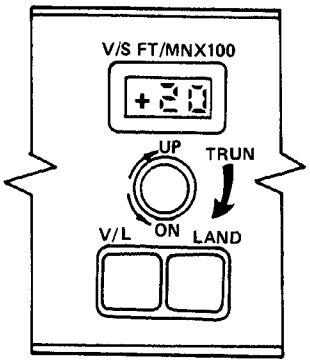
If the V/S mode is not engaged the vertical speed value display goes off after 10 seconds. The vertical speed selector knob has two functions :

- in rest position : turing the knob enables to select a reference value
- in pulled positon : V/S Mode is engaged (holding of the selected vertical speed or of the aircraft vertical speed at engagement)

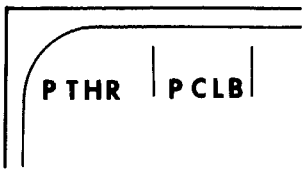
(Ref. Fig. 012)

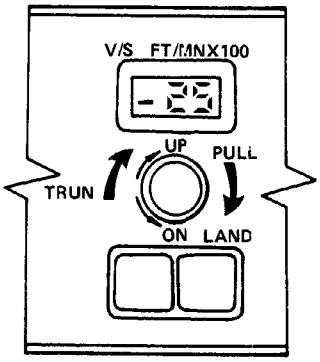




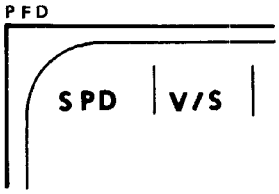


TURNING V/S SELECTOR KNOB LEADS TO SYNCHRONIZATION OF THE COUNTER ON THE PRESENT VERTICAL SPEED





SETTING A NEW V/S AND PULLING THE V/S SELECTOR KNOB LEADS TO/ V/S MODE ENGAGEMENT



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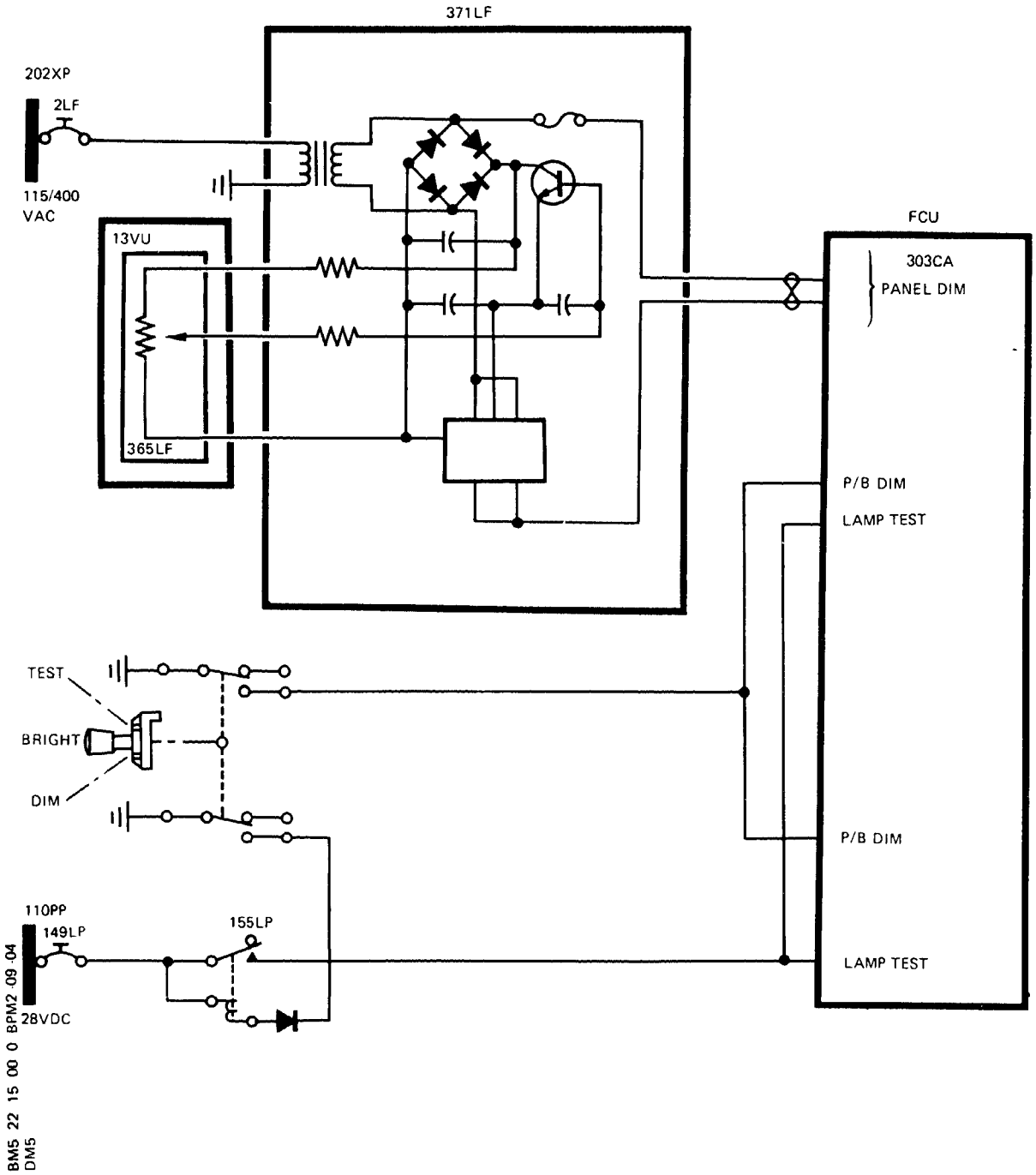
Operational Use of FCU - Engagement of V/S Mode
Using V/S Selector Knob
Figure 012

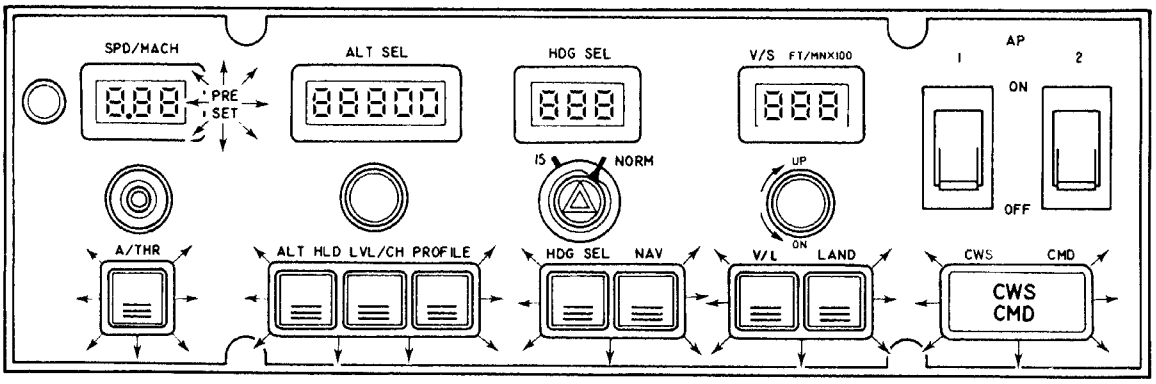
(4)Test of displays and dimming control
The test is performed by placing the **ANN LTS** selector switch in **TEST** position all the digits show 8 on the **FCU** except :
- the last two digits of the **ALT SEL** display which show 0 (zeros)
- the first digit of the **V/S** display which shows +

In addition :
- the **PRESET** light is on
- all the pushbutton switch lights indicating mode confirmation are on (=)
- the **CWS** and **CMD** legends come on
- the dot (for Mach information) in the **SPD/MACH** display is visible

When the **ANN LTS** selector switch is moved from **BRIGHT** to **DIM** position, the lighting of the pushbutton switches can be adjusted from a bright to a dimmed lighting.
Adjustment of the placard integral lighting and display lighting is achieved by means of a potentiometer located on the glareshield under the **FCU**.
(Ref. Fig. 013)
(Ref. Fig. 014)



FCU Dimming and Test Circuits Schematic
Figure 013



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FCU - Display and Filament Test
Figure 014

D. FCU Operational Logic

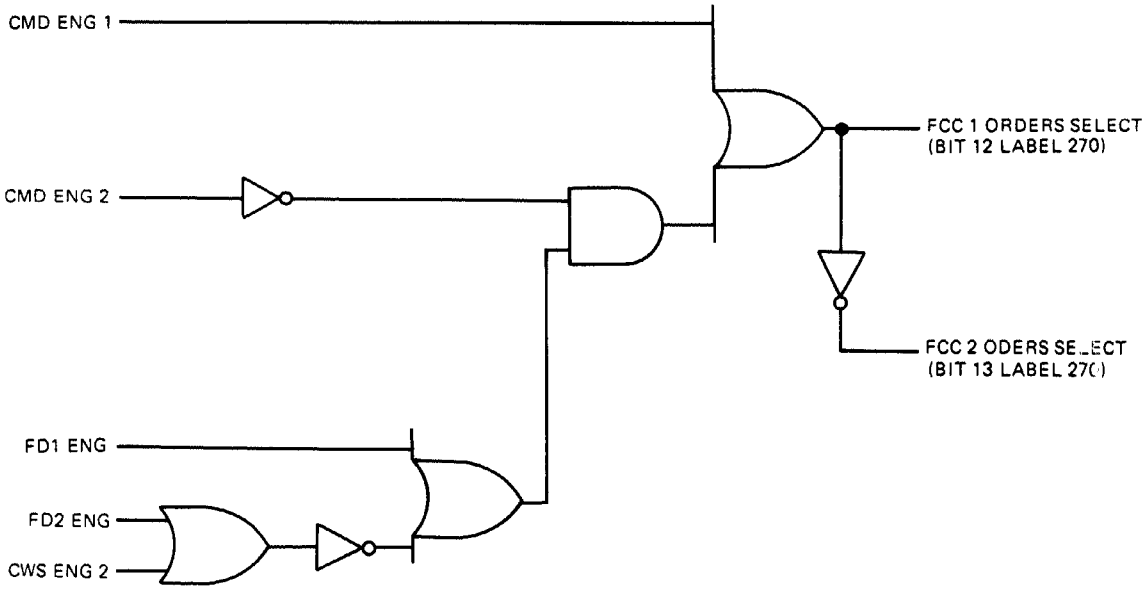
(1)FCC bus selection by the FCU (Ref. Fig. 015)

The FCU receives FCC1 and FCC2 buses with the following data :

- reference values (altitude, heading, vertical speed, aircraft Mach and speed)
- computed airspeed corresponding to selected Mach
- mode selection confirmation (for illumination of the pushbutton switches) etc.

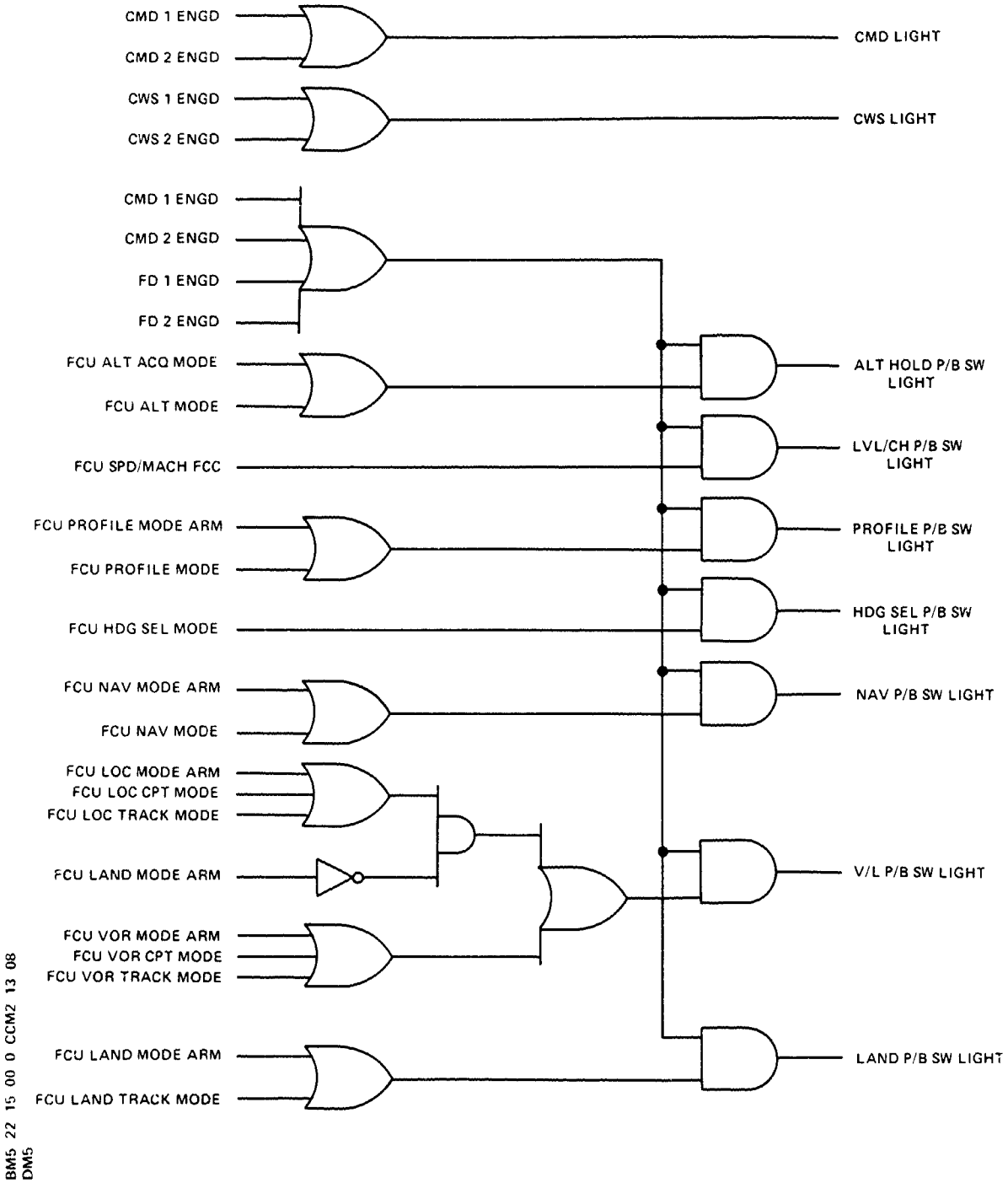
(2)Pushbutton switch illumination

The mode selection pushbutton switch illumination and the CMD/CWS selection logic is shown in figure (Ref. Fig. 016).



BM5 22 15 00 0 BRM2 00

FCU Operational Logic - FCC Bus Selection
Figure 015



FCU - Mode Selection Pushbutton Switch Illumination Logic
Figure 016

3. Individual Controls Located in Flight Compartment

In addition to autopilot engage levers located on the FCU, the FCC is connected to other controls :

- AP disconnect pushbutton switches
- TO/GA levers
- FD/FPV selector switches
- FD bars switching pushbutton switches (to the PFDs).

A. AP Disconnect Pushbutton Switches

Two AP disconnect pushbutton switches are located respectively on the Captain's and First Officer's control column wheels (Ref. Fig. 018)

These pushbutton switches enable quick disconnection of the autopilot :

- disconnection of the AP engaged in cruise
- disconnection of both APs if they have been engaged in LAND or GO AROUND modes.

AP disconnection is accompanied by specific aural and visual warnings. Warnings are cancelled by pressing one of both AP disconnect pushbutton switches (it is therefore necessary to press a second time one of the pushbutton switches if AP disconnection has been obtained by a first action on one of the pushbutton switches).

The AP disconnection can be indifferently obtained from any of the two pushbutton switches.

B. TO/GA Levers

The engagement of the takeoff (TO) or go around (GA) mode is achieved by two levers hinged on the throttle control levers.

TO/GA lever wiring principle is the same as that used for AP disconnect pushbutton switches : action on only one lever is sufficient to activate this function (Ref. Fig. 019).

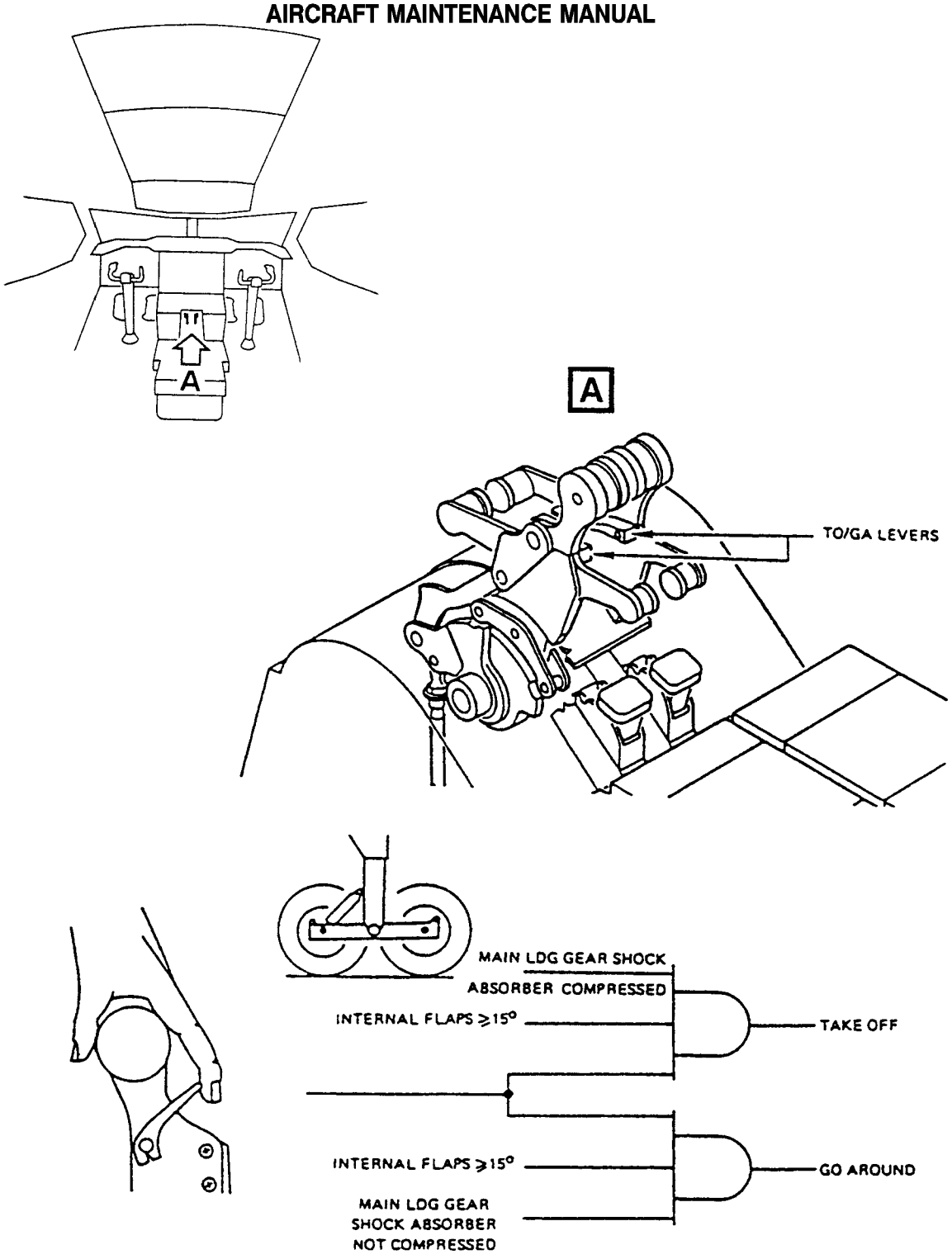
(Ref. Fig. 017)

C. FD/FPV Selector Switches

Ref. 22-16

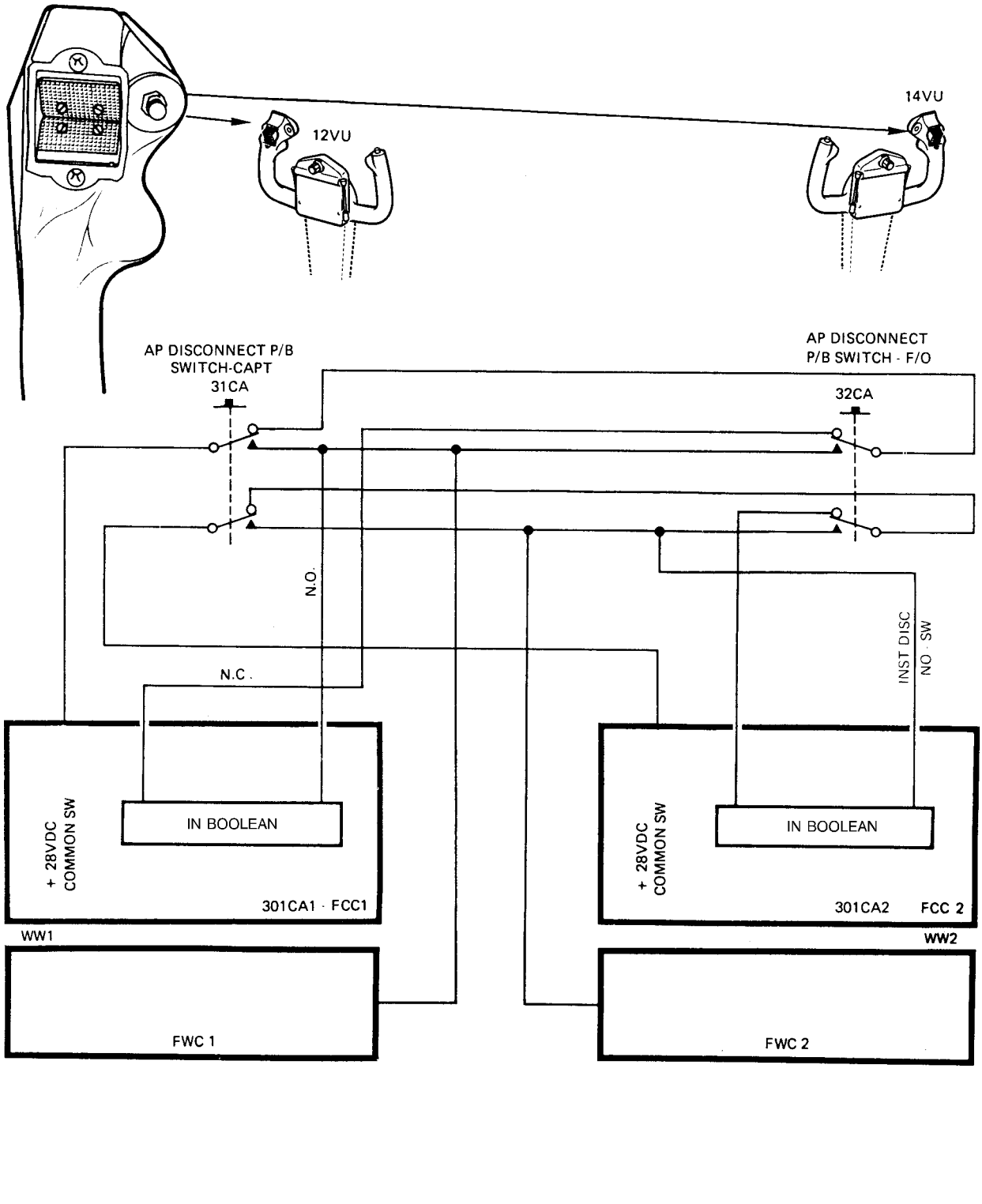
D. FD Bars Switching Pushbutton Switches

Ref. 22-16

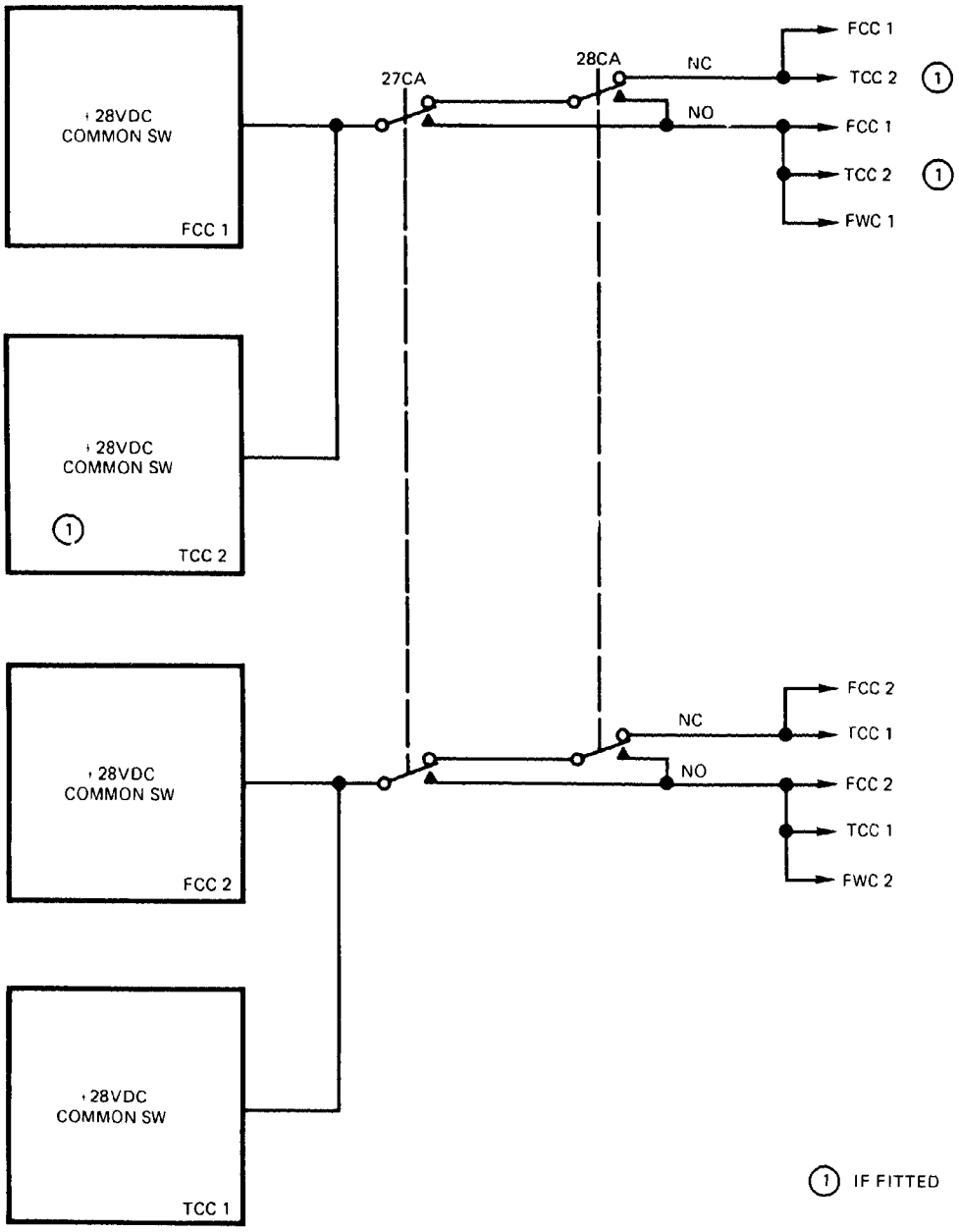


BM5 22 15 00 0 CGP2 01

TO/GA Levers
Figure 017



AP Disconnect Pushbutton Switches - Schematic
Figure 018



BM5 22 15 00 0 CJM2 09 04
DM5

TO/GA Levers - Schematic
Figure 019

4. Flight Mode Annunciators

A. General

The Flight Mode Annunciator (FMA) constitutes the upper part of the PFD (Ref. Fig. 020).

The PFD upper part is divided into five sections separated by a green vertical line.

These five sections respectively show (from left to right) :

- Section 1 : ATS mode engagement
  ATS mode arming
- Section 2 : AP/FD longitudinal mode engagement
  AP/FD longitudinal mode arming
- Section 3 : AP/FD lateral mode engagement
  AP/FD lateral mode arming
- Section 4 : Landing capability
- Section 5 : AP/FD engagement status

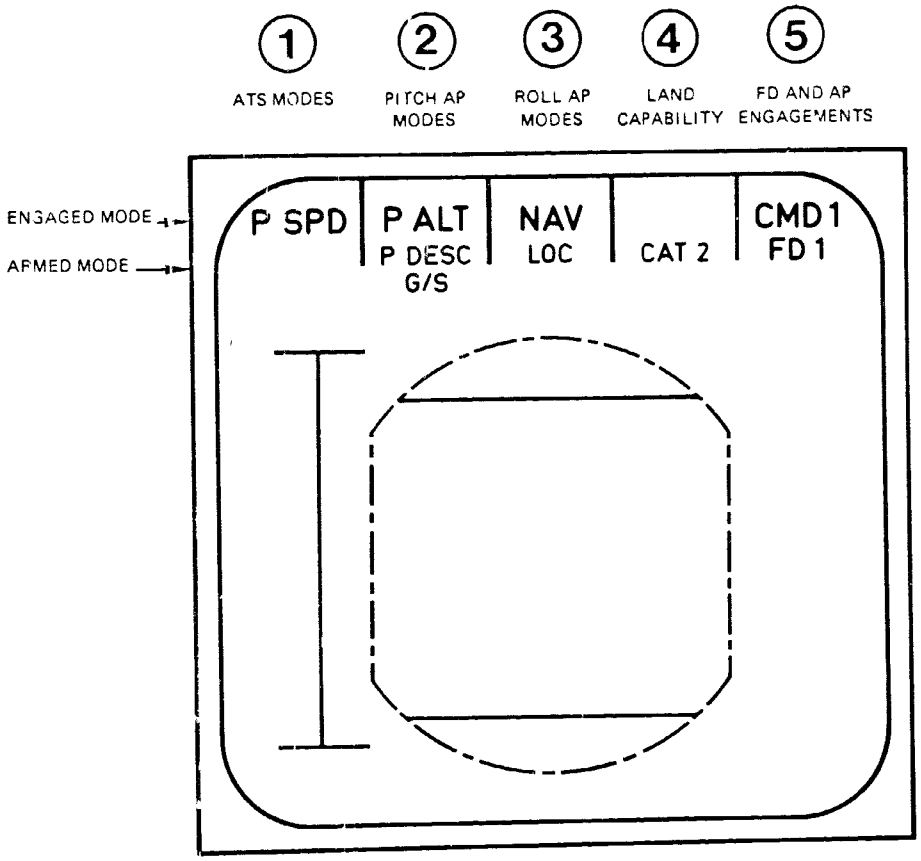
A description of the ATS modes appearing in the first section is given in 22-30.

Section 1, 2 and 3 are made up of several lines :

- upper line : engaged modes (green digits)
- lower line : armed modes (cyan digits).

NOTE : Section 2 (longitudinal modes) shows armed modes on two lines.

Section 4 shows magenta/white messages and section 5 shows white messages on two lines.



BM5 22 15 00 0 CLM2

PFD - Mode Annunciator
Figure 020

B. Longitudinal Modes

(1)Engaged modes

DISPLAY	COLOR	LOCATION ON	MEANING
V/S	Green	1st line 2nd section	Vertical speed hold mode active
ALT	Green	1st line 2nd section	ALT mode active
ALT*	Green	1st line 2nd section	Altitude acquire mode active
SPD	Green	1st line 2nd section	Speed hold mode active
MACH	Green	1st line 2nd section	Mach hold mode active
P.ALT	Green	1st line 2nd section	PROFILE mode active (coupling to FMC = flight level hold)
P.CLB	Green	1st line 2nd section	PROFILE mode active (coupling to FMC = climb)
P.DES	Green	1st line 2nd section	PROFILE mode active (coupling to FMC = descent)
SRS	Green	1st line 2nd section	Longitudinal guidance in takeoff or go around mode after selection of a lateral mode
G/S*	Green	1st line 2nd section	Glide capture
G/S	Green	1st line 2nd section	Glide hold
FLARE	Green	1st line 2nd section	LAND mode flare-out phase

(2)Armed modes

DISPLAY	COLOR	LOCATION ON PFD	MEANING
P.DES	CYAN	2nd line, 2nd section	PROFILE mode armed for descent
P.CLB	CYAN	2nd line, 2nd section	PROFILE mode armed for climb
G/S	CYAN	2nd line, 2nd section	GLIDE mode armed (LAND mode)
ALT	CYAN	2nd line, 2nd section	ALT ACQUIRE mode armed

C. Lateral Modes

(1)Engaged modes

DISPLAY	COLOR	LOCATION ON PFD	MEANING
HDG	GREEN	1st line, section 3	Heading hold
HDG/S	GREEN	1st line, section 3	Selected heading hold
NAV*	GREEN	1st line, section 3	Lateral track capture (NAV mode)
NAV	GREEN	1st line, section 3	Lateral track hold (NAV mode)
LOC*	GREEN	1st line, section 3	LOC beam capture (LOC or LAND mode)
LOC	GREEN	1st line, section 3	LOC beam hold (LOC or LAND mode)
VOR*	GREEN	1st line, section 3	VOR beam capture
VOR	GREEN	1st line, section 3	VOR beam hold
ROLL OUT	GREEN	1st line, sections 2 and 3	ROLL OUT phase (LAND mode)
RWY	GREEN	1st line, section 3	ROLL OUT phase (TO mode)

(2)Armed Modes

DISPLAY	COLOR	LOCATION ON PFD	MEANING
HDG/S	CYAN	2nd line, section 3	Selected heading armed at takeoff (TO mode) This indication will be displayed at the same time as green SRS (section 2) and RWY (section 3) indications
LOC	CYAN	2nd line, section 3	LOC mode armed (LOC or LAND mode)
NAV	CYAN	2nd line, section 3	NAV mode armed
VOR	CYAN	2nd line, section 3	VOR mode armed

D. AP/FD Common Modes

DISPLAY	COLOR	LOCATION ON PFD	MEANING
GO AROUND	GREEN	1st line, sections 2 and 3	GO AROUND mode engaged on both axes (indications displayed across dividing lines)
LAND	GREEN	1st line, sections 2 and 3	GS track and LOC track phases active, and H < 400 ft. This indication is displayed across sections 2 and 3

Engagement in TAKEOFF mode is indicated by SRS RWY

E. AP/FD Engagement

DISPLAY	COLOR	LOCATION ON PFD	MEANING
FD	WHITE	1st line, section 5	Flight Director engaged. The FD(1 or 2) which commands hte bars shall be mentioned on each PFD
CWS	AMBER	2nd line, section 5	AP engaged in CWS. CWS1 or CWS2 mentioned, depending on CWS engaged
CMD	WHITE	2nd line, section 5	AP engaged in CMD. CMD1 or CMD2 mentioned, depending on side engaged
DUAL	WHITE	2nd line, section 5	Both APs engaged (LAND or GA mode)

F. Capability Indication

DISPLAY	COLOR	LOCATION ON PFD	MEANING
CAT 1	MAGENTA	2nd line, section 4	LAND mode engaged (armed or active) only Category I is available
CAT 2	MAGENTA	2nd line, section 4	LAND mode engaged (armed or active) only Category II is available
CAT 3	MAGENTA	2nd line, section 4	LAND mode engaged (armed or active) Category III autoland is available

G. Operational Logic
Each PFD receives the two FCC1 and FCC2 data buses.
The FCC bus selection logic used for AP/FD mode indicating is as follows :
- priority to AP engagement in CMD mode over FD engagement
- priority to FCC1 over FCC2.
The following table gives the FCC bus selected according to AP/FD engagement.

FD COMMANDS "BOTH ON 2"	FD1 ENGAGEMENT	FD2 ENGAGEMENT	CMD1 ENGAGEMENT	CMD2 ENGAGEMENT	FCC BUS ON PFDs
-	-	-	YES	-	FCC1
NO	YES	-	NO	NO	FCC1
YES	YES	NO	NO	NO	FCC1
-	-	-	NO	YES	FCC2
YES	-	YES	NO	NO	FCC2
NO	NO	YES	NO	NO	FCC2

- = neutral status
FD command BOTH ON 2 = switching of FCC2 commands over the activated PFD1.

5. Individual Warnings

A. AP OFF Warning
AP OFF Warning is activated when the autopilot is lost (both AP engage levers in OFF position).
This warning consists of :
- an intermittent cavalry charge (this warning sounds for 1 s, then stops for 3 s, etc.).
- a red visual warning message on L ECAM display unit :

AFS

AP OFF

The visual warning is cancelled immediately, the aural warning is maintained for at least 1 s.

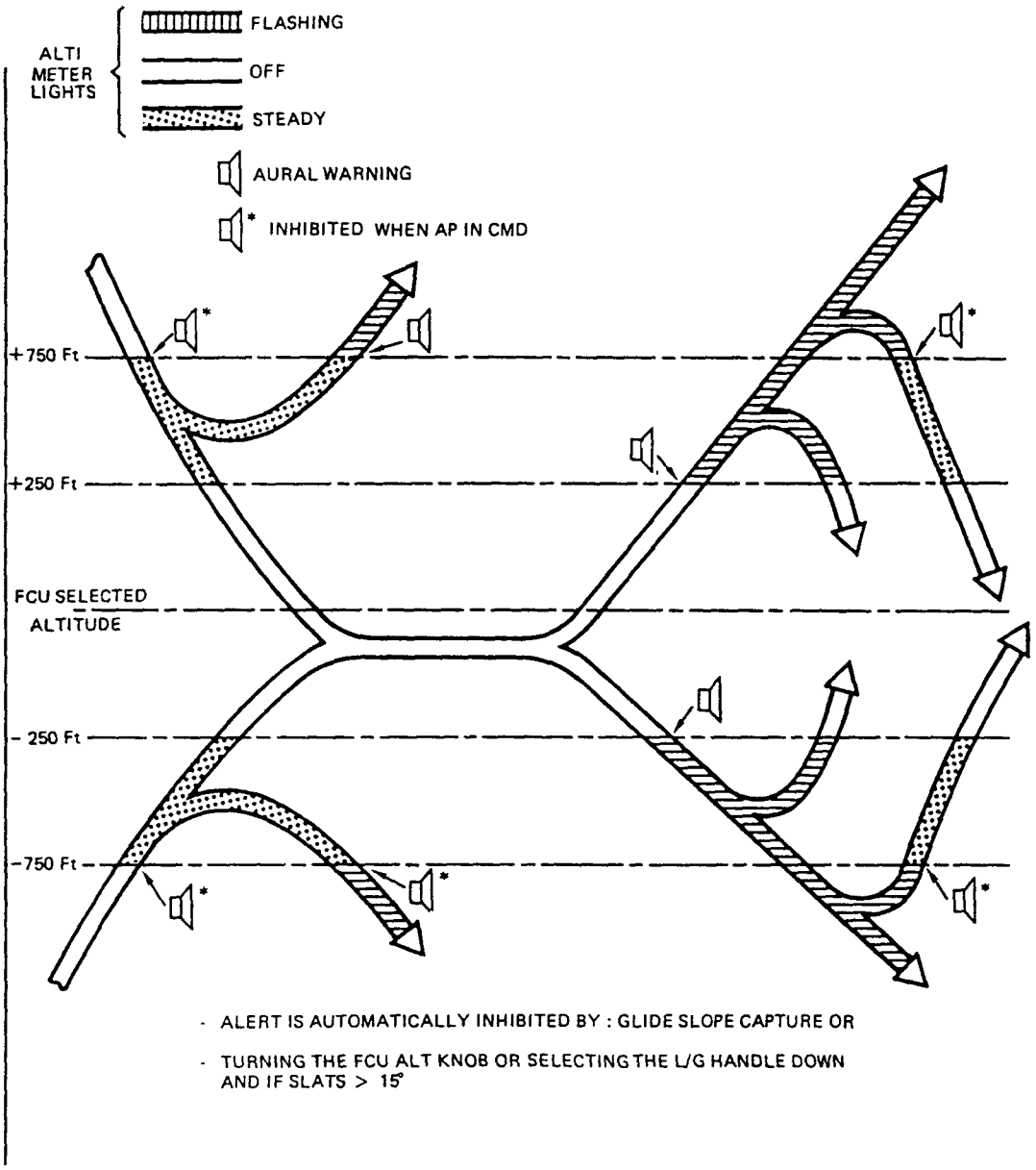
B. MASTER WARN Light
The MASTER WARN lights and the AP OFF message on the left ECAM display unit replace the AP OFF warning lights. Activation of the MASTER WARN light in the event of AP disconnection is identical to the AP OFF warning.

C. Altitude Alert
The Flight Warning Computers (FWC) generate an altitude alert warning (ALTITUDE ALERT) from altitude data corresponding to the value displayed

on the FCU. This warning can be activated in manual or automatic flight. This warning consists of an aural warning (C chord) and a visual warning (amber warning light on Captain and First Officer altimeters). The activation conditions for this warning are shown in figure (Ref. Fig. 021).
Warning is inhibited :
- when approaching selected level if AP is engaged in CMD mode,
- at glide capture,
- during selection of a new altitude on the FCU.

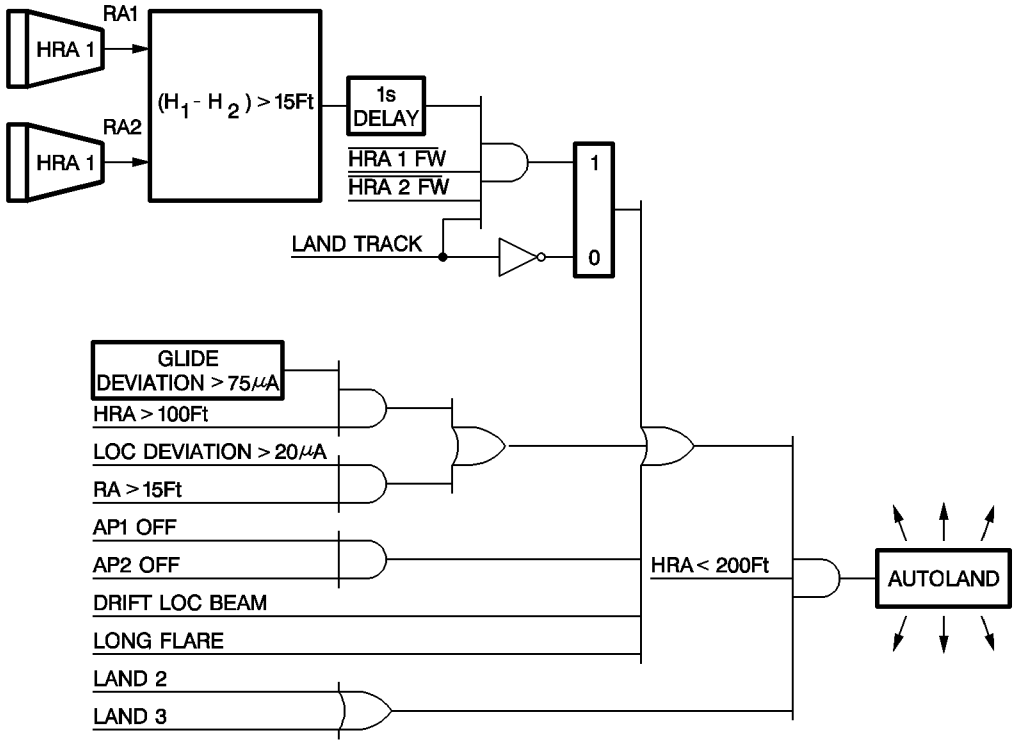
D. AUTOLAND Warning Light
An AUTOLAND warning light located on the glareshield in front of each pilot, indicates to the pilots that a manual take-over is required in the event of any abnormal automatic approach.
The AUTOLAND warning lights are red flashing lights which can be tested by pressing one of the lights.
The AUTOLAND warning is activated below 200 ft. if :
- AP OFF warning is activated
- flare duration counted from 50 ft. level is greater than 13.5 s
- LOC or G/S excessive deviation warning is activated
- comparison of altitudes from radio altimeters 1 and 2 shows a deviation greater than 15 ft. during 1 s (in LAND TRACK phase).

Figure (Ref. Fig. 022) shows the conditions for activating the AUTOLAND warning.
Figure (Ref. Fig. 023) shows the generating principle of the long flare warning.


ALTI METER LIGHTS
FLASHING
OFF
STEADY
AURAL WARNING
* INHIBITED WHEN AP IN CMD
+750 Ft
+250 Ft
FCU SELECTED ALTITUDE
- 250 Ft
-750 Ft
- ALERT IS AUTOMATICALLY INHIBITED BY : GLIDE SLOPE CAPTURE OR
- TURNING THE FCU ALT KNOB OR SELECTING THE L/G HANDLE DOWN AND IF SLATS > 15°

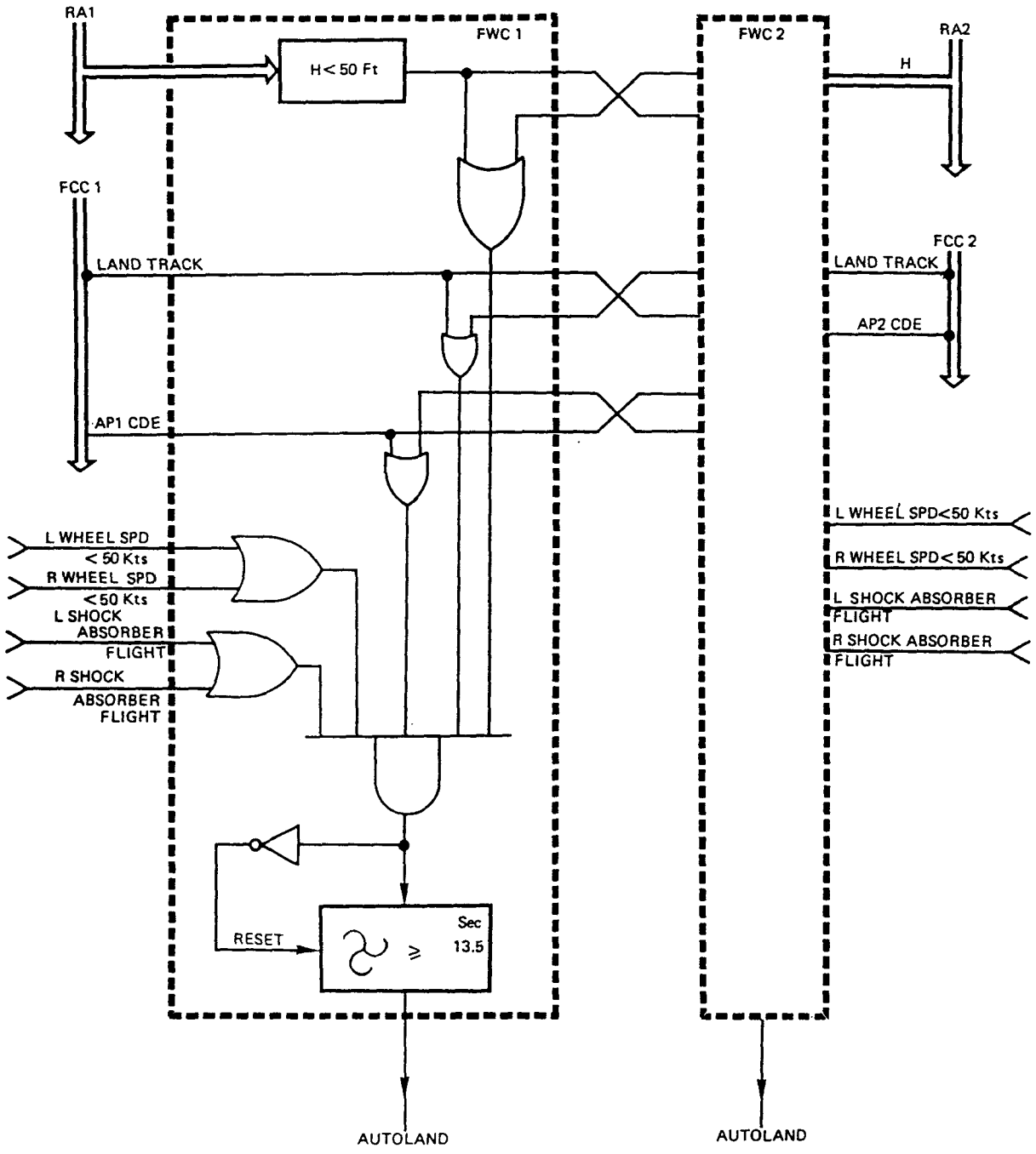
BM5 22 15 00 0 CPN2 00

Altitude Alert - Activation Function
Figure 021



BM5 22 15 00 0 CTN2 00

AUTOLAND Warning Logic
Figure 022



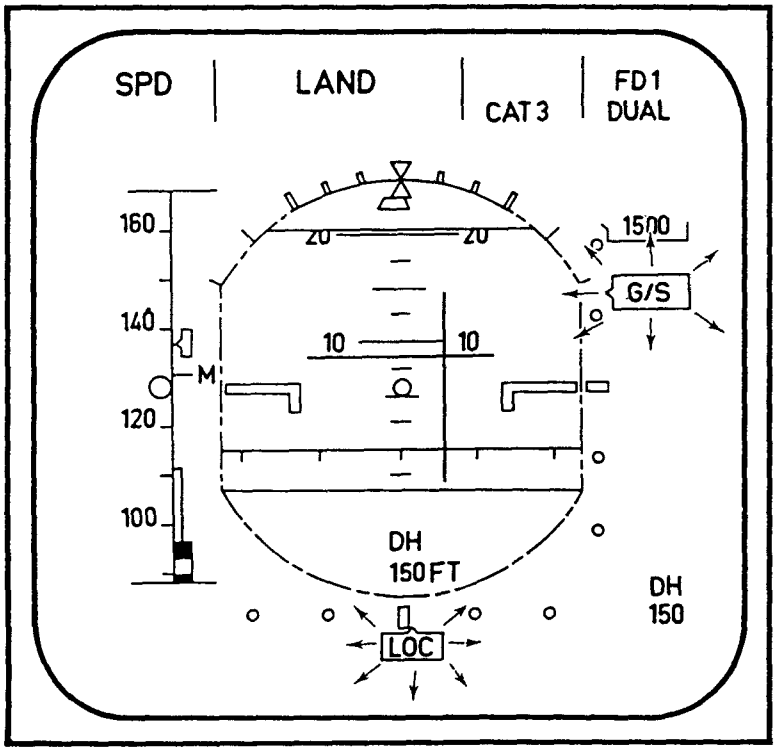
LONG FLARE Warning Logic
Figure 023

E. Excessive Deviation Warnings
In approach phase (APPR2, LAND2 or LAND3) a LOC or glide slope excessive deviation warning appears on the PFDs when the following conditions are met :

Glide slope excessive deviation	- glide slope deviation > 75 μ A - H > 100 ft. - GLIDE TRACK or LAND TRACK
LOC excessive deviation	- LOC deviation > 20 μ A - H > 15 ft. - LOC TRACK or LAND TRACK

This warning consists in a G/S or LOC indication flashing in the index representing the glide slope or LOC deviation on the PFD.
Detection of a transmitter failure also results in excessive deviation warnings in the same conditions.
(Ref. Fig. 024)

F. Landing Capability Reduction
A triple click aural warning is generated in the event of landing capability reduction (e.g. : LAND 3 to LAND 2).



GLIDE SLOPE
EXCESSIVE DEVIATION

LOC EXCESSIVE DEVIATION

BM5 22 15 00 0 CZN2 00

G/S & LOC Excessive Deviation Warnings
Figure 024

FLIGHT CONTROL UNIT (303CA) - REMOVAL/INSTALLATION

1. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit breaker Safety Clips
B.	Blanding Caps
Referenced Procedures	
- 22-00-00, P. Block 501	Auto Flight - General
- 22-40-00, P. Block 501	System Monitor

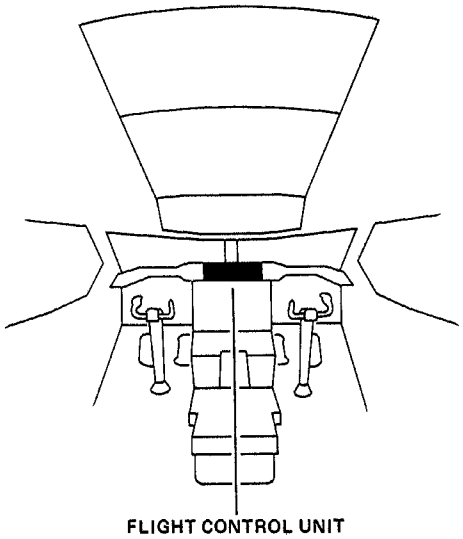
(Ref. Fig. 401)

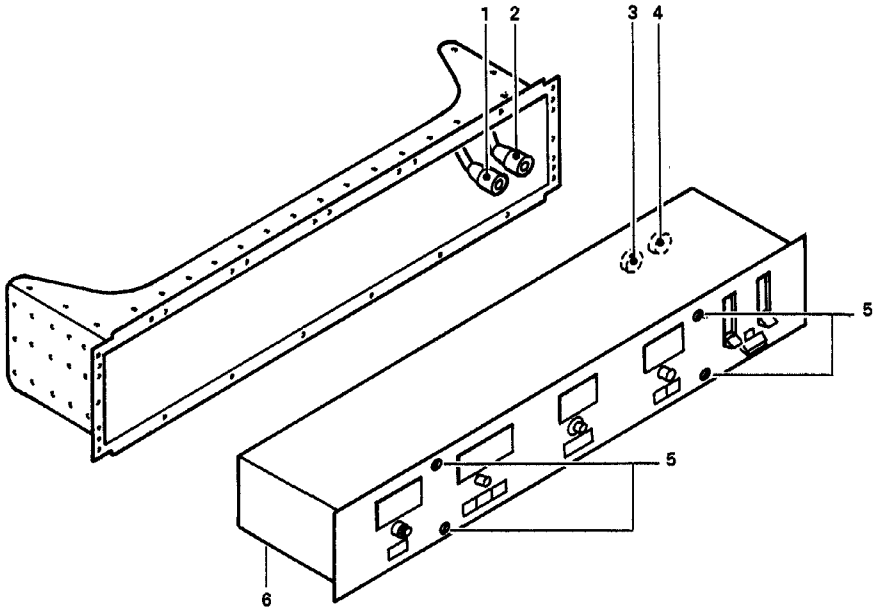
2. Procedure

A. Job Set-Up

(1)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H06
21VU	AFS/AP1-FD1/FCU	312CA1	102/H04
21VU	AFS/AP2-FD2/FCU	312CA2	102/H05
21VU	AFS/FAC1/28VDC	306CC1	103/G03
21VU	AFS/FAC2/28VDC	306CC2	103/G12
132VU	INTEGRAL LT/MAIN PNL & GLARESHLD	1LF	322/N76
132VU	INT LT/ANN LT/TEST/MAN	149LP	324/L82
132VU	AFS CONTROL PANELS/FCU & TRP & MTP	313CA1	325/K77





BM5 22 15 12 4 AAM2 00

Flight Control Unit
Figure 401

NOTE : To facilitate FCU removal, remove the upper part of the glareshield panel.

B. Removal
(1)Unlock the four quick-release fasteners (5) from FCU (6).
(2)Withdraw FCU (6) from housing.
(3)Disconnect electrical connectors (1) and (2).
(4)Cap electrical connectors (1), (2), (3), (4).
(5)Remove FCU (6).

C. Installation
(1)Clean and inspect component adjacent area.
(2)Remove blanking caps from electrical connectors (1), (2), (3) and (4).
(3)Connect electrical connectors (1) and (2) to connectors (3) and (4).
(4)Install FCU (6) in its housing.
(5)Lock the four quick-release fasteners.
(6)Install upper part of glareshield panel.
(7)Remove safety clips and tags and close circuit breakers 306CA2, 312CA1, 312CA2, 306CC1, 306CC2, 1LF, 313CA1 and 149LP.

D. Test
Carry out AFS light test and display test (Ref. 22-00-00, P. Block 501, paragraph 1.B.). Carry out LAND test (Ref. 22-40-00, P. Block 501, paragraph 1.B.), if aircraft is operated in actual AUTO LAND CAT 3 conditions.

E. Close-Up
Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

AP DISCONNECT SWITCH (31CA, 32CA) - REMOVAL/INSTALLATION

For removal of the AP disconnect switch (31CA and 32CA), remove the control wheel (Ref. AMM 27-11-11, P. Block 401).

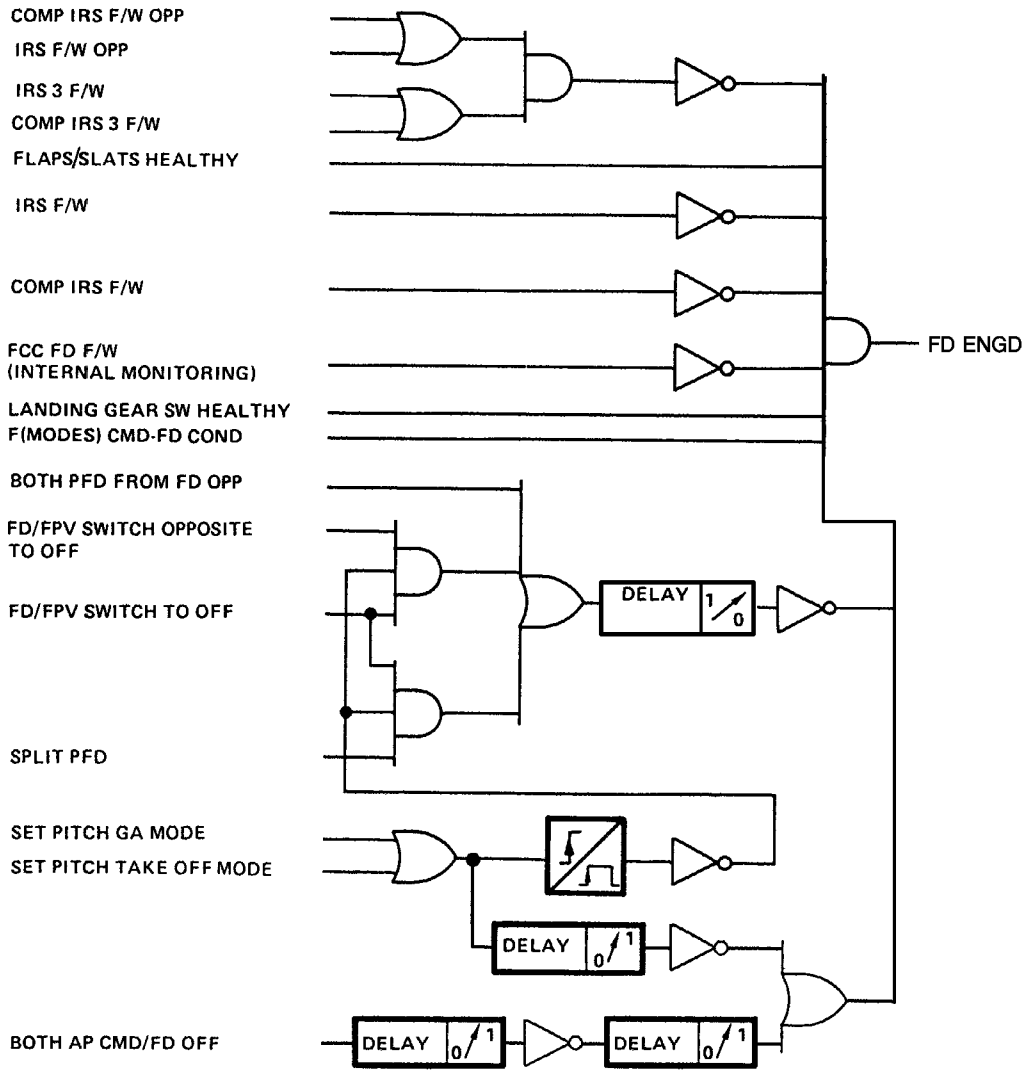
R 1. Flight Director

A. Engagement (Ref. Fig. 002)

The flight director is engaged automatically as soon as the FCCS is energized, provided all the logic conditions are satisfied.
Therefore no engage pushbutton switch is provided.
The initial engagement is made in :

- V/S : vertical speed hold at engagement.
  The V/S counter on the FCU is synchronized with this vertical speed.
  It is then possible to modify the reference value to hold.
- HDG : heading hold at engagement.

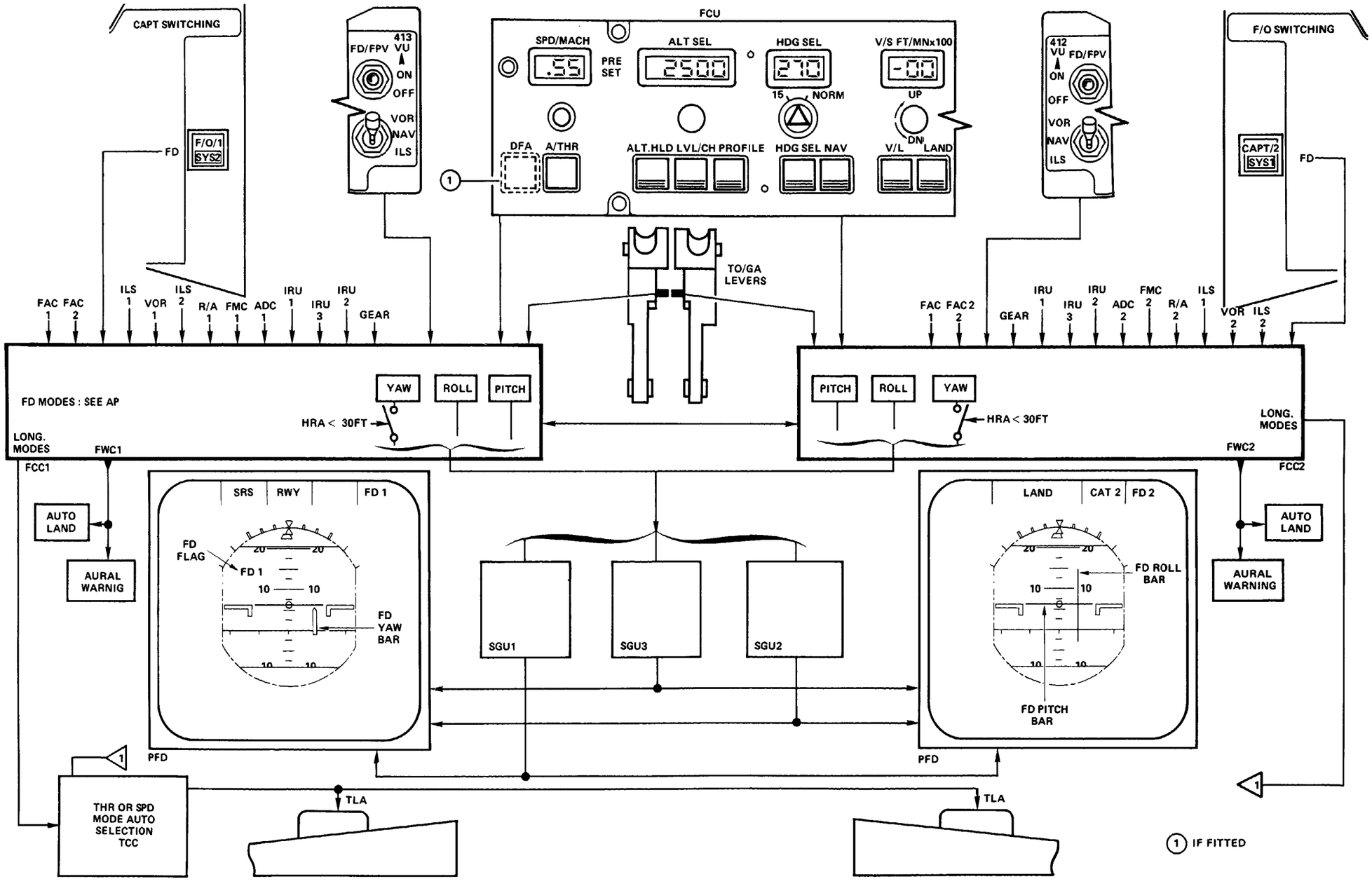
When the V/S and HDG modes are engaged, the FD pitch and roll bars are visible on the PFDs. When the aircraft is on the ground, there is synchronization with the aircraft heading so that the roll bar is always centered.
The FD engage logic depends on conditions common to the AP and FD (Ref. paragraph 2).
The FD bar display on the PFD is given in figure (Ref. Fig. 003).
The yaw bar only appears in the ALIGN and ROLL OUT phases of the LAND modes, if no AP is engaged in CMD and in TAKEOFF mode.
In PITCH TO or PITCH GA mode, the loss of both AP/FDs will not cause a return to the basic modes.
(Ref. Fig. 001)



BM5 22 16 00 0 AAT2 01

FD Engage Logic
Figure 001

R



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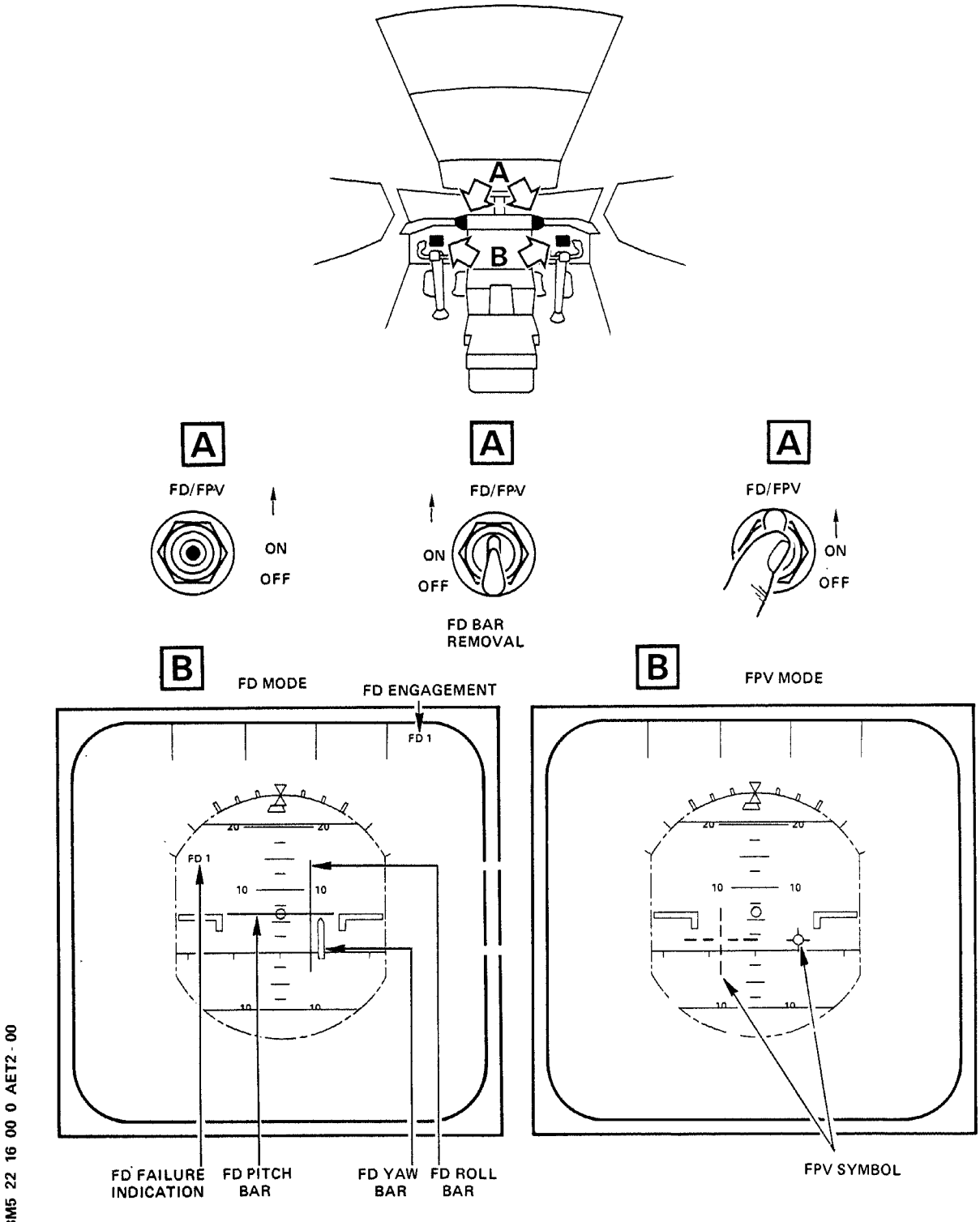
FD - Block Diagram
Figure 002

B. Disengagement
Since there is no pushbutton switch to engage the FD, it is not possible to disengage it directly.
Nevertheless the Captain and the First Officer can clear the bars from the PFDs by means of the FD/FPV selector switches located on the glareshield (see paragraph B).
Both FDs are disengaged when selection of the flight path vector (FPV) is made on the PFDs or when both FD/FPV selector switches are in OFF position. (Ref. Fig. 003)



BM5 22 16 00 0 AET2-00

FD Bar Display and FD/FPV Selector Switches
Figure 003

C. FD Bar Removal

(1)FD bar removal manual control

The FD bars on the PFDs can be cleared by means of two FD/FPV selector switches (one is associated with the Captain PFD, the other with the First Officer PFD).

When the selector switch is moved from OFF to ON position or on energization, priority is given to the FD bar display.

The upper position of the FD/FPV selector switch enables alternate reversion to FPV symbol when FD bars are displayed and to FD bars when FPV symbol is displayed. The selector switch returns to intermediate position.

In addition, when the flight path vector is displayed on the PFD, selection of the following modes :

- TAKEOFF
- GO AROUND
- ALIGN
- ROLL OUT

will cause the FD bars to reappear on the PFD.

Since the position of the selector switch is not representative of the presence of the FD bars, the FD bar removal information is transmitted to the FCCs by the EFIS symbol generator units for capability logic conditions when the FD bars are not cleared.

R FD bar removal associated with FD operation

FD1 and FD2 are independent and the information displayed on the PFDs may be different as long as AP is not engaged in CMD mode.

These differences in commands or mode indications can be due to differences during altitude acquire, in barometric corrections on both altimeters.

When one AP is engaged in CMD mode, a comparison between the modes engaged on both FCCs is performed for each axis. In the event of any discrepancy, the FD bar corresponding to the FD not associated with AP in CMD is cancelled on axis concerned. When the modes of both FCCs are again in agreement, the bars reappear on the PFD.

The logic of the mode comparison and the FD bar removal is given in the following figures.

(Ref. Fig. 004)

(Ref. Fig. 005)

R (2)FD bar flashing

(a)FD pitch bar

The FD pitch bar flashes for 10 seconds :

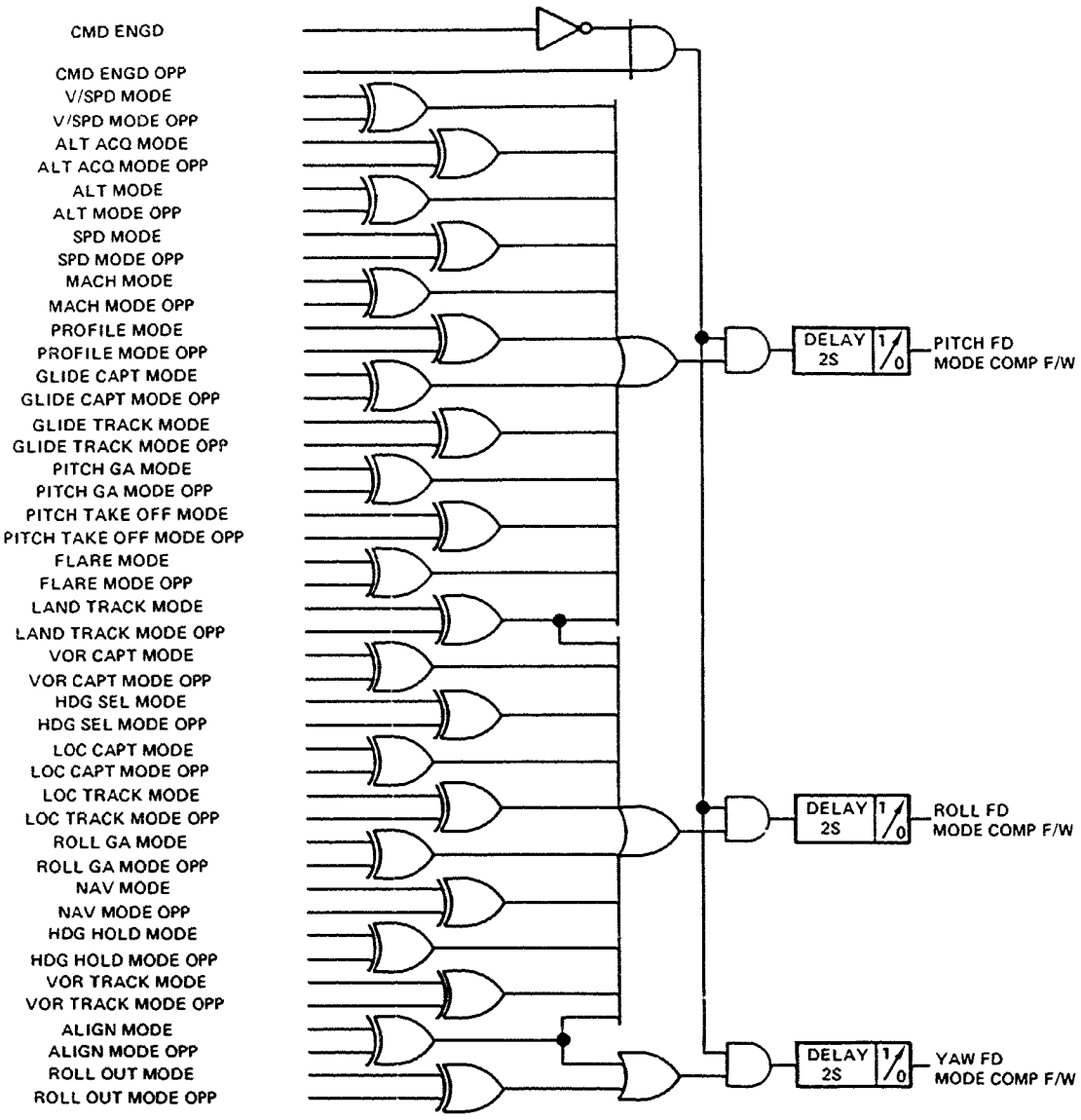
- between glide capture phase and 100 ft. if two GLIDE data are NCD
- in ALT ACQ mode if ALT SET CHANGE is performed when ALT ACQ COND is not present.
- at LAND mode ARM reset in glide capture phase or glide track phase
- at engagement of V/S and HDG HOLD modes.

(b)FD roll bar

The FD roll bar flashes for 10 seconds :

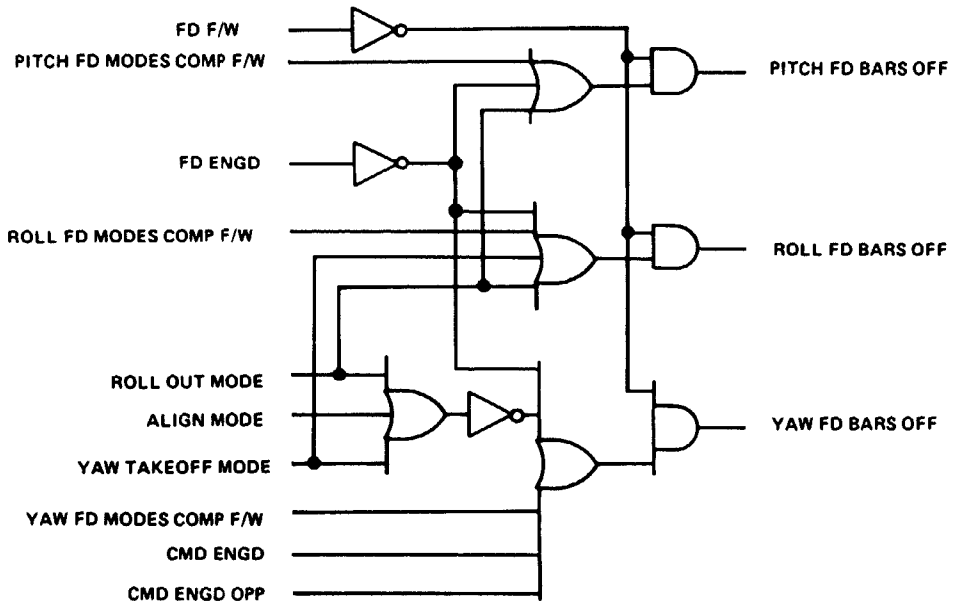
- at LAND mode ARM reset in LOC CPT phase or LOC TRACK phase.
- at engagement of V/S and HDG HOLD modes.

(Ref. Fig. 006)



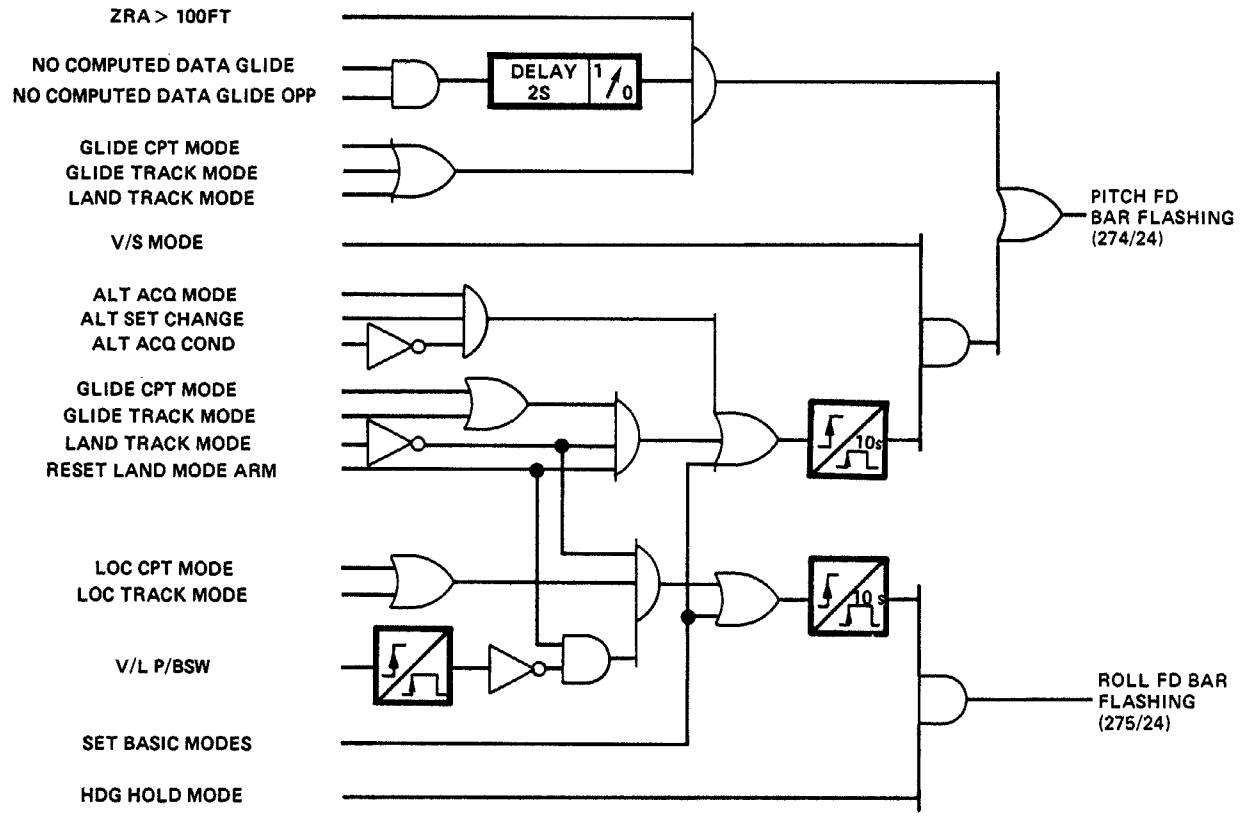
BM5 22 16 00 0 ALP2 12 07
DM5

FD Mode Comparison Logic
Figure 004



BM5 22 16 00 0 AJP2 12

FD Bar Removal Logic
Figure 005



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DM5

FD Bar Flashing - Logic Conditions
Figure 006

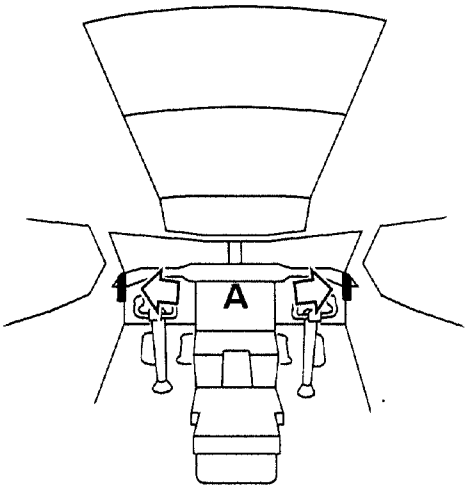
D. FD Bar Switching
Under normal conditions, the Captain PFD shows the FD commands generated by FCC1 and the First Officer PFD shows the FD commands generated by FCC2. In addition, each pilot is given the possibility to display, on his PFD, the FD commands generated by the opposite FCC by means of the FD bars switching pushbutton switches located on the R and L instrument panels (CAPT SWITCHING and F/O SWITCHING panels).

(Ref. Fig. 007)

E. Warning
In the event of FD failure, a red FD failure indication (FD1 or FD2) appears on the PFD (Ref. Fig. 003).

F. FCC-PFD Interface
As the PFDs display the FD1 or FD2 commands, the EFIS symbol generator units 1, 2 and 3 are connected to the FCC1 and 2 output data buses. In addition, the FCC data bus is fed back into the FCC monitoring channel input to monitor the correct operation of the FD commands transmission. If a discrepancy is detected the ARINC transmission of the concerned FCC is interrupted.

G. FD Command Generation
The FD control command is generated from θC and phi C commands computed in the FCC for both the AP and FD functions on the pitch and roll axes. The pitch and roll bar control command takes into account specific feedbacks (Ref. Fig. 008, 009).
The FD yaw control command displayed on the PFD in align and roll out phases is generated from the Sr ROLL OUT and Sr ALIGN commands computed in the FCC for both the AP and the FD (Ref. Fig. 010).



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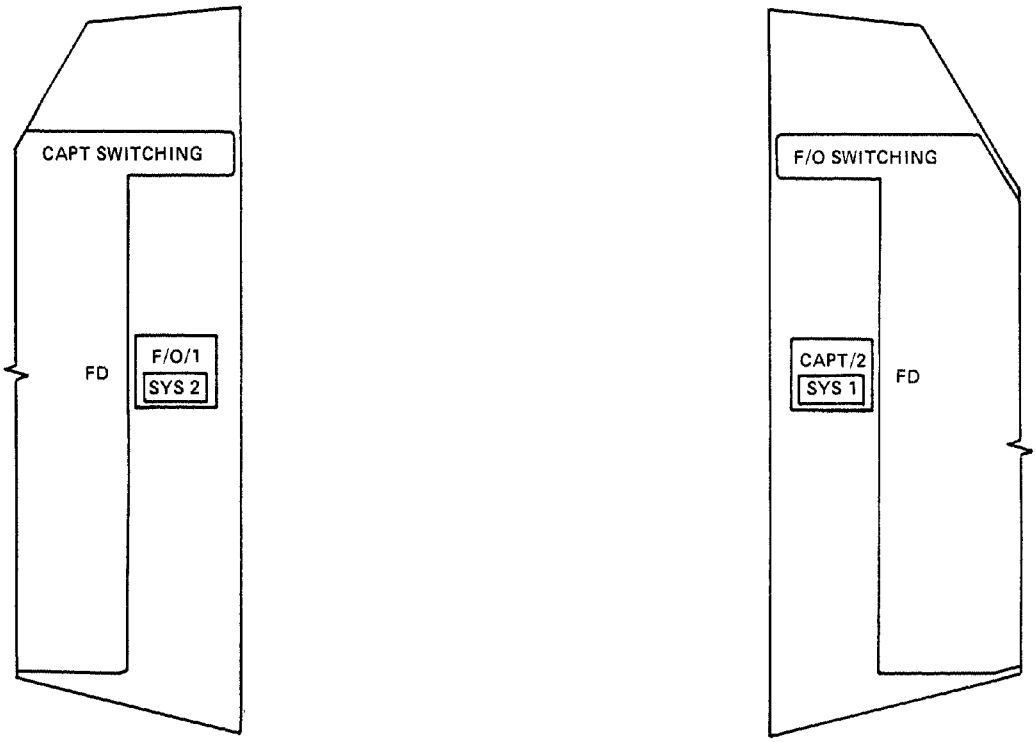

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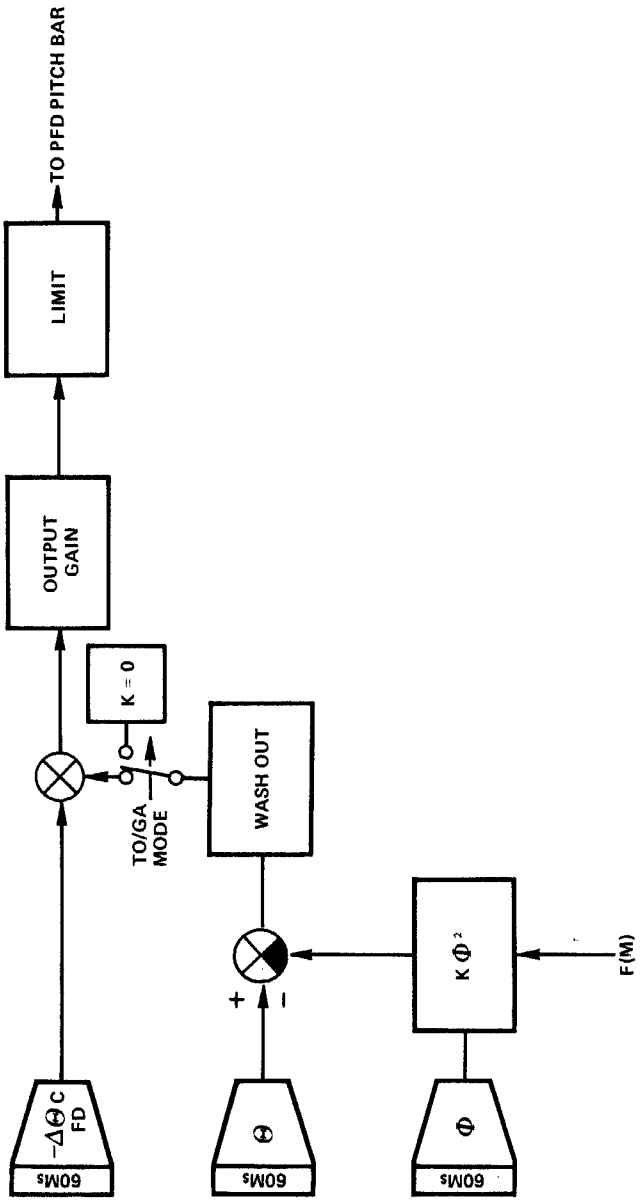
FD Switching - Location
Figure 007



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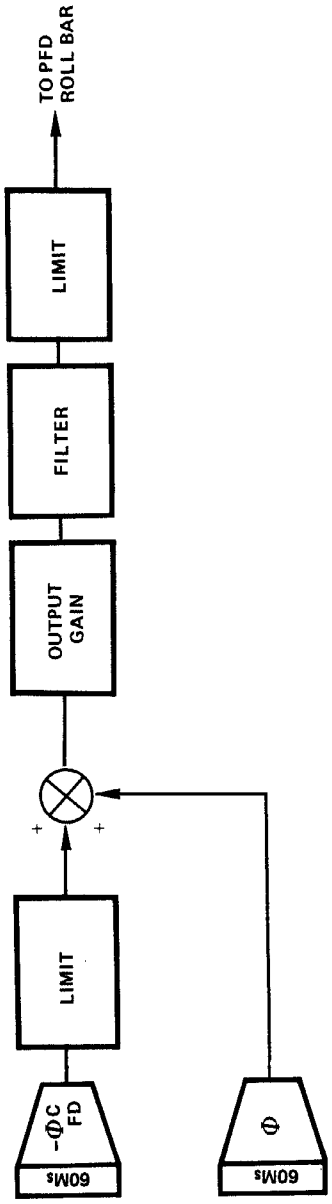
DM5

FD Pitch Basic Loop
Figure 008



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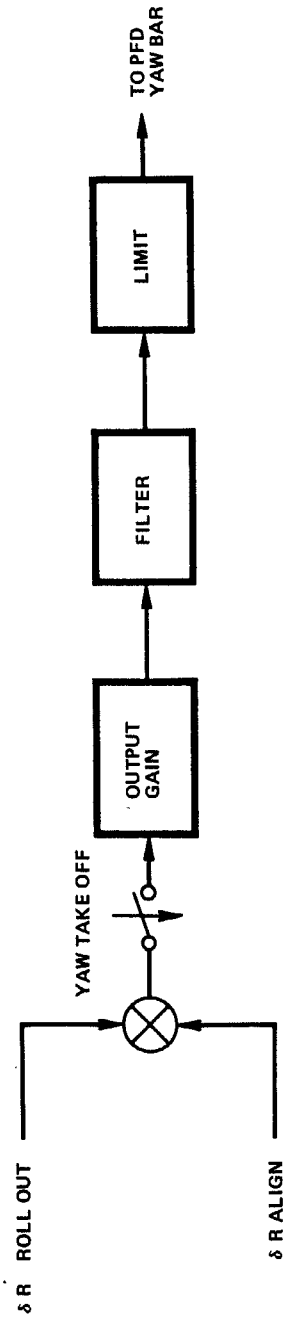
FD Roll Basic Loop
Figure 009



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FD Yaw Loop
Figure 010

2. Autopilot

A. Principles

(1)AP engagement principle

AP is engaged by means of two engage levers (one per side) located on the right part of the FCU.

In cruise only one AP can be engaged at a time (priority is given to the last one engaged).

In approach, both APs can be engaged at the same time after selection of the LAND mode ; both APs remain engaged during all the phases of the LAND mode including go around.

During a go around mode, both APs being engaged, selection of a cruise mode results in disconnection of AP2.

AP engage levers have two positions : ON and OFF.

In flight initial engagement of the AP is always made in CMD mode (selection of the CMD mode is authorized on the ground with engines shut down to enable maintenance tests to be carried out).

The AP engagement on the ground with engines running is made in CWS mode and action on the CWS/CMD pushbutton switch has no effect.

Change over from CWS to CMD and vice versa is made by means of the CWS/CMD pushbutton switch located under the AP engage levers.

The AP engagement in CWS or CMD is indicated by the PFDs and by the CWS/CMD pushbutton switch on the FCU.

Selection of the CMD mode is inhibited on the ground with engines running.

(2)AP engage logic

This figure shows that the AP engage lever is self-held in ON position only if the logic conditions in the command and monitoring channels are met (hard-wired logic).

When an attempt is made to engage the AP and if all the conditions are satisfied, the command logic sends a ground signal to one end of the self-holding relay coil (AP TOGGLE HOLD C) and the monitoring logic closes the contacts of relay 307CA (in the aircraft wiring) which results in sending a 28VDC signal to the other end of the self-holding relay coil.

These logic conditions are : the software safety devices, the specific conditions of AP engagement in CMD and CWS modes (logic CMD conditions and logic CWS conditions), the hardware safety devices (comparators, clutch rwnmonitoring etc.) and the watchdogs.

It is to be noted that the actuator clutching conditions depend on the status of the command and monitoring engage logics and that the clutching line is physically interrupted by a contact depending on the engage lever position (AP TOGGLE SW C).

The AP engage lever self-hold conditions are given in following figure.

(3)Basic loop generation

The generation of the AP basic loops is given in the following figures.

**ON A/C ALL

R

(Ref. Fig. 011)

**ON A/C ALL

(Ref. Fig. 012)
(Ref. Fig. 013)
(Ref. Fig. 014)

**ON A/C ALL

(4)AP disengagement
The AP can be disengaged by :
- the detection of an internal failure
- placing the lever in OFF position
- pressing one of the disconnect pushbutton switches located on the control column handwheels.

AP voluntary disengagement does not cancel the modes which remain available for the FD.
It is to be noted that the AP override devices (torque limiter levers mounted on the actuator output shafts) should not be used as a normal way to regain the A/C control. When AP pitch actuator is overriden by more than 4 degrees, the AP disconnects.

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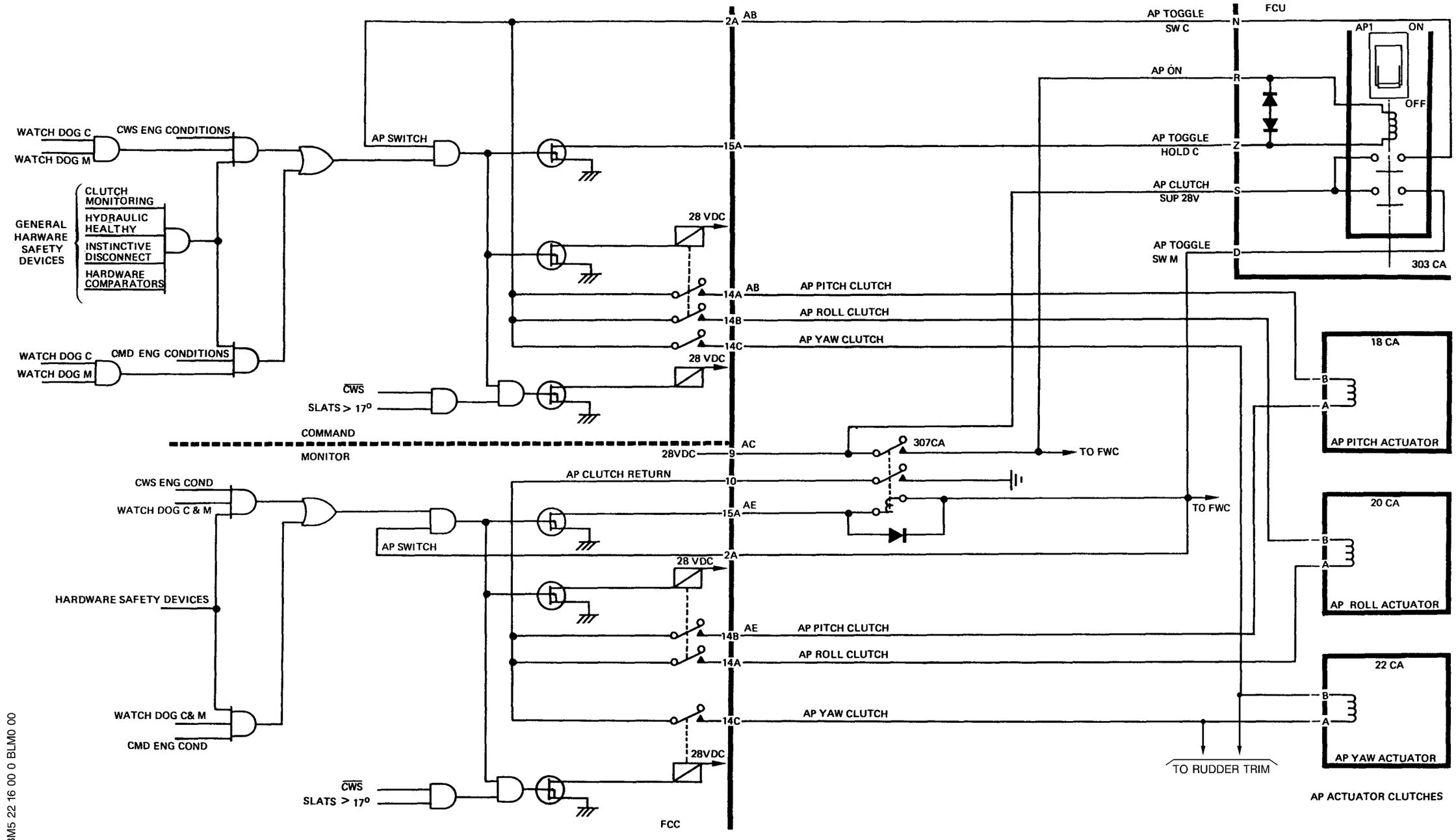
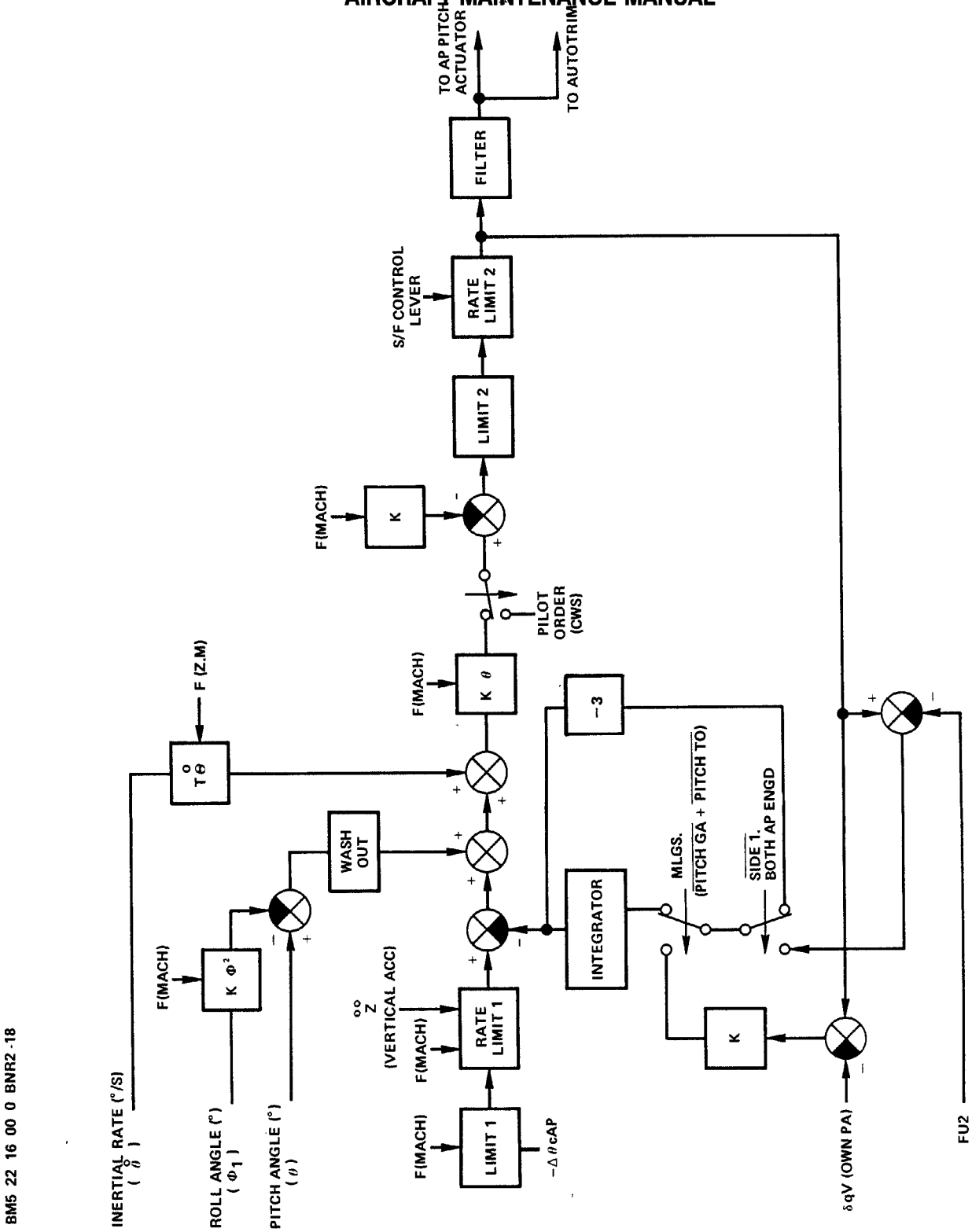
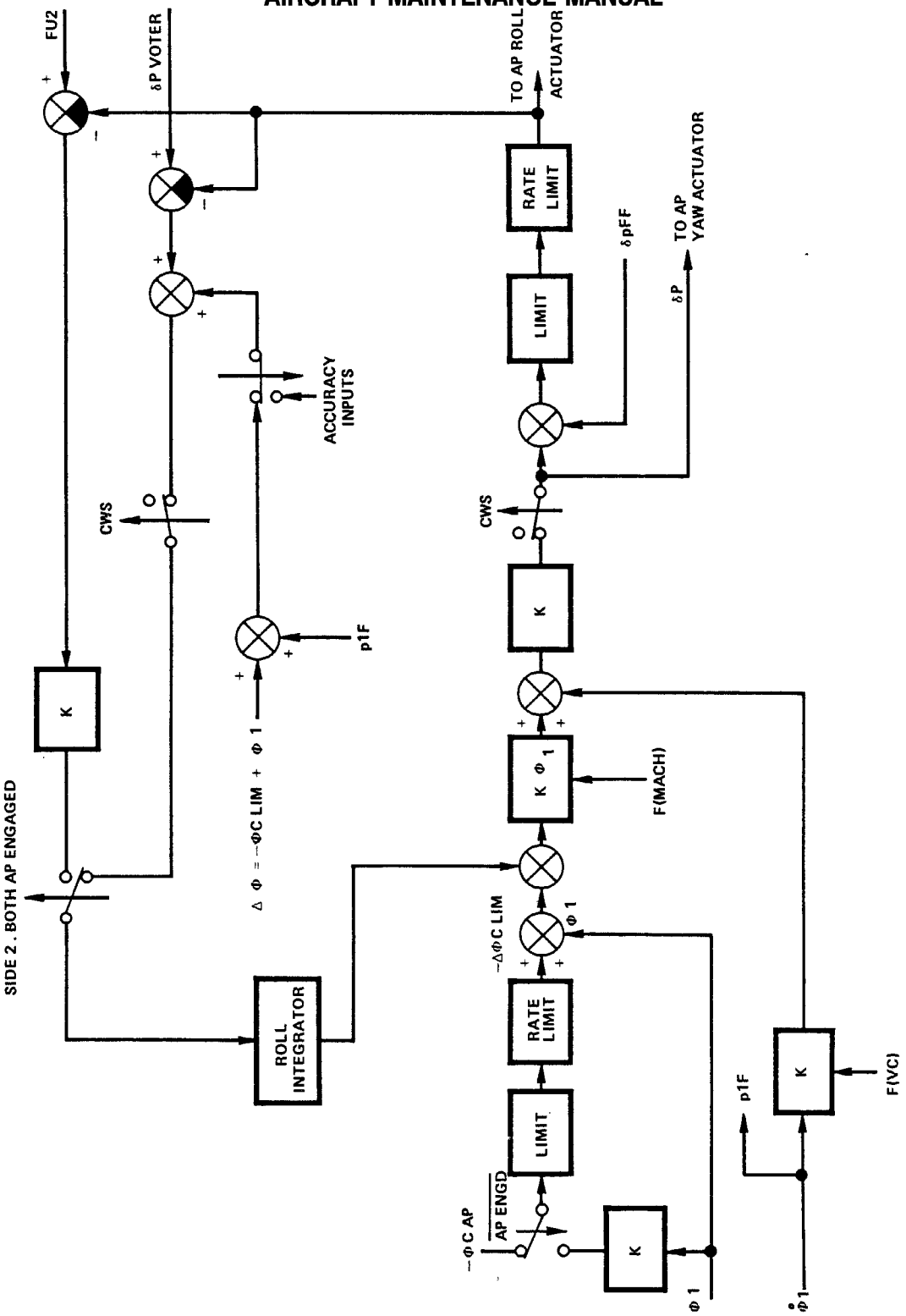


AP Engage Logic
Figure 011

BM5 22 16 00 0 BLM0 00



Pitch AP Basic Demand Loop
Figure 012



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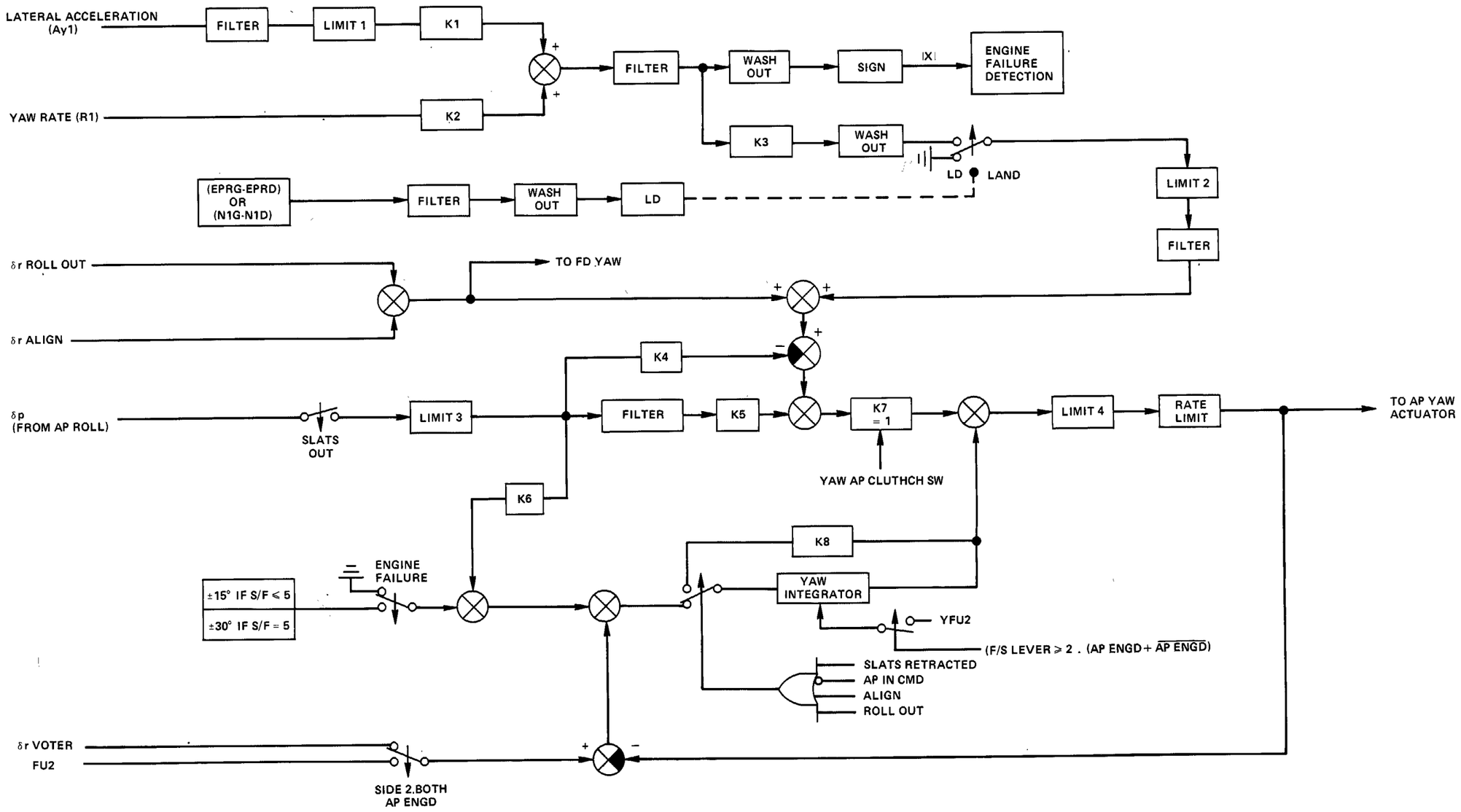
Roll AP Basic Demand Loop
Figure 013



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Yaw AP Basic Demand Loop
Figure 014

**ON A/C ALL

B. AP Engagement in CWS Mode

(1)Principle and use (Ref. Fig. 015)

When the AP is engaged in CWS mode, it enables to hold the longitudinal θ and lateral Phi attitudes which were the aircraft attitudes at the engagement of the function. It is to be noted that the CWS function is not active on the yaw axis.

The control wheel steering (CWS) enables basic parameters to be modified at any moment by operating the control column and/or the control wheel. As soon as the pilot releases his input on the control column, the CWS holds the new values obtained (Ref. Fig. 016).

If the CWS function is selected on the ground, attitude hold modes are no longer available but laws giving control surface deflections approximately proportional to the loads applied by the pilot are obtained.

In fact, the pilot controls the aircraft through the AP taking advantage of the stabilization and limitation computations specific to the CWS.

The use of the CWS mode is possible on takeoff and on landing.

When the AP is engaged in CWS mode, the autotrim function is inhibited on the ground and becomes active 5 seconds after lift-off

Action on the pitch trim control switches (electric trim function) is effective in flight only when the pilot applies a load on the control column (the electric trim function is active on the ground whatever the load applied).

The CWS function is not available when the AP is engaged in CMD mode but in VOR mode and in LOC and LAND mode capture phases (LOC CAPT and G/S CAPT) the supervisory override function is available.

In these phases, the pilot, by applying his load on the flight controls (control column, control wheel), operates a control surface deflection approximately proportional to the load applied. When the pilot releases his load, the AP will guide the aircraft along the flight path corresponding to the mode engaged.

NOTE : When the AP is engaged in CWS mode, the upper modes are still available for the FDs so that the pilot can follow the FD commands when using the CWS function.

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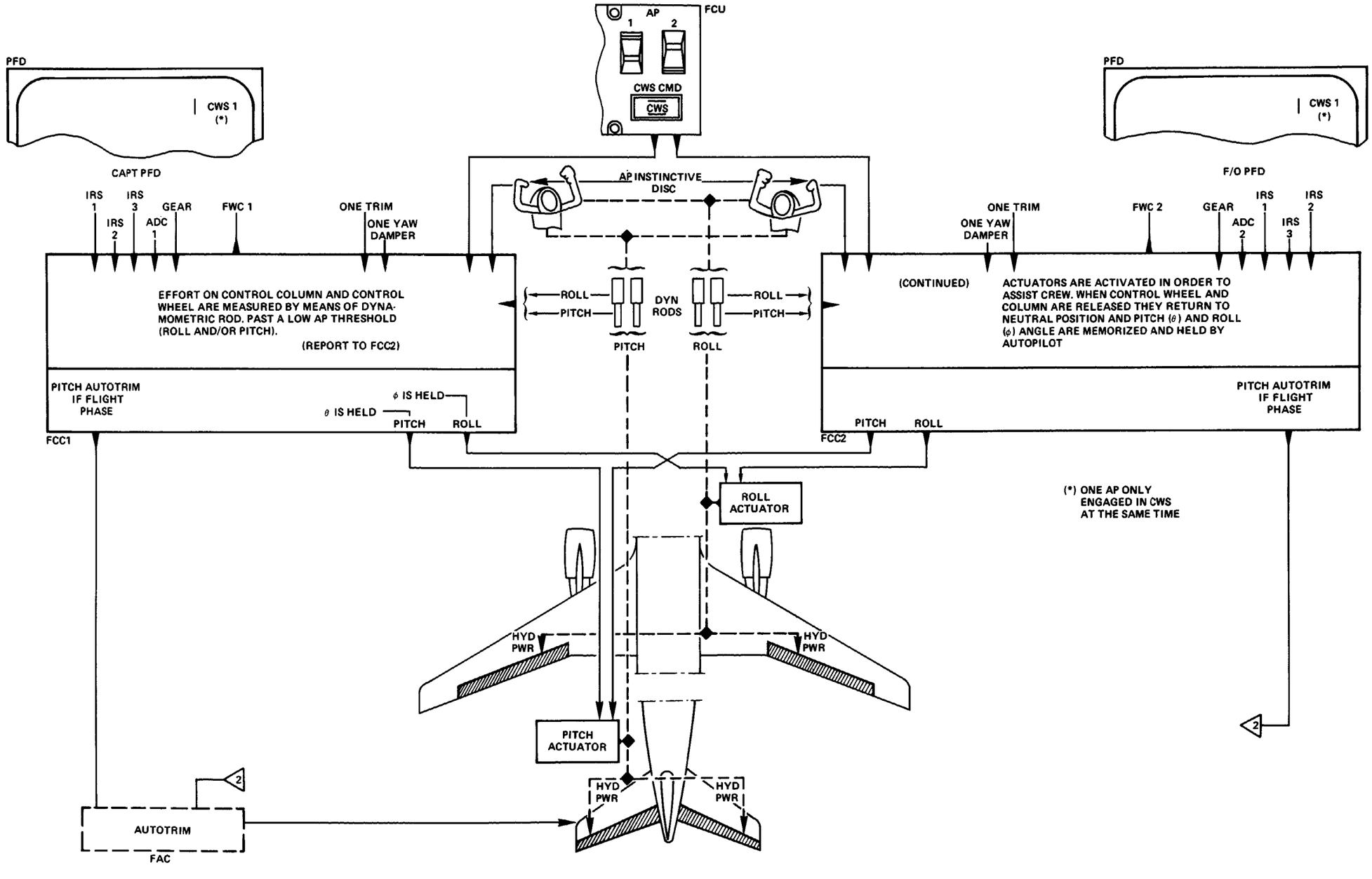

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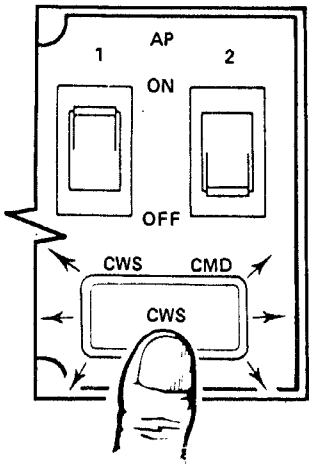
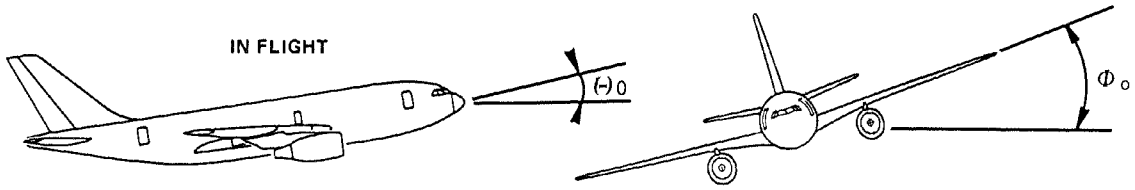
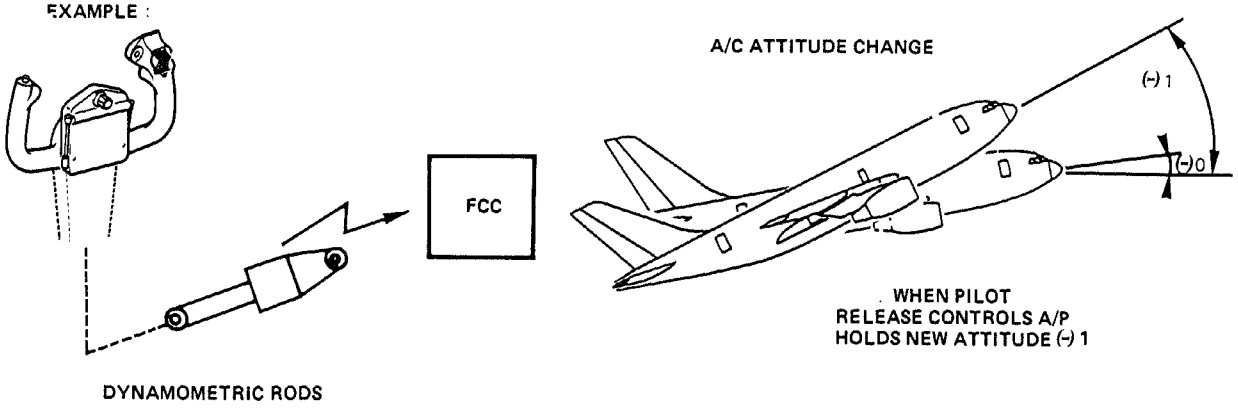
AP CWS Principle
Figure 015



PRESS CWS/CMD
P/BSW



- (-)0 AND Φ0 ACTUAL A/C PITCH AND ROLL ATTITUDES WILL BE HELD
- CWS FUNCTION ENABLES THE PILOT TO CHANGE THESE REFERENCE VALUES AS REQUIRED



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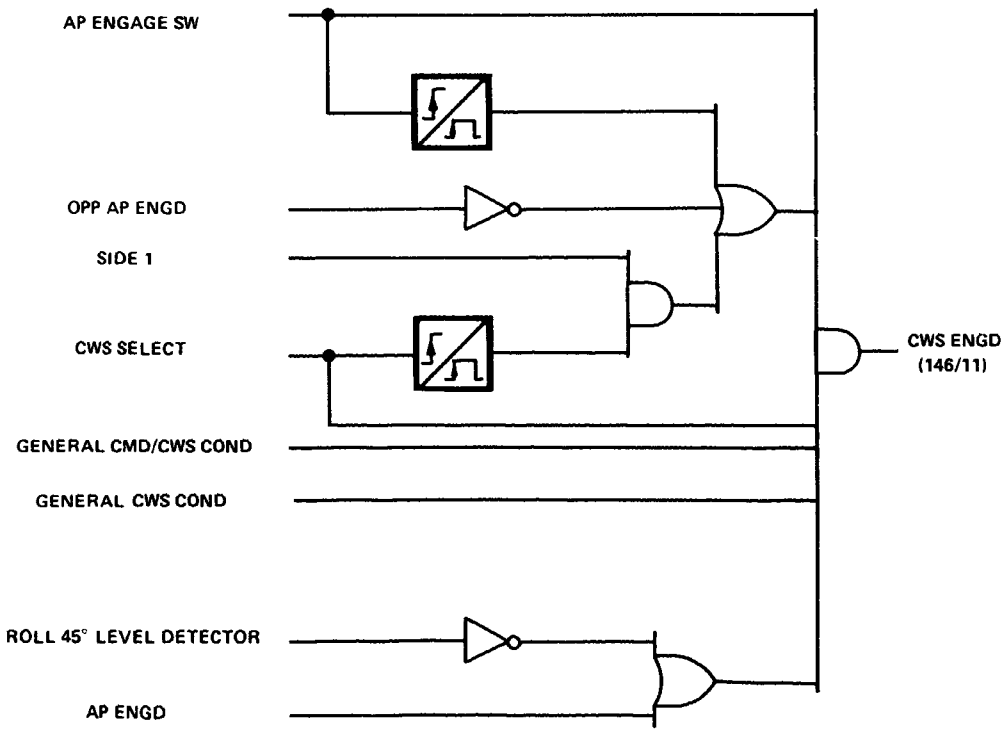
DM5

Use of AP in CWS
Figure 016

(2)Engage logic
The **AP** engage lever self-hold conditions depend on the **AP** engagement in **CMD** or **CWS** mode.
The conditions related to the engagement of the **CWS** function are given in figures (Ref. Fig. 017, 018)
The **CWS** engage logic condition acts directly on the engage lever self-hold when the **AP** is engaged in **CWS** mode.
The **CWS** selection logic condition is achieved during initial engagement of **AP** on the ground or if **AP** is engaged in **CMD** pressing the **CWS/CMD** pushbutton switch (this action is not taken into account in **LAND TRACK** mode if both **AP**s are engaged).

(3)Command generation
The generation of the pitch and roll **CWS** commands is shown in figures (Ref. Fig. 019, 020).
The commanded deflections are proportional to the loads applied on the control column and control wheel, to the load factor limitations and in bank attitude to ensure safety in **CWS**.



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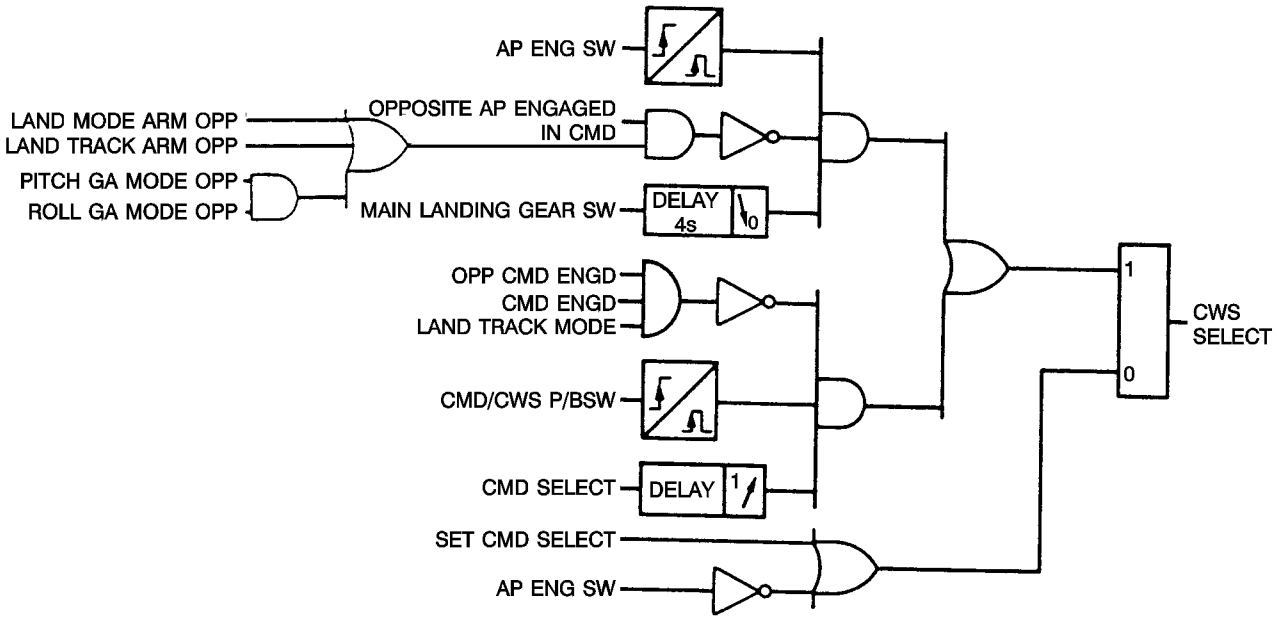
CWS Engage Logic
Figure 017



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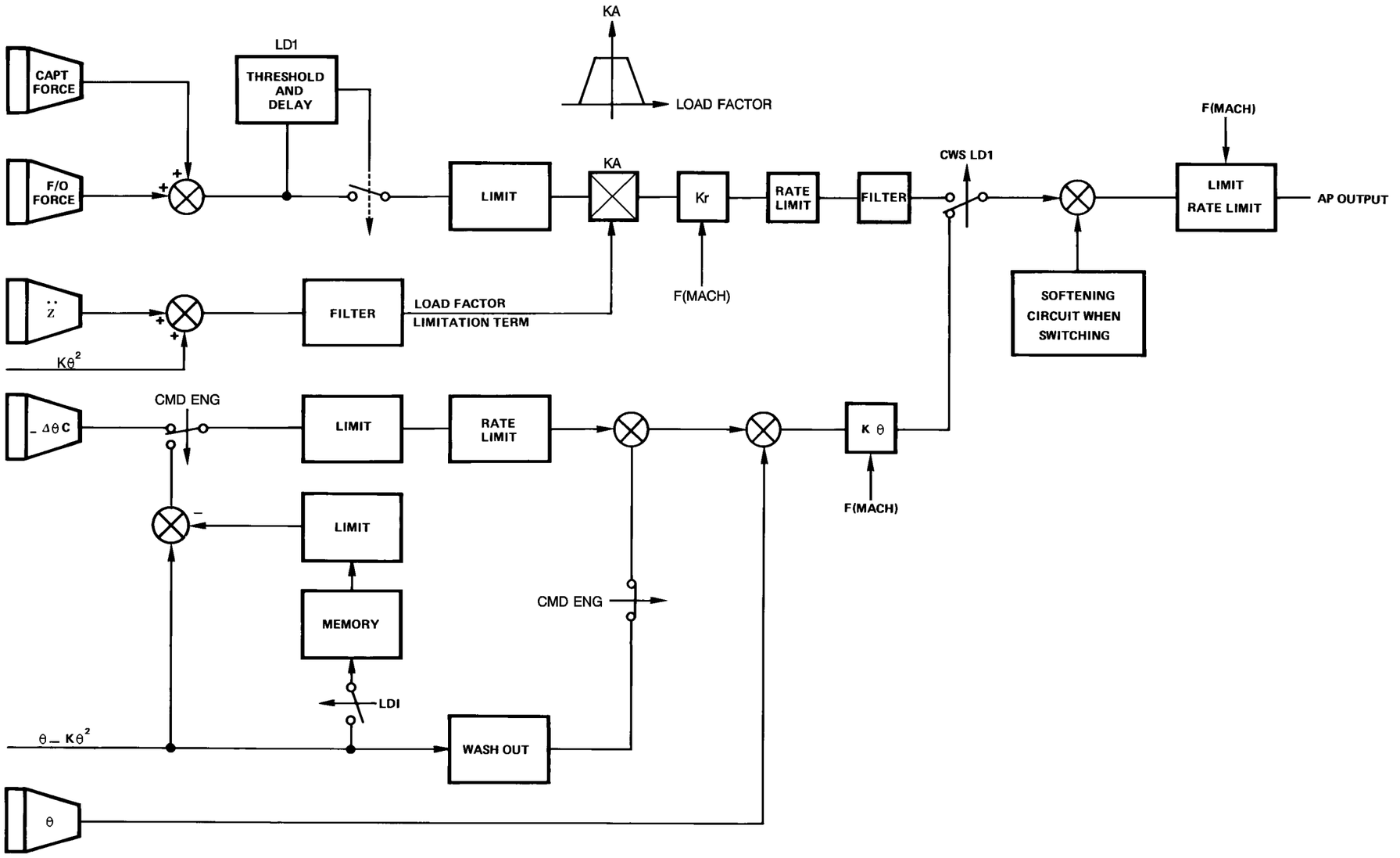
CWS Select Logic
Figure 018



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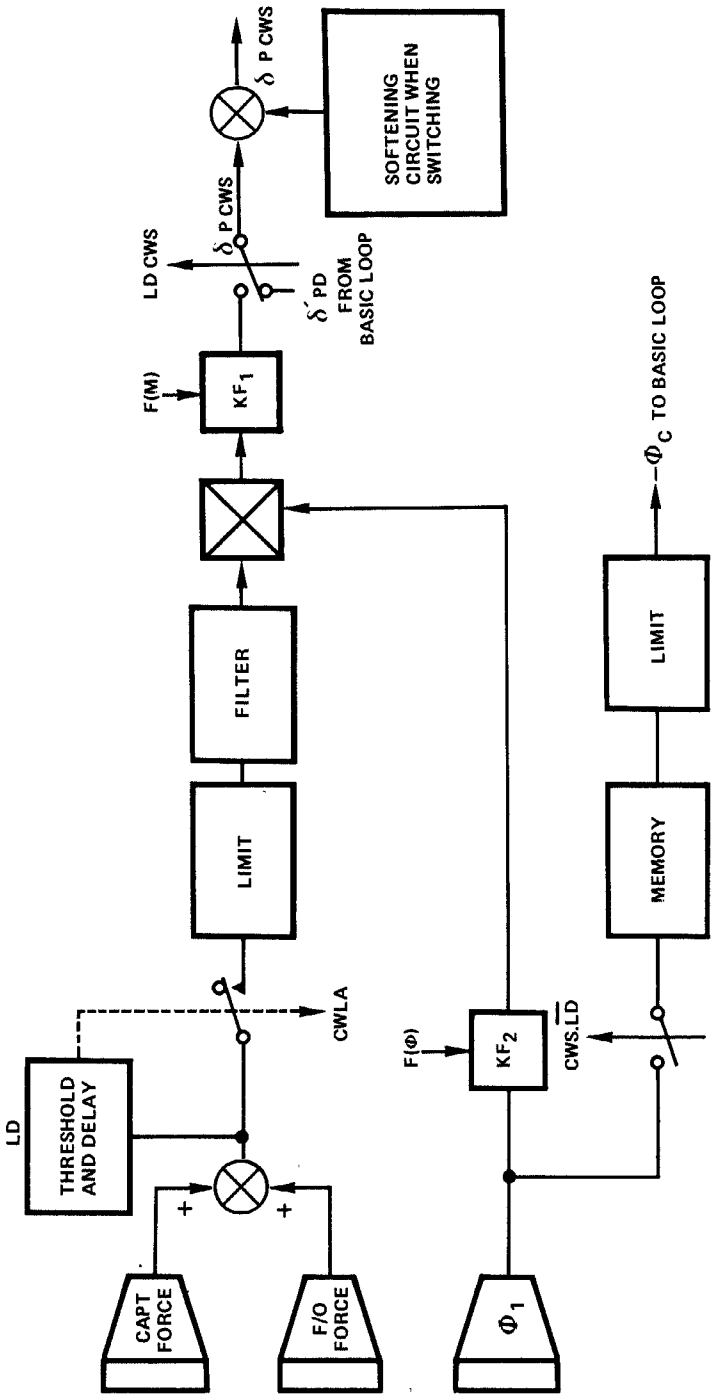
AP Pitch CWS Computation
Figure 019



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AP Roll CWS Computation
Figure 020

C. AP Engagement in CMD Mode (Basic Modes)

(1)Engage Logic

The CMD SELECT condition is achieved if :

- in flight, the AP engage lever is placed in ON position.
- at one AP engagement, the opposite AP is already engaged in CMD in LAND or GO AROUND mode
- AP being engaged in CWS, the CWS/CMD pushbutton switch is pressed (AP engagement in CMD is not possible on the ground and with engines running.

The CMD engage logic condition acts directly on the engage lever self-hold when AP is engaged in CMD mode.

AP engagement in CMD is not possible if bank angle is greater than 45°.

The logic is such that :

- AP2 disengages when LAND or GO AROUND mode is released
- AP disconnects if the GO AROUND mode is selected on the ground.

The CMD engage logic is dependent on logic conditions.

The general CMD/CWS conditions take into account the logic conditions required for engagement in CMD or CWS mode (hydraulic pressure correct, pitch trim and yaw damper engaged etc.).

The F (modes) conditions take into account the logic conditions required for AP/FD engagement in a specific mode (e.g. VOR receiver healthy signal when VOR mode is engaged).

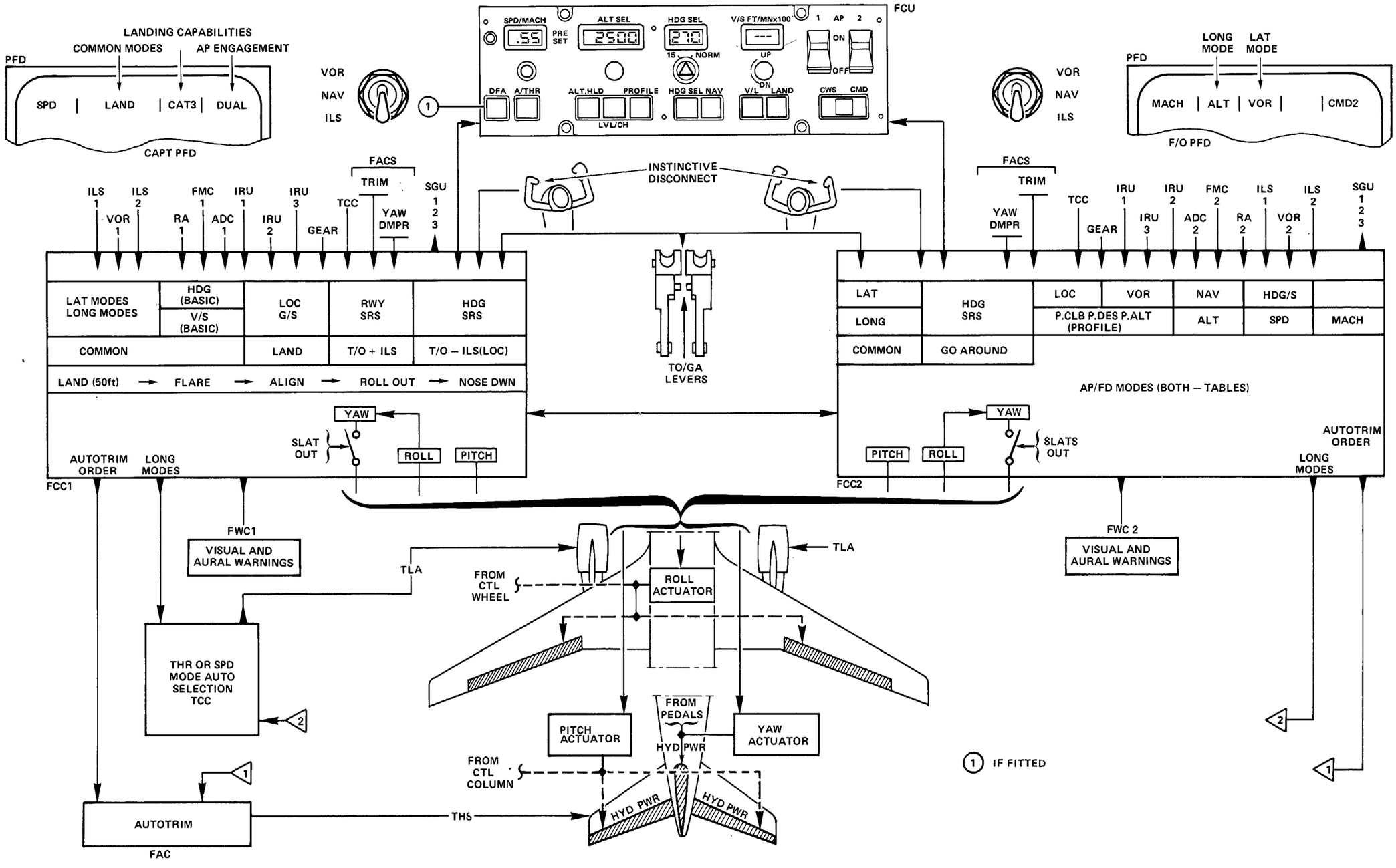
(Ref. Fig. 021)
(Ref. Fig. 022)
(Ref. Fig. 023)
(Ref. Fig. 024)

**ON A/C ALL

R

(Ref. Fig. 025)

**ON A/C ALL



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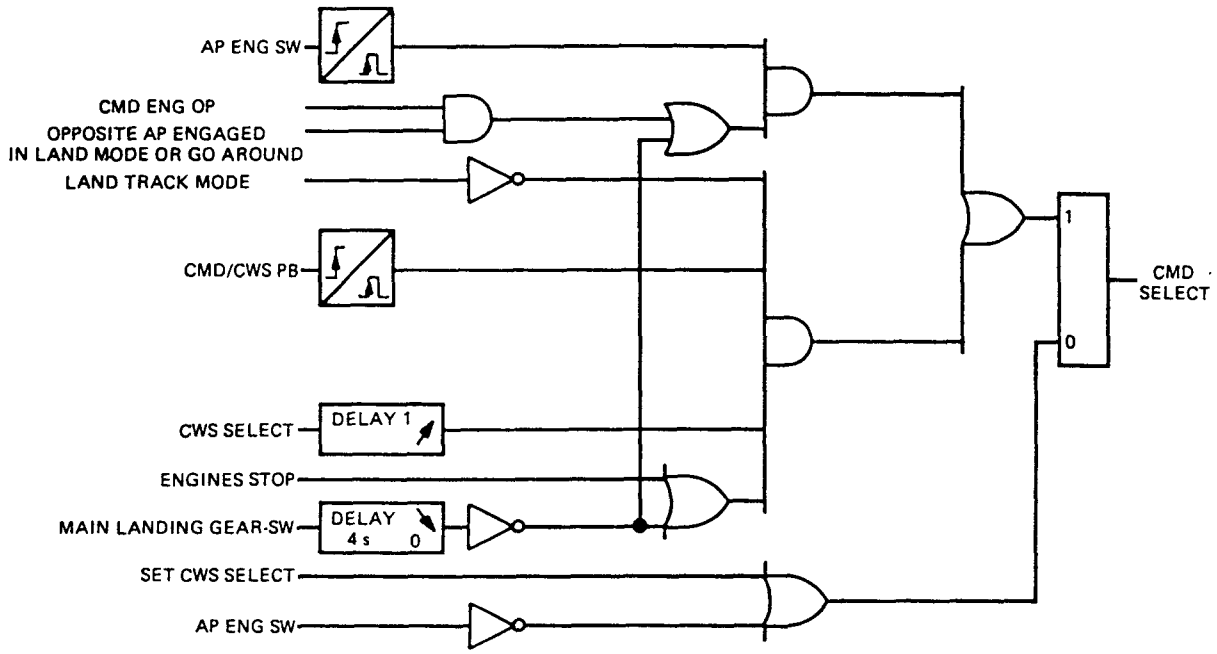
AP CMD Principle
Figure 021



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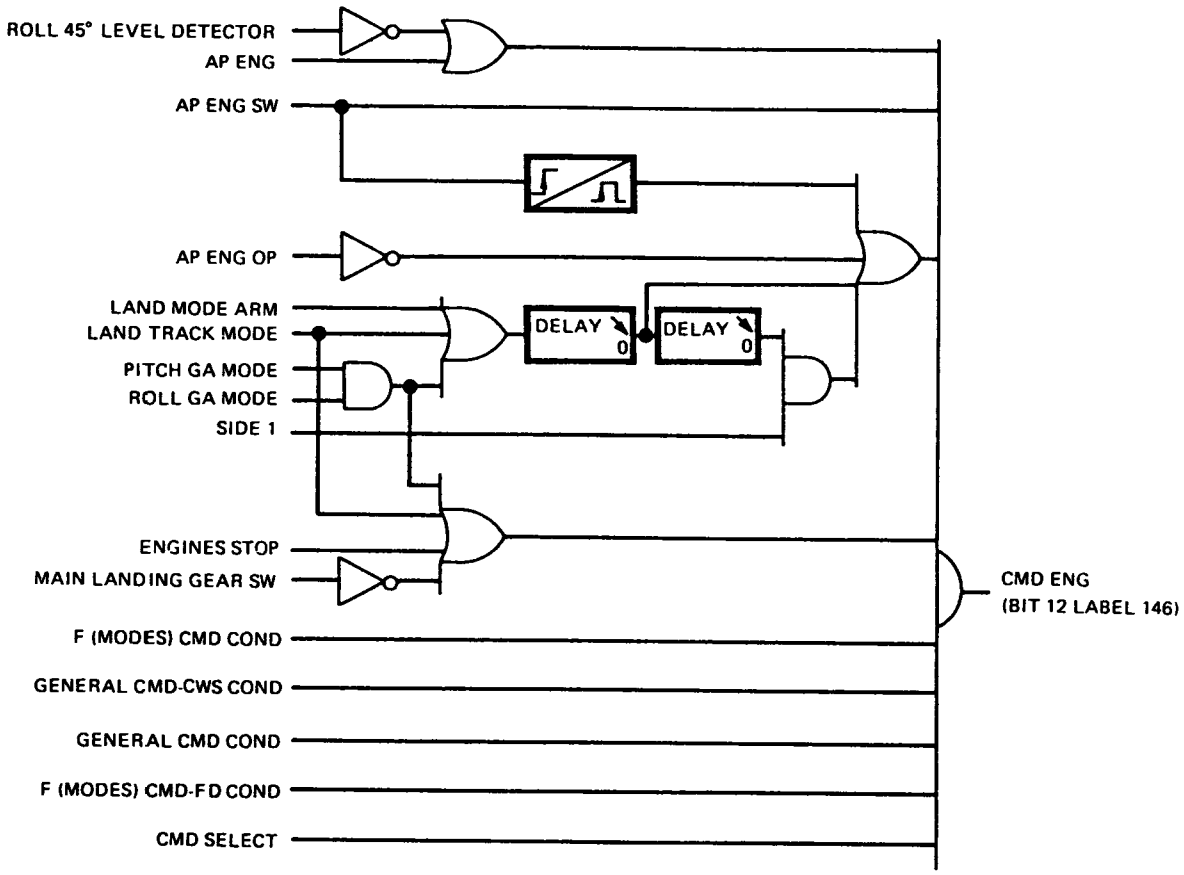
CMD Select Logic
Figure 022



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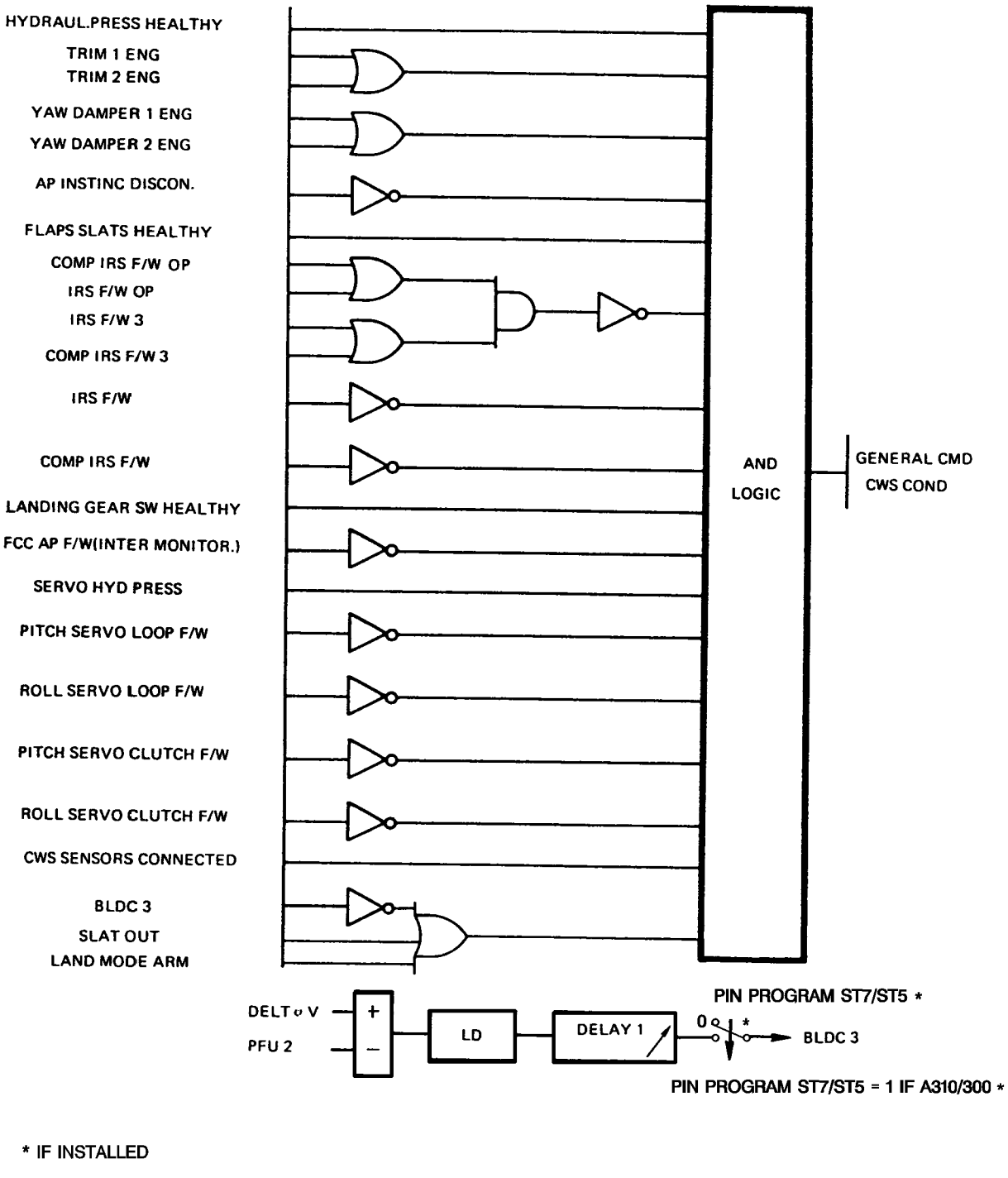
CMD Engage Logic
Figure 023



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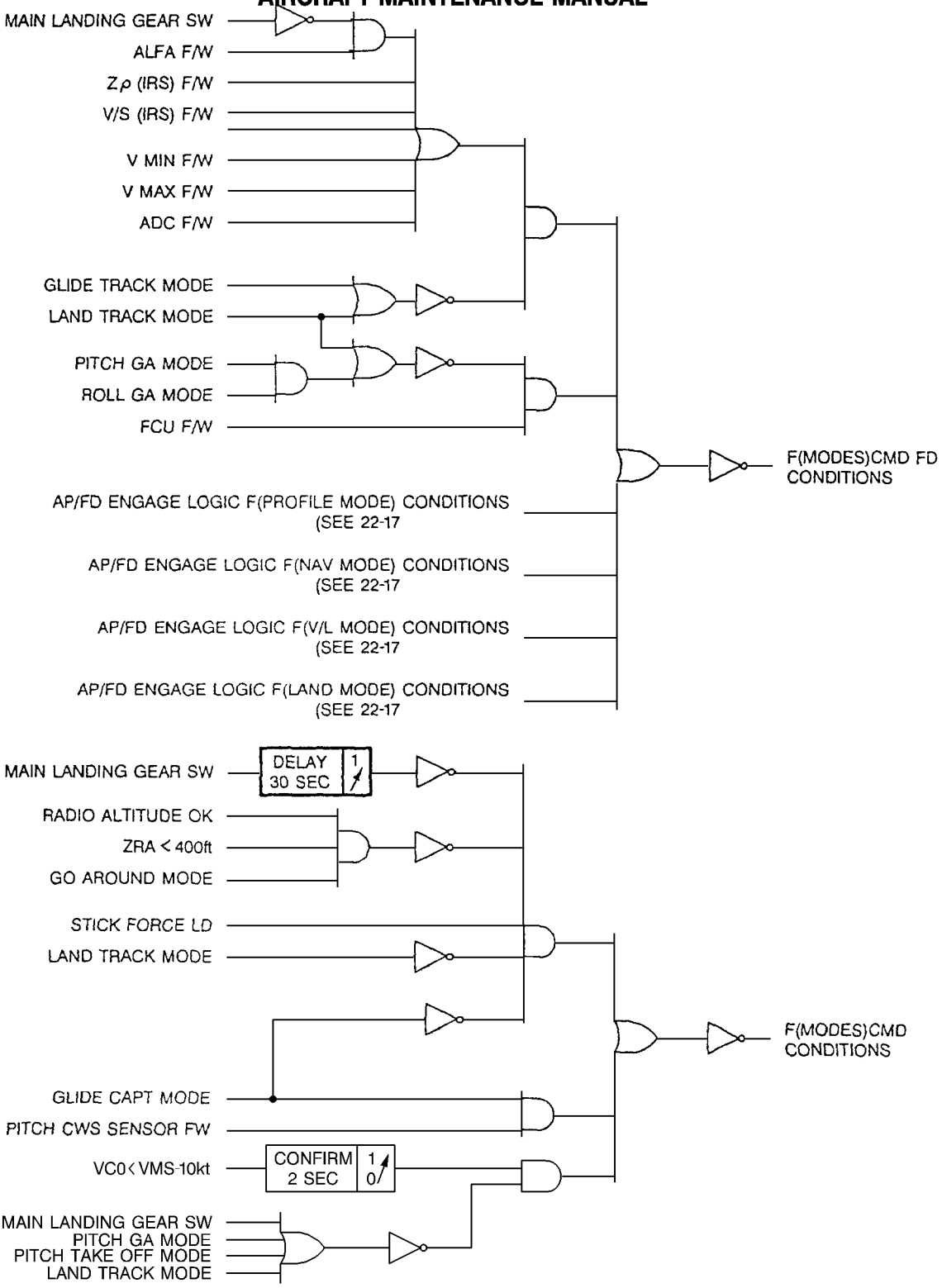
* IF INSTALLED

AP Engagement - General CMD/CWS Conditions
Figure 024



AP Engage Logic F (MODES) Conditions
Figure 025

(2)Engagement of basic modes

The AP (and FD) basic modes are :

- V/S : vertical speed hold at engagement
- HDG : heading hold at engagement

The AP engagement in CMD is made in the modes selected for the FD.

The AP engagement in basic modes assumes that these modes were active for the FD or that both FDs were disengaged (both FD/FPV selector switches in OFF position or flight path vector selected on both PFDs).

The AP is engaged in CMD mode by pressing the CWS/CMD pushbutton switch on the FCU (AP being previously engaged in CWS mode). The engagement in CMD mode is indicated on :

- the FCU : green CMD legend on the CWS/CMD pushbutton switch comes on
- the PFDs : CMD1 or CMD2 message comes on white depending on AP engaged (AP1 or AP2).

The engagement in V/S and HDG modes is indicated by green messages on the PFDs. The engagement in V/S mode causes synchronization of the V/S counter on the FCU with the aircraft vertical speed at the time of engagement.

Then the pilot can select, on the FCU, a new vertical speed value which becomes the new reference to be held.

(3)Return to basic mode (with AP and/or FD engaged)

When an upper mode is engaged, the light in the corresponding pushbutton switch on the FCU comes on.

It is possible to return to a basic mode by pressing the illuminated pushbutton switch (second action on a mode selection pushbutton switch).

In addition :

- in LAND mode, selection of another upper mode on one axis (longitudinal or lateral) causes return to basic mode for the other axis if LAND mode was active on the first axis.
- in ALT ACQ mode, if the capture conditions are no longer met after a new altitude value has been selected, the AP (or FD) returns to V/S mode.

Return to V/S mode is also possible through operation of the vertical speed selector knob on the FCU.

When pulling the selector knob, the V/S counter synchronizes with the aircraft parameter which will be used as a reference.

It is possible to preset a vertical speed value by first turning the selector knob which causes synchronization of the counter with the aircraft vertical speed. This value can then be modified.

The following figures show the different means of returning to basic modes (V/S and HDG modes) (Ref. Fig. 026, 027, 028).

The basic modes are engaged on both axes :

- on energization
- when the associated FD is not engaged and when AP is engaged in CMD mode (except if the opposite AP is engaged in LAND or GO AROUND mode).

(Ref. Fig. 029)

(a)V/S mode

The V/S mode is engaged :

- by pulling the vertical speed selector knob (action inhibited in LAND TRACK)

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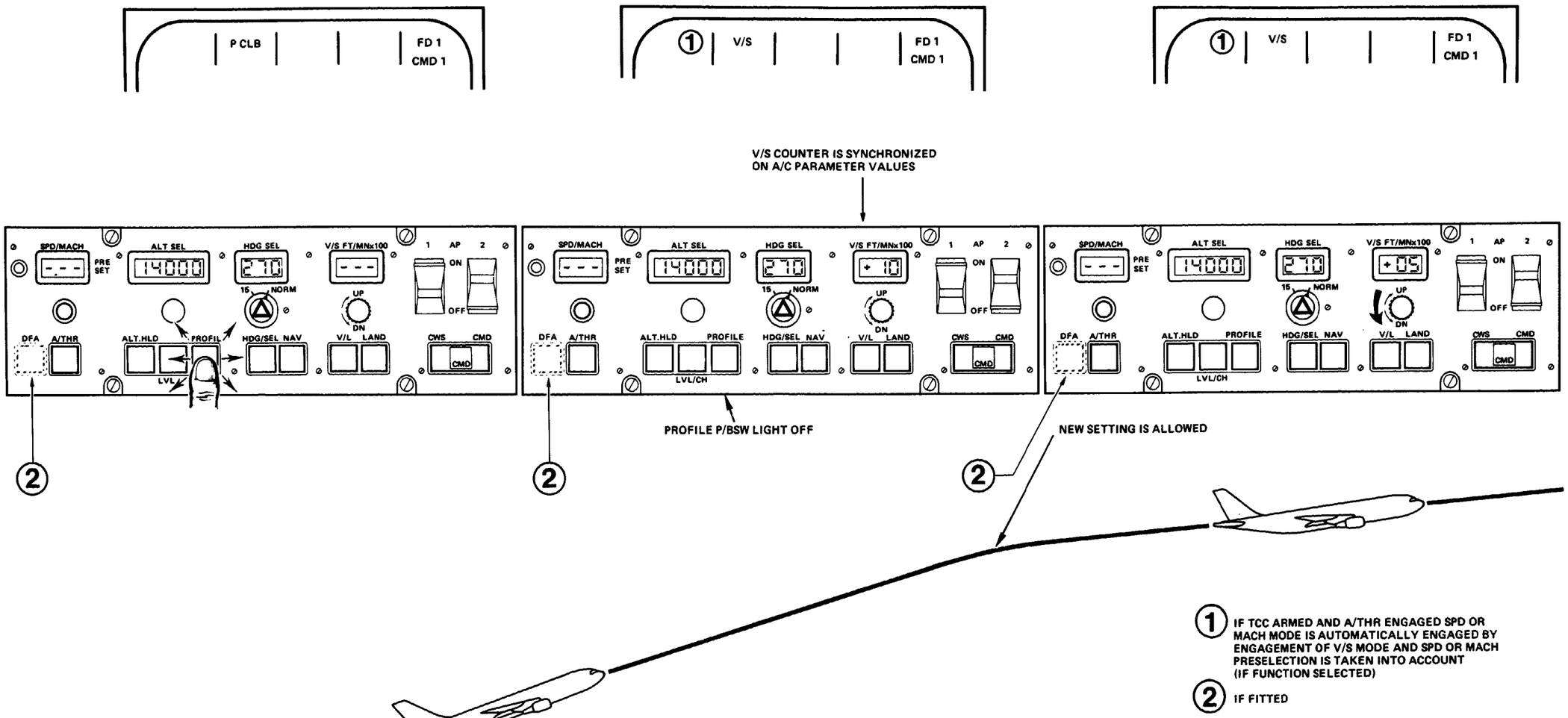

① IF TCC ARMED AND A/THR ENGAGED SPD OR MACH MODE IS AUTOMATICALLY ENGAGED BY ENGAGEMENT OF V/S MODE AND SPD OR MACH PRESELECTION IS TAKEN INTO ACCOUNT (IF FUNCTION SELECTED)

② IF FITTED

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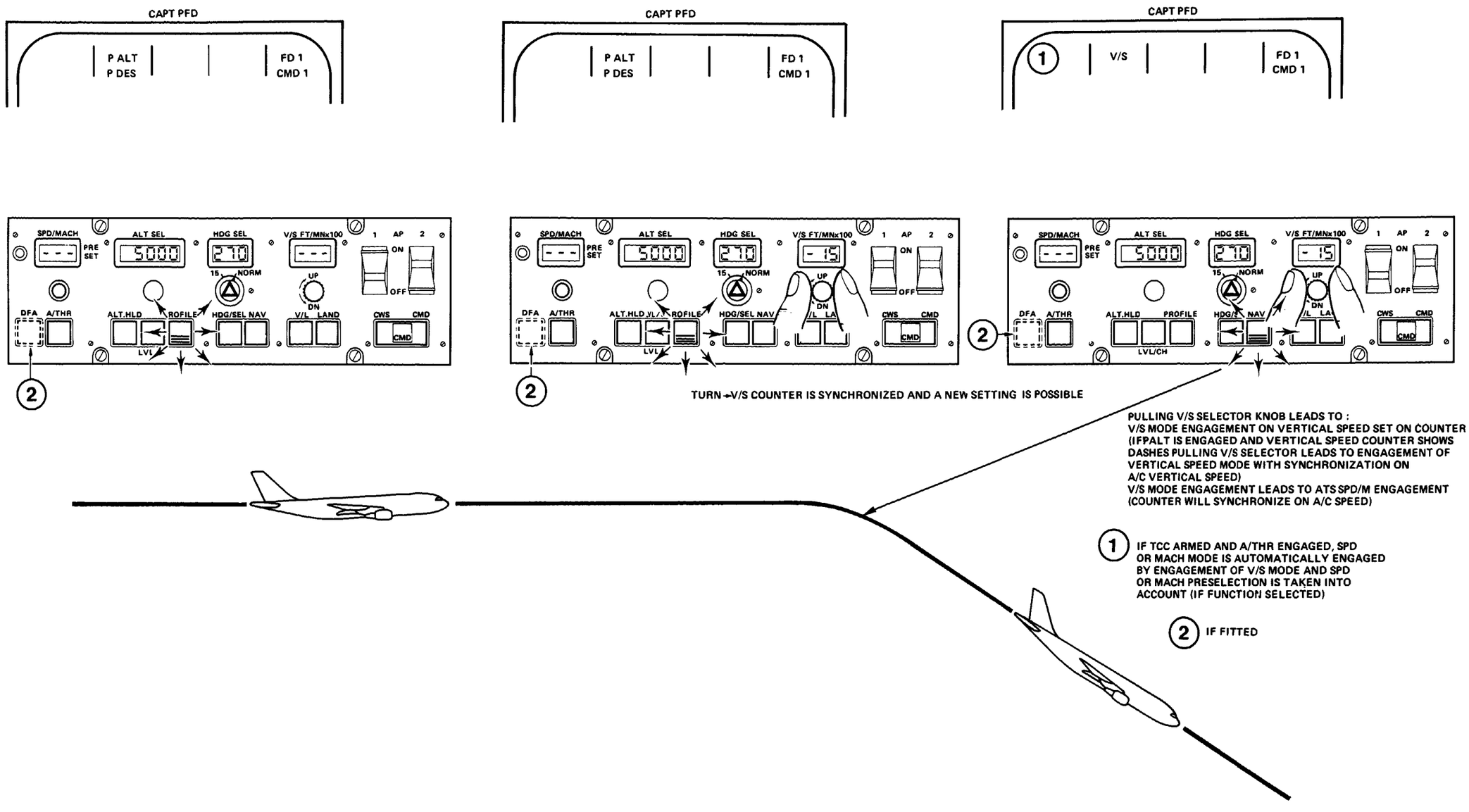
V/S Engagement by Pressing Illuminated Upper Mode P/B switch
Figure 026



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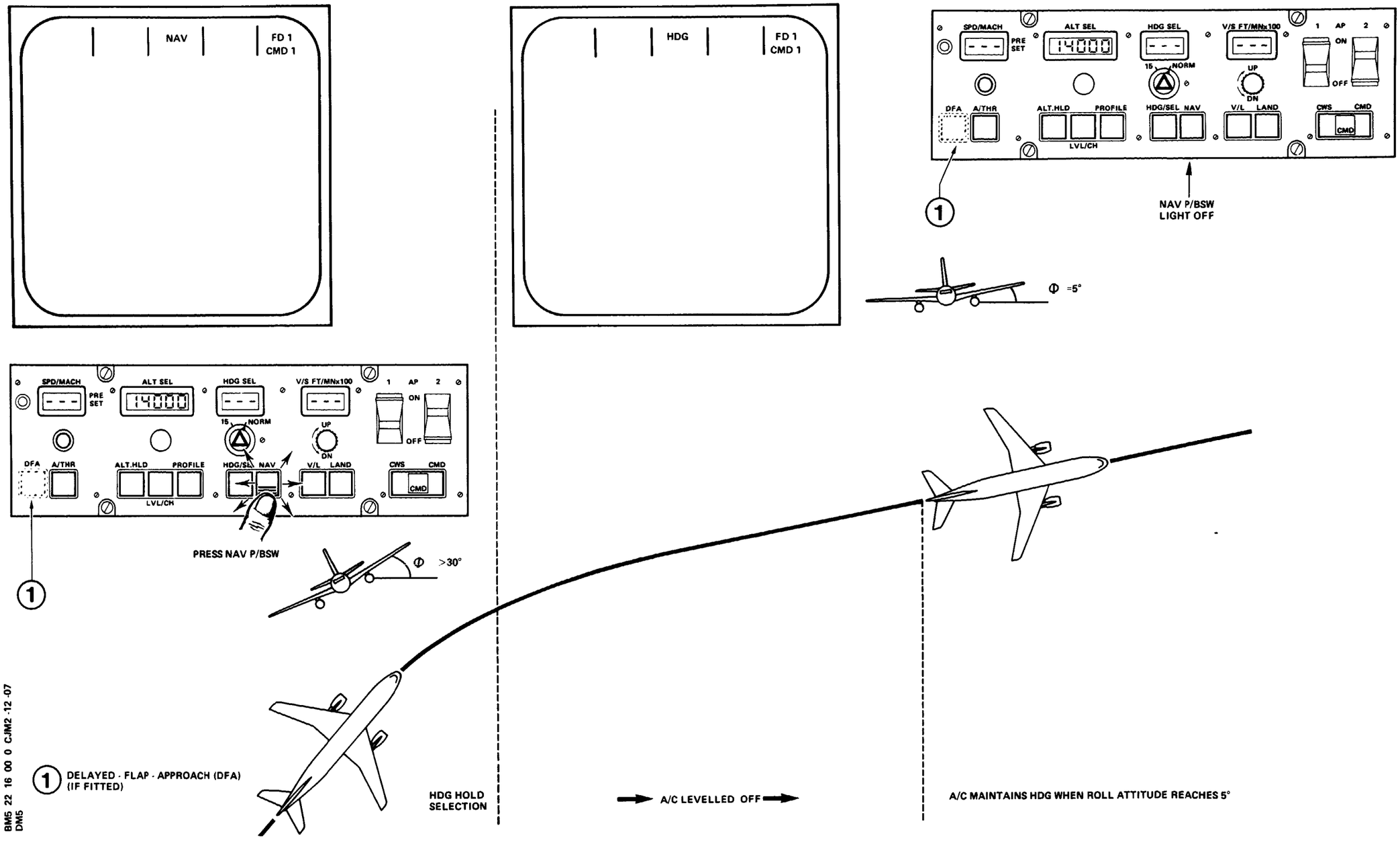
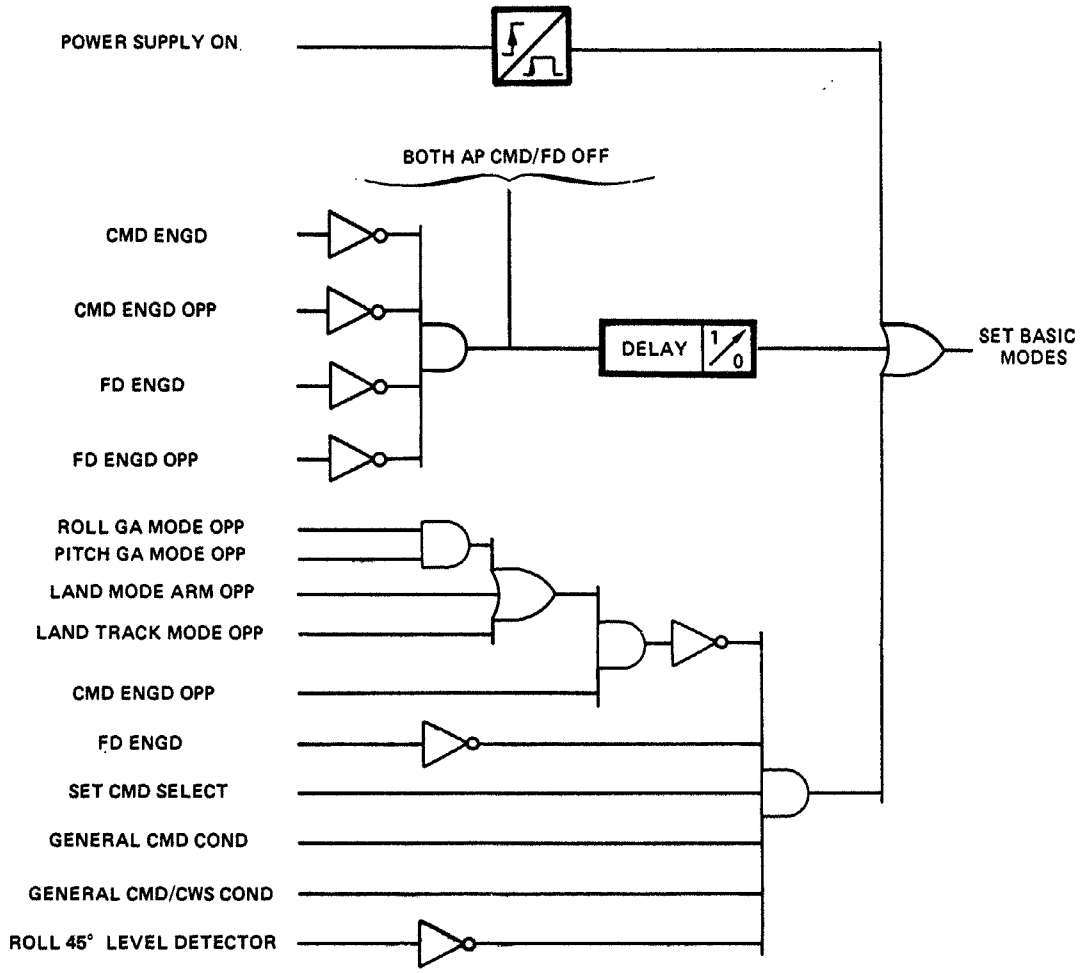
V/S Mode Engagement Using (Turn and Pull) V/S Setting Knob
Figure 027



HDG Mode Operation
Figure 028



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BASIC MODE Engagement Logic Conditions for Both Axes
Figure 029

- when in altitude capture phase, the capture conditions are lost because of a change in the selected flight level
- by pressing a pushbutton switch associated with an active longitudinal mode a second time
- when in active phase of the G/S mode the armed LAND mode is lost
- if any request to return to basic mode appears except in PITCH TO or PITCH GA mode where the loss of both AP/FDs will not cause a return to basic mode).

(Ref. Fig. 030)

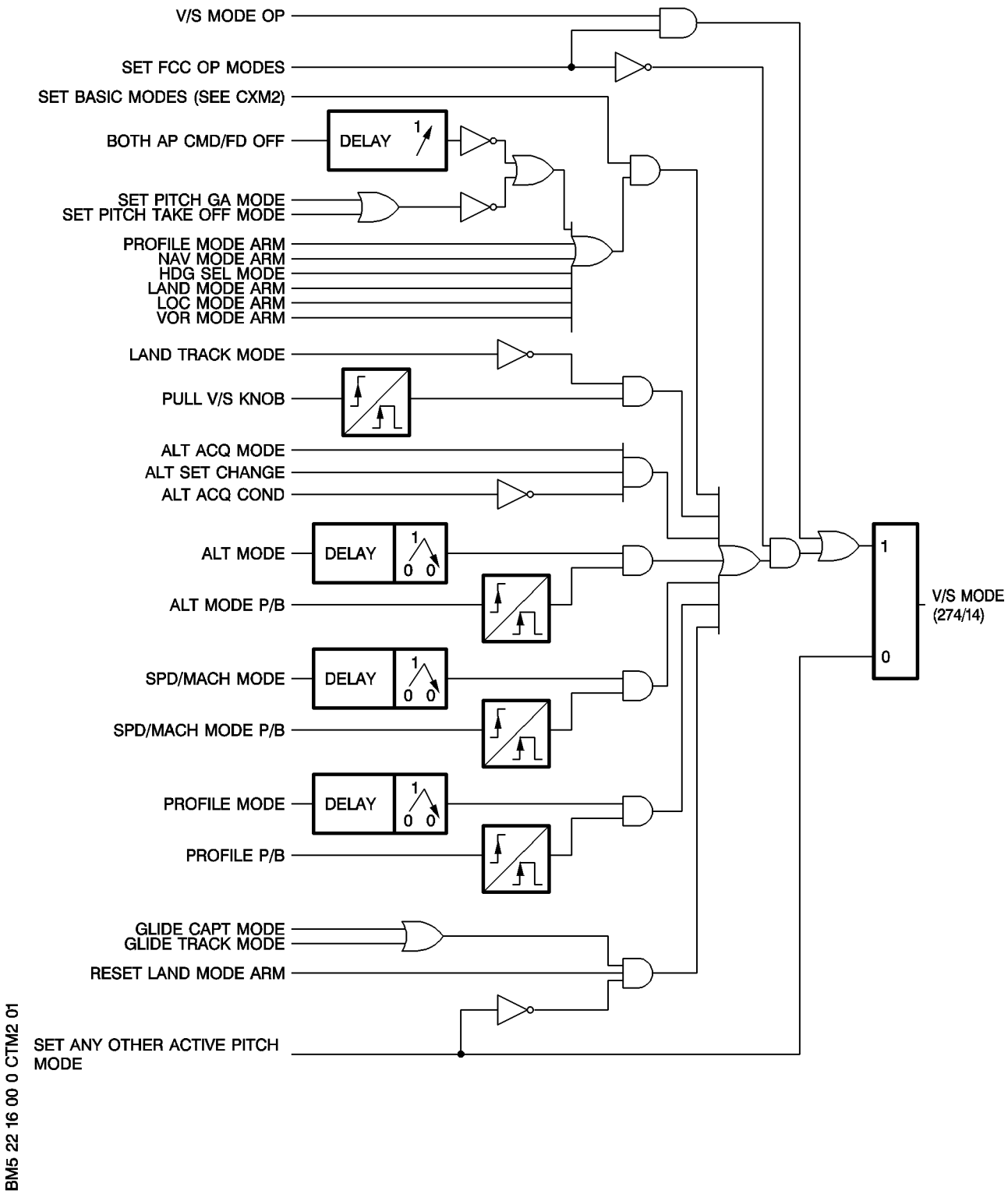
(b) HDG Mode

The HDG mode is engaged :

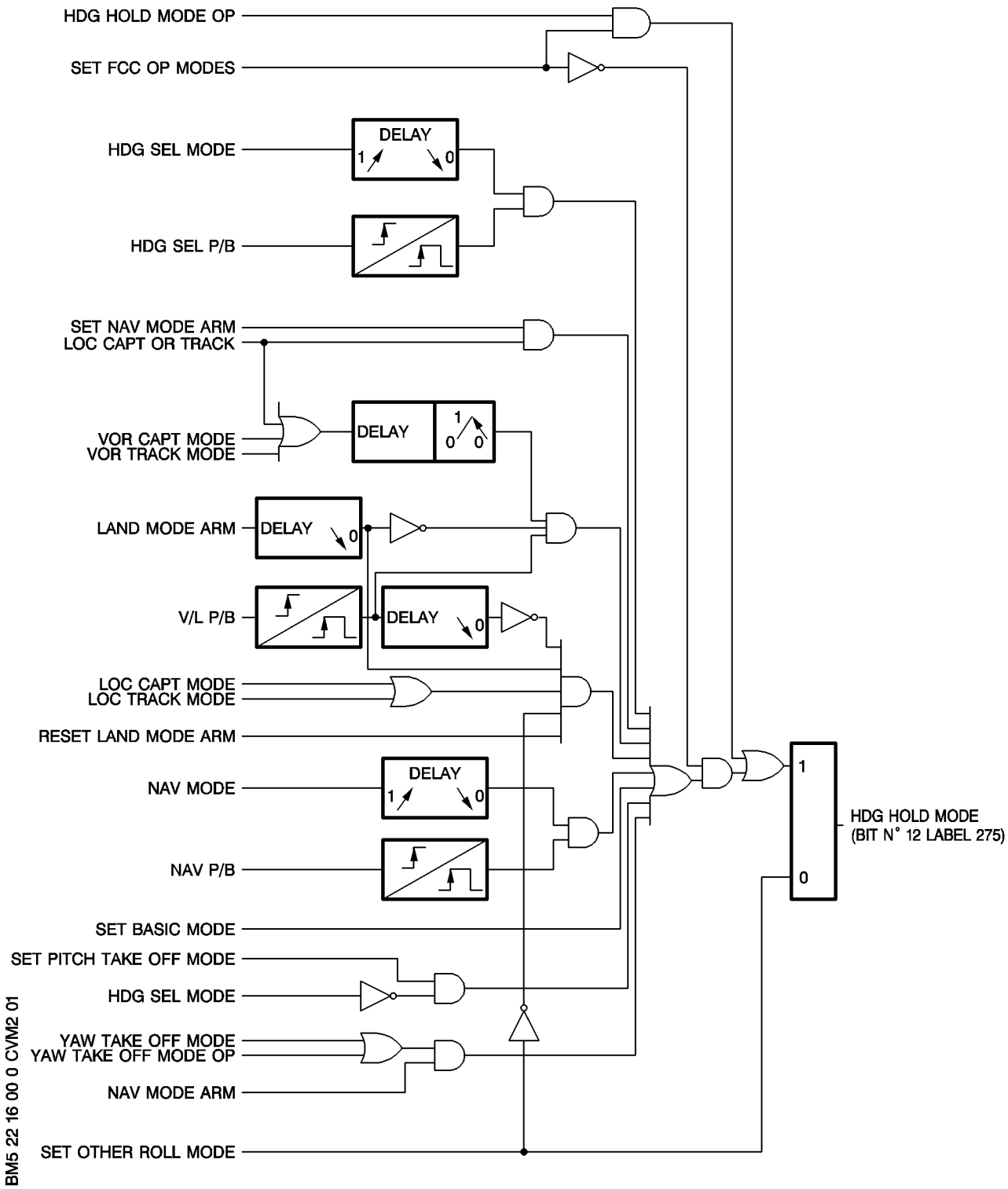
- by pressing a pushbutton switch associated with an active lateral mode a second time.
- by arming the NAV mode, LOC CPT or LOC TRACK phases being active
- through the loss of the LAND mode, LOC CPT or LOC TRACK phases (LAND mode) being active.
- if any request to return to basic mode appears.
- at TAKEOFF mode engagement if a lateral mode other than HDG SEL was engaged.
- when arming the NAV mode, YAW TAKEOFF mode being active.

NOTE : An ILS transmitter failure in an approach phase does not disengage the AP/FD.

(Ref. Fig. 031)

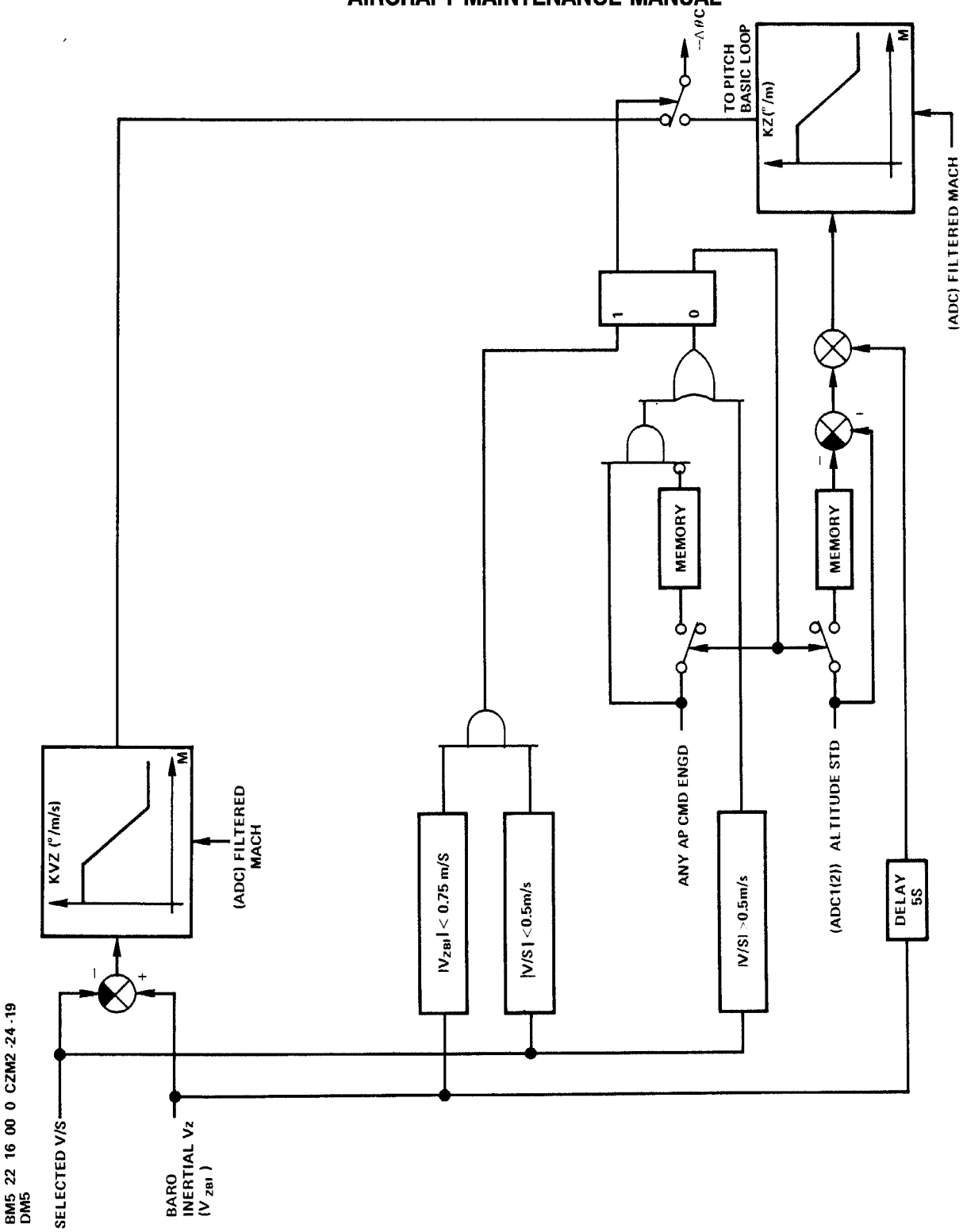


V/S Mode Logic
Figure 030

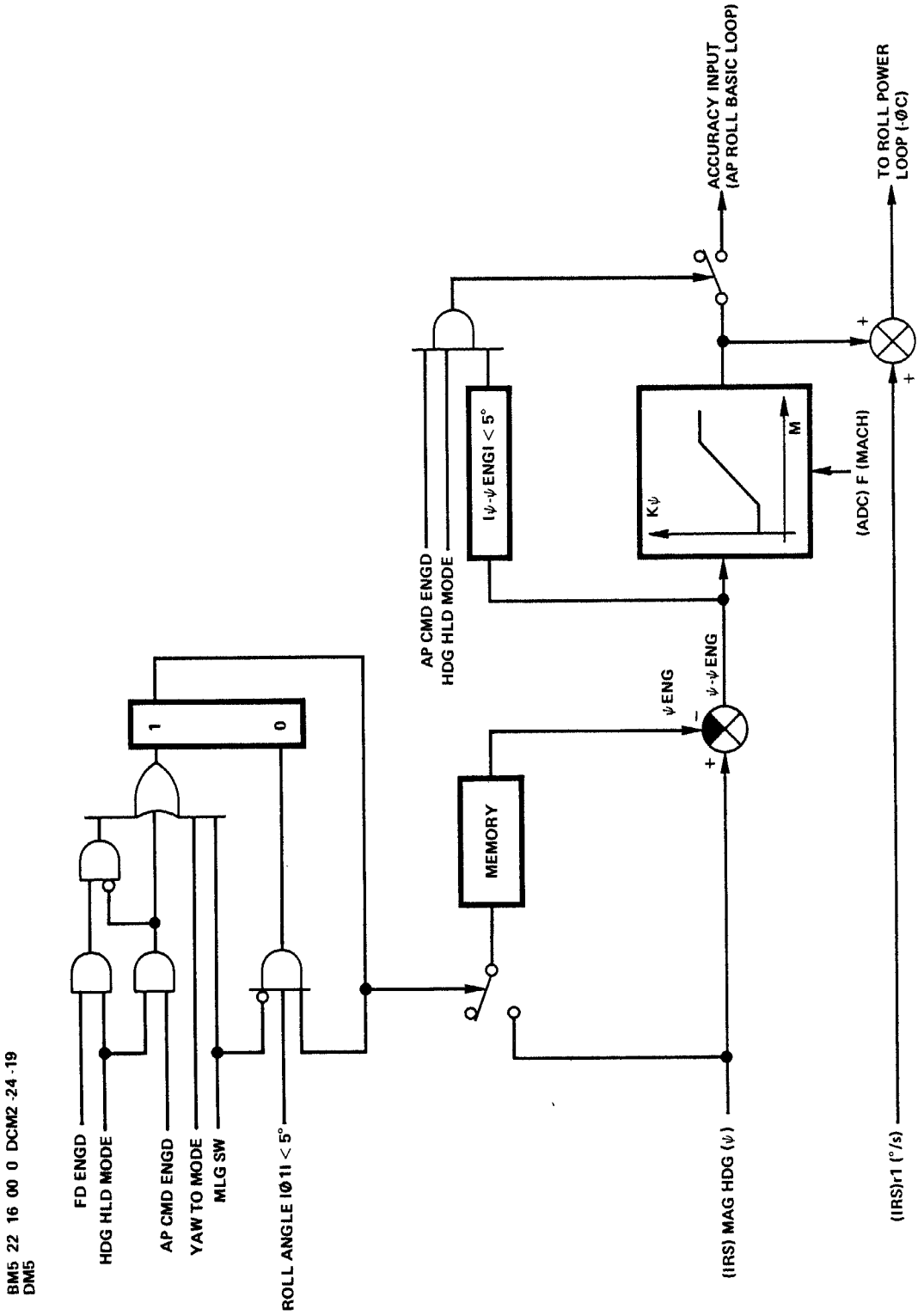


HDG HLD Mode Logic
Figure 031

R (4)Command generation
R (a)V/S mode
R The generation principle of the guiding commands in V/S mode is given
R in the following figure.
(Ref. Fig. 032)
(b)HDG mode
The generation principle of the guiding commands in HDG mode is given in the following figure.
If the aircraft is engaged in a turn, it is first levelled off and the heading obtained at 5° of roll attitude is then used as a reference (Ref. Fig. 028).
(Ref. Fig. 033)



V/S Mode Computation
Figure 032



HDG HOLD Computation
Figure 033

(5)AP/FD speed protection

The speed protection is active with the AP or the FD engaged in the following Modes : V/S, ALT*, LVL CH and PROFILE (FM).

The speed protection is applied in order to protect against Vmin in climb or Vmax in descent with the AP authority limited to ± 0,3g when the protection is active.

It is inactive on ground and until 5s after lift off.

(a)Description of speed protection (Vc prot):

1 VLS-5kt for low speed protection

This low speed protection is active when:

- In V/S mode: selected Vz is greater than 100ft/min
- In ALT* mode: selected altitude is greater than aircraft altitude at ALT* engagement
- In LVL CH mode: selected altitude is greater than aircraft altitude
- In PROFILE mode: if PROFILE CLIMB is active.

2 VMAX +4kt for high speed protection (with VMAX = VMO/MMO in clean configuration gear up or VLE in clean configuration gear down or VFE in slats out configuration).

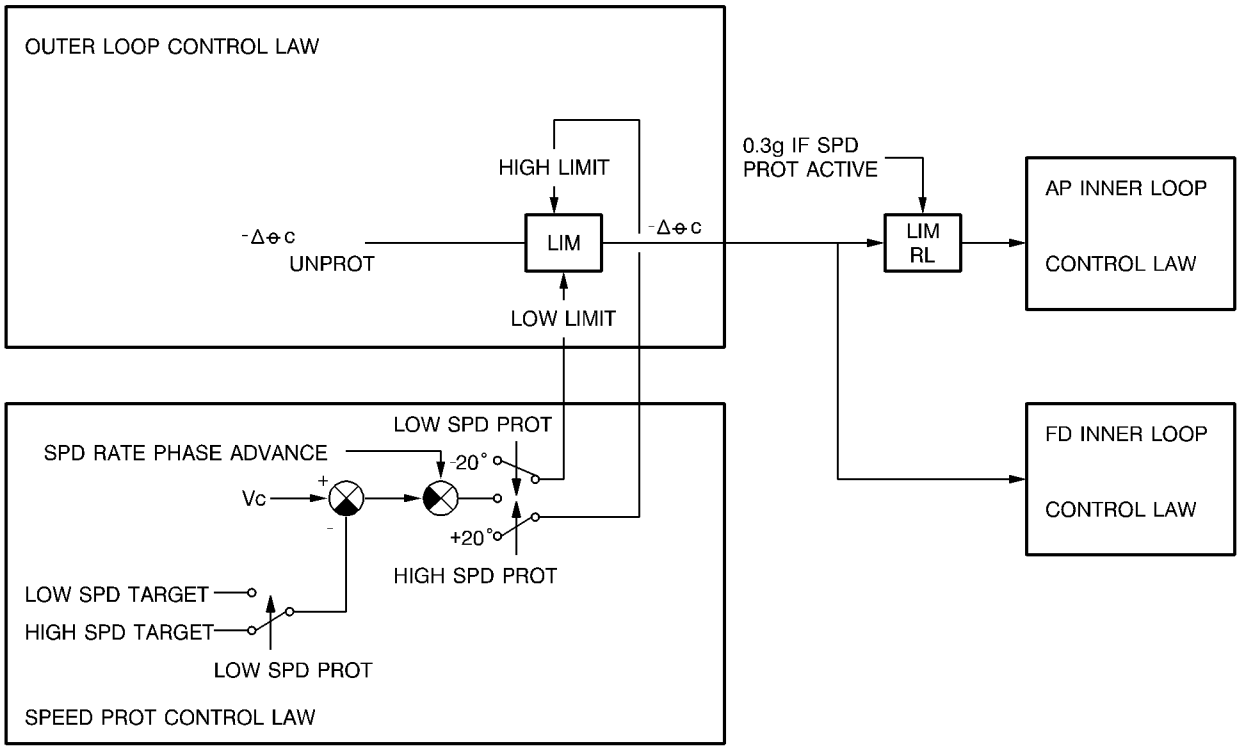
This high speed protection is active when:

- In V/S mode: selected Vz is lower than -100ft/min
- In ALT* mode: selected altitude is lower than aircraft altitude at ALT* engagement
- In LVL CH mode: selected altitude is lower than aircraft altitude
- In PROFILE mode: if PROFILE DESCENT is active.

Only one speed protection (low or high) can be active at a time.

When the speed protection is not active, the operational speed envelope remains VLS for low speed and VMAX for high speed.

(Ref. Fig. 034)



BM5 22 16 00 0 DCN2 00

AP/FD Speed Protection
Figure 034

D. AP OFF Warning

AP OFF warning is activated when both autopilots are lost (AP levers in OFF position).

This warning consists of :

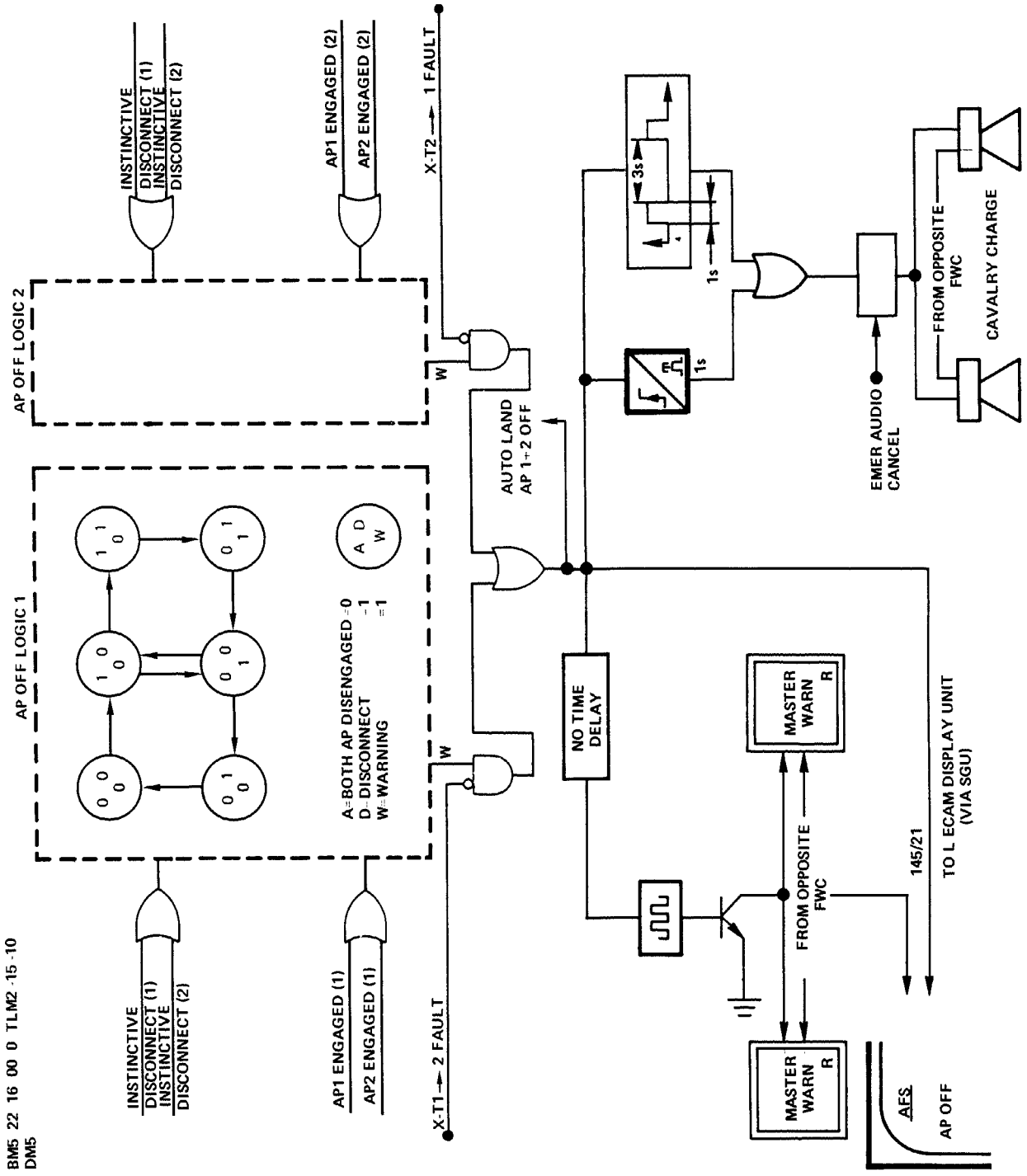
- two red MASTER WARN flashing lights on CAPT and F/O main instruments panels.
- a red visual warning message on the L ECAM display unit :
  AFS
  AP OFF
- an intermittent cavalry charge (this warning sounds for 1 s then stops for 3 s etc.).

Below 200 ft. this warning is repeated by the AUTOLAND warning lights.

The crew can cancel this warning by pressing one AP disconnect pushbutton switch.

The visual warning is cancelled immediately, the aural warning is maintained for at least 1 s.

(Ref. Fig. 035)



AP OFF Warning
Figure 035

UPPER MODES - DESCRIPTION AND OPERATION

1. General

All the modes that can be selected are available for the AP and the FD but :
- the TAKEOFF mode is available for the FD only as long as the aircraft is on the ground (AP engagement in CMD is inhibited on the ground with engines running)
- the flare phase of the LAND mode is not certified for the FD.

Since both FDs are automatically engaged on energization, AP is engaged in the mode in which FD was engaged except if AP opposite to FD already engaged is engaged (associated FD being disengaged) : in this case there is a return to basic mode (Ref. 22-16).

As the pushbutton switches corresponding to the modes engaged come on and as the systems operate in dual FD and dual AP configuration (in LAND or GO AROUND mode) it is necessary to synchronize modes and phases for those modes using an arming phase.

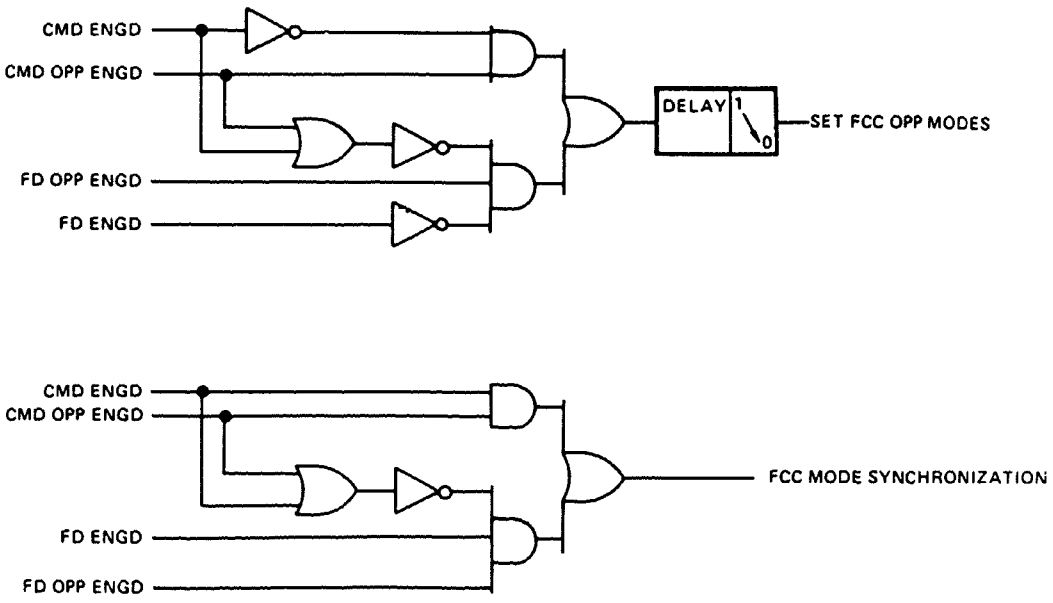
To this end, one FCC synchronizes on the modes transmitted by the opposite FCC in the following cases :
- only opposite FCC is engaged in CMD
- no AP is engaged in CMD and only opposite FD is engaged.

Phase synchronization for the modes using an arming phase (change from the arming phase to the capture phase etc...) is carried out :
- inside an FCC between the command and monitoring channels. This is a loose synchronization : when a computation channel detects the phase change, the detection thresholds of the other channel are reduced to a certain limit
- between the FCCs, if one FCC receives the phase change detected by the command and monitoring channels from the opposite FCC.

Engagement of a mode when a logic condition (required by this mode) is incorrect, is made as follows :
- when only one FCC system is engaged :
  - . if the failure exists when the mode is selected, the mode is not taken into account
  - . if the failure occurs after the mode has been engaged, the AP and the FD disengage (the FD will return to basic mode on the axis concerned).
- when both FCCs are operating (dual FD or dual AP configuration in LAND or GO AROUND mode).

The selection of the mode is fed into both FCCs but the FCC associated with the incorrect logic condition disengages.

(If only one AP is engaged in CMD, the modes taken into account by the AP will be imposed on both FCCs. In the event of a failure, the mode acquisition logic corresponds to the case when only one FCC is engaged).
(Ref. Fig. 001)



BM5 22 17 00 0 AAM2 13 08
DM5

Mode Synchronization Conditions
Figure 001

R 1. Altitude Hold Mode (ALT)

A. Characteristics
This mode can be engaged by means of the ALT HLD pushbutton switch. The aircraft holds altitude at mode engagement (if the mode is engaged with an aircraft vertical speed other than zero, the ALT mode maintains the altitude de obtained after vertical speed is reduced to 85 ft/mn) (Ref. Fig. 003).

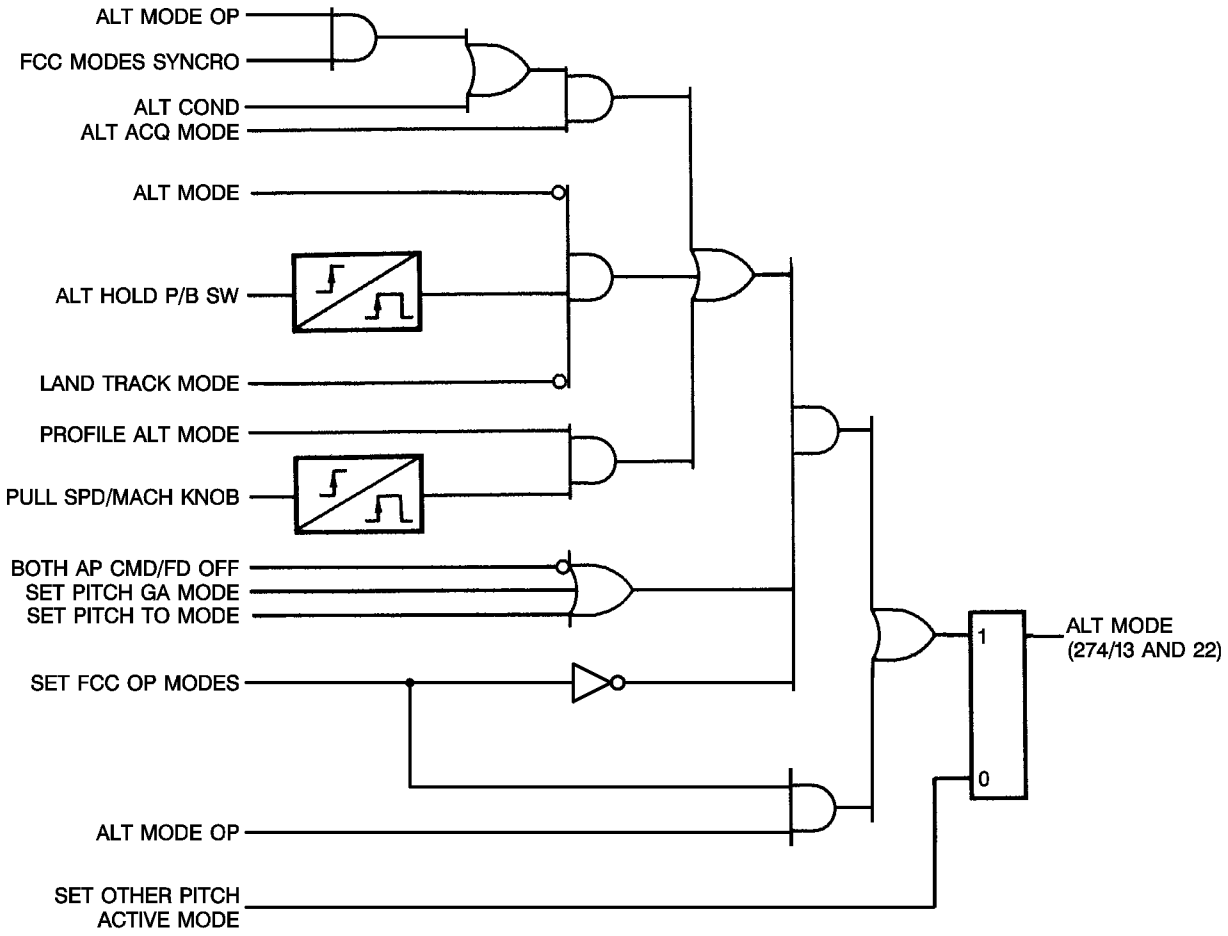
B. Operation
As far as mode phase synchronization is concerned refer to 22-17-00 paragraph 1. Change from ALT ACQ to ALT mode can be initiated by the opposite FCC.
If FD only was engaged in ALT mode and altitude selected on the FCU was maintained, AP engagement in CMD results in :
- capture of the altitude displayed in the ALT SEL counter on the FCU if the altitude deviation is lower than 250 ft.
- hold of the aircraft altitude at AP engagement in CMD if the altitude deviation is greater than 250 ft. compared with the altitude displayed in the counter (if the vertical speed at AP engagement in CMD is not null, the resulting altitude is held after vertical speed is cancelled) (Ref. Fig. 004).

In the event of loss of both AP/FDs, the ALT mode is inhibited and there is a return to the basic mode except if the PITCH TO or PITCH GA mode is engaged.
The engage and disengage conditions of the ALT mode are given in the following figure.

**ON A/C ALL

(Ref. Fig. 001)

**ON A/C ALL



BM5 22 17 10 0 AJV2 00

ALT Mode Logic
Figure 001

C. Command generation
The generation principle of the AP/FD commands in ALT mode is given in following figure.

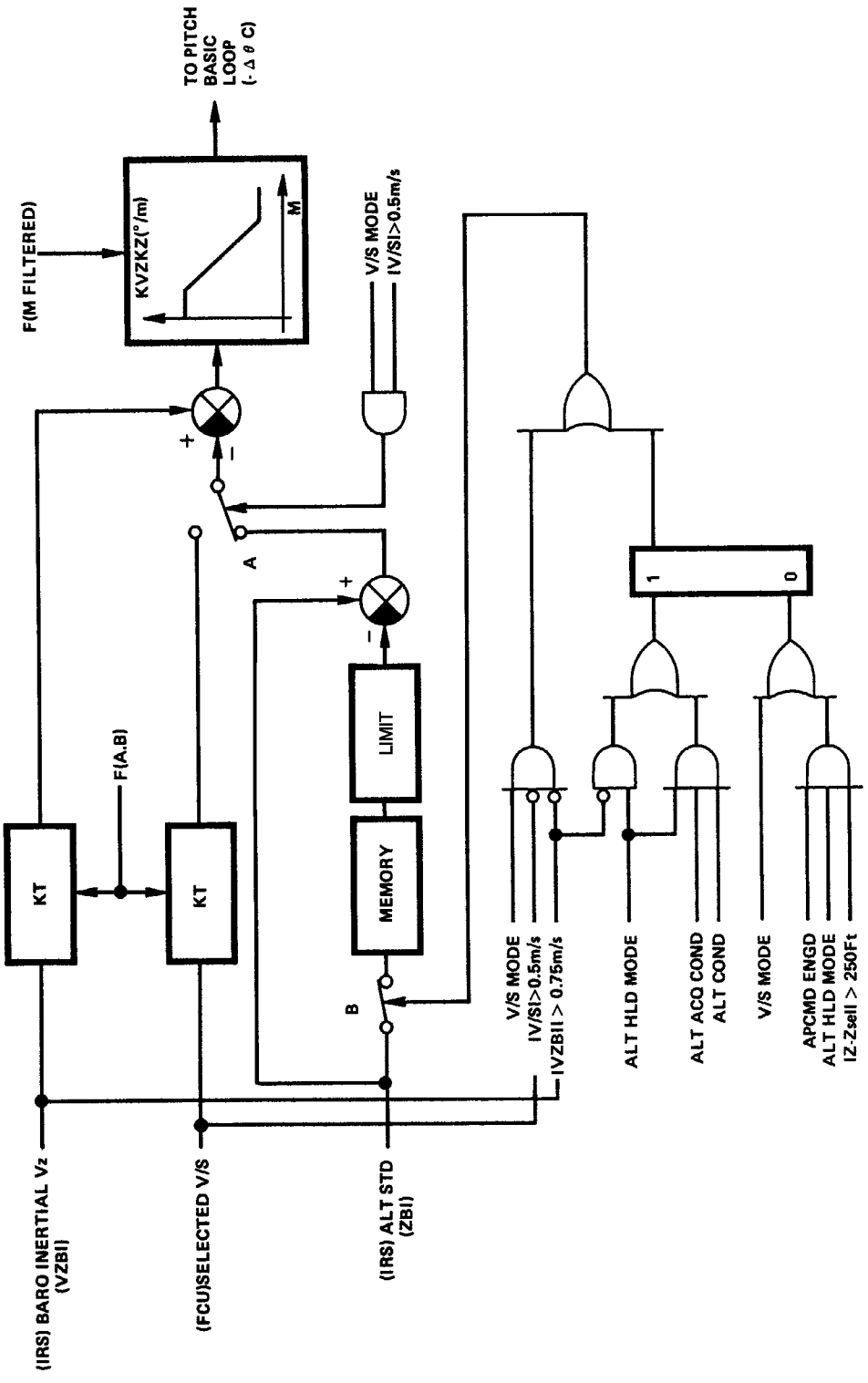
**ON A/C ALL

R

(Ref. Fig. 002)

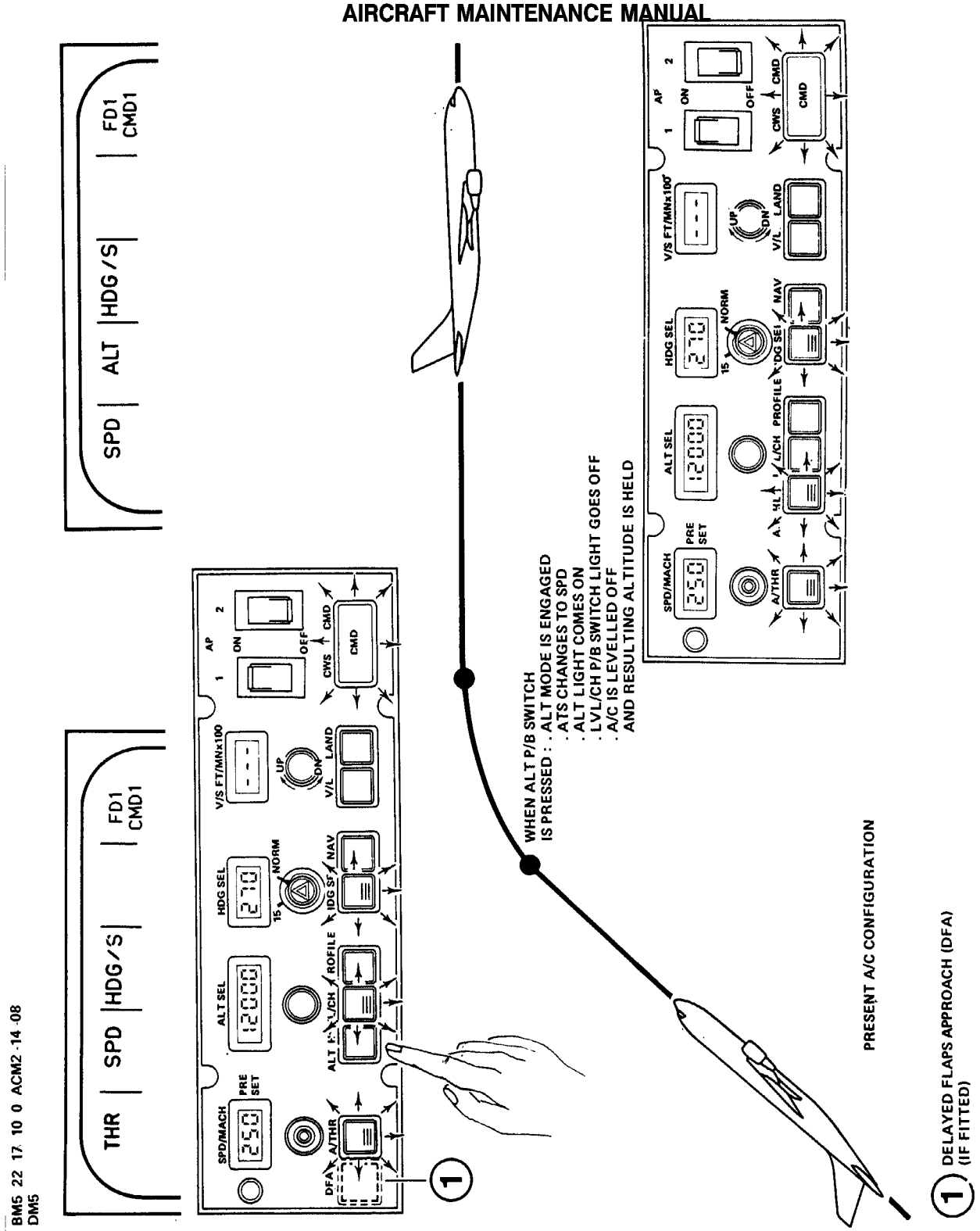
**ON A/C ALL

D. Display
The green ALT indication appears on the PFDs when the mode is active.



BM5 22 17 10 0 AGT2 00

ALT HLD Mode Computation
Figure 002



ALT Mode Engagement with A/C VERT SPEED not Null
Figure 003

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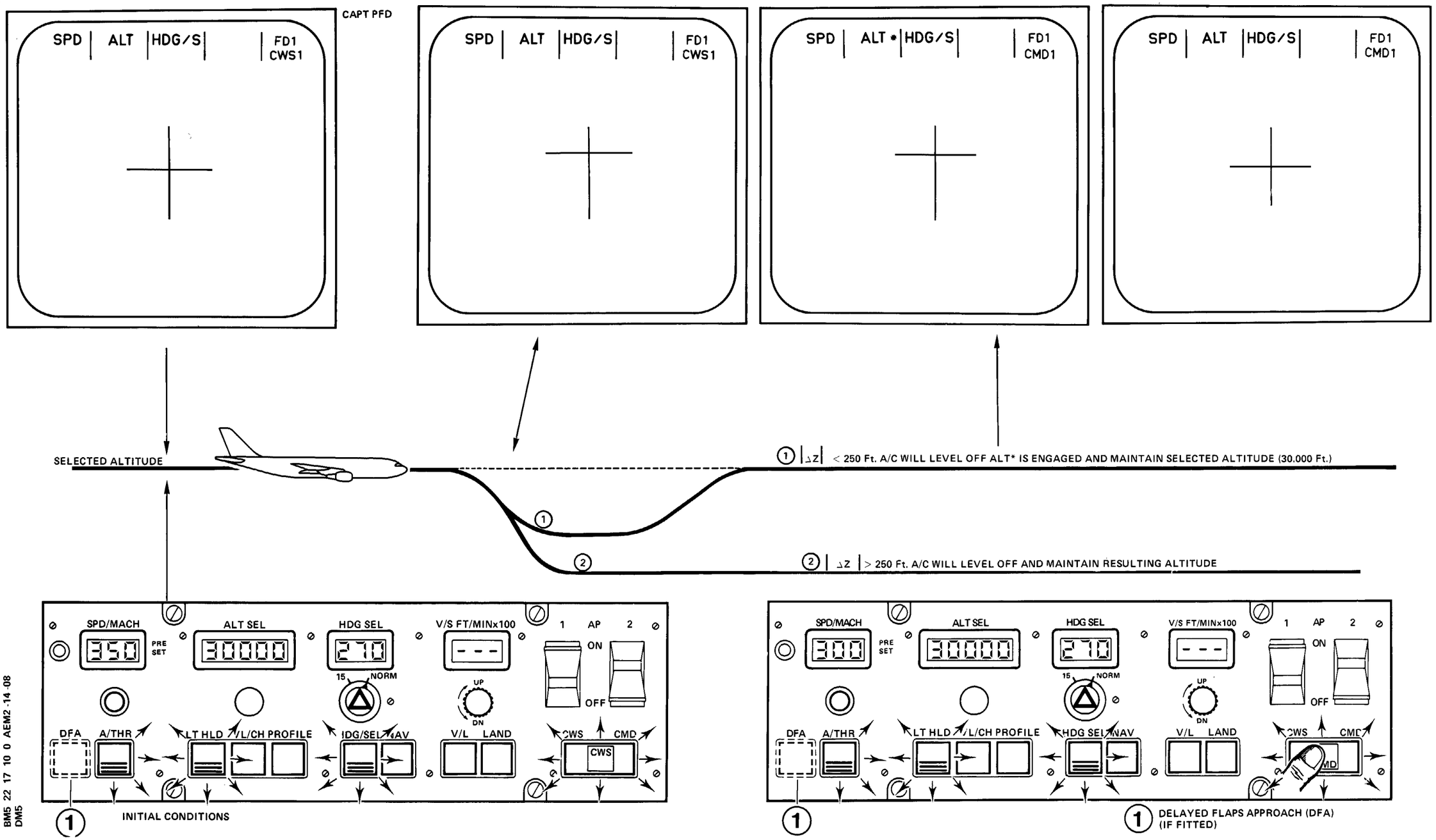


BM5 22 17 10 0 AEM2.14.08
DM5

Engagement of AP in CMD When FD in ALT
Figure 004

2. Altitude Acquisition Mode (ALT ACQ)

A. Characteristics
The ALT ACQ mode enables to acquire the altitude selected in the ALT SEL counter on the FCU.
It includes an arming phase in which a longitudinal support mode (LVL/CH, V/S...) ensures convergence toward the desired level.
It also includes a capture phase which starts as soon as the aircraft altitude deviation compared with the level selected on the FCU IS lower than a certain value dependent on vertical speed.
When the desired level is reached, the ALT mode engages automatically and replaces the ALT ACQ mode.

B. Operation
No specific pushbutton switch is associated with the ALT ACQ mode.
The logic conditions associated with the arming phases of the ALT ACQ mode are given in following figure.

**ON A/C ALL

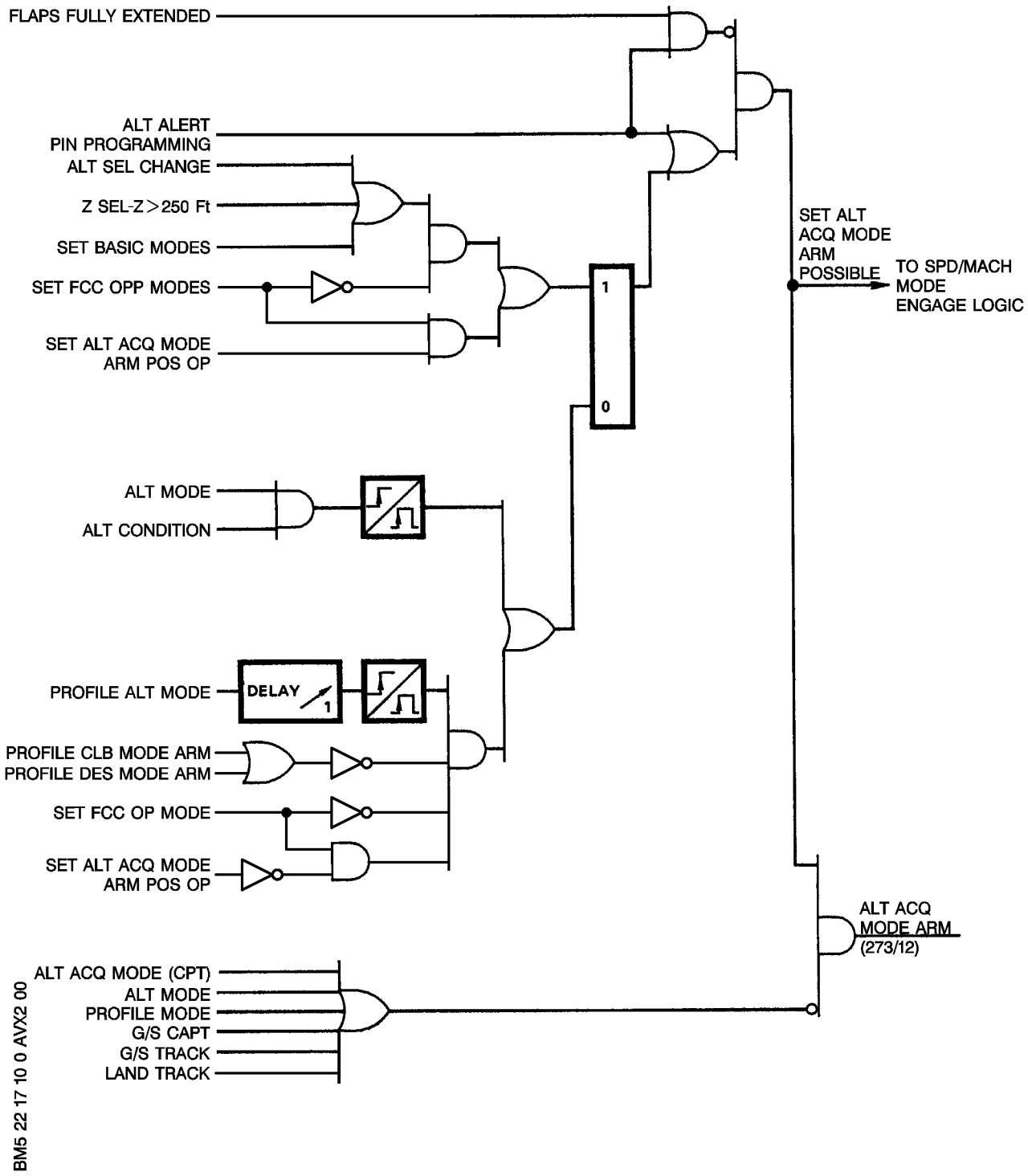
R

(Ref. Fig. 005)

**ON A/C ALL

(1)The sequence of operation is as follows : (Ref. Fig. 007)
- Selection of a flight level on the FCU (the ALT ACQ mode is inhibited during selection)
- Engagement of a support mode enabling capture of selected level
- Capture of selected level ; the ALT ACQ mode becomes active and the autothrottle, if armed, switches to the SPD/MACH mode, (ATS arming lever in ON position and A/THR mode engaged).
- Switching to ALT mode when altitude deviation is lower than 20 ft.

(2)This sequence of operation can be modified if the pilot modifies the reference altitude :
(a)In arming phase :
if, when the pilot stops operating selector knob, the capture conditions are met, the ALT ACQ mode is engaged immediately (Ref. Fig. 008).
(b)In capture phase (Ref. Fig. 009)
If the capture conditions are still satisfied in spite of the reference modification, the ALT ACQ mode remains active with possibly reduced performance.
If the capture conditions are no longer satisfied after the reference modification has been made, the AP/FD returns to V/S basic mode holding the aircraft vertical speed at that moment. The ALT ACQ arming phase is re-activated. The actual altitude level held depends on the barometric condition made on the altimeters. Each FCC is connected to the ADC of corresponding side transmitting the barometric correction made on the



ALT ACQ Mode-Logic Arming Conditions
Figure 005

associated altimeter.
The air data computer sends two altitude signals :
- label 203 : standard altitude
- label 204 : barometric pressure corrected altitude.

Any of the two signals can be used by the FCC depending on the discrete sent to the altimeter by the FCC :
- **BARO** knob pulled : ground-no correction (standard altitude of 1013 mbar is taken into account)
- **BARO** knob pushed : open circuit = corrected altitude is taken into account.

The corrected barometric altitude-standard altitude reference modification is not taken into account in **ALT** mode or in capture phase of **ALT ACQ** mode.
In the event of loss of both **AP/FDs**, the **ALT ACQ** mode is inhibited and there is a return to basic mode except if the **PITCH TO** or **PITCH GA** mode is engaged.
The logic conditions associated with the capture phase of the **ALT ACQ** mode are given in the following figure.

**ON A/C ALL

R

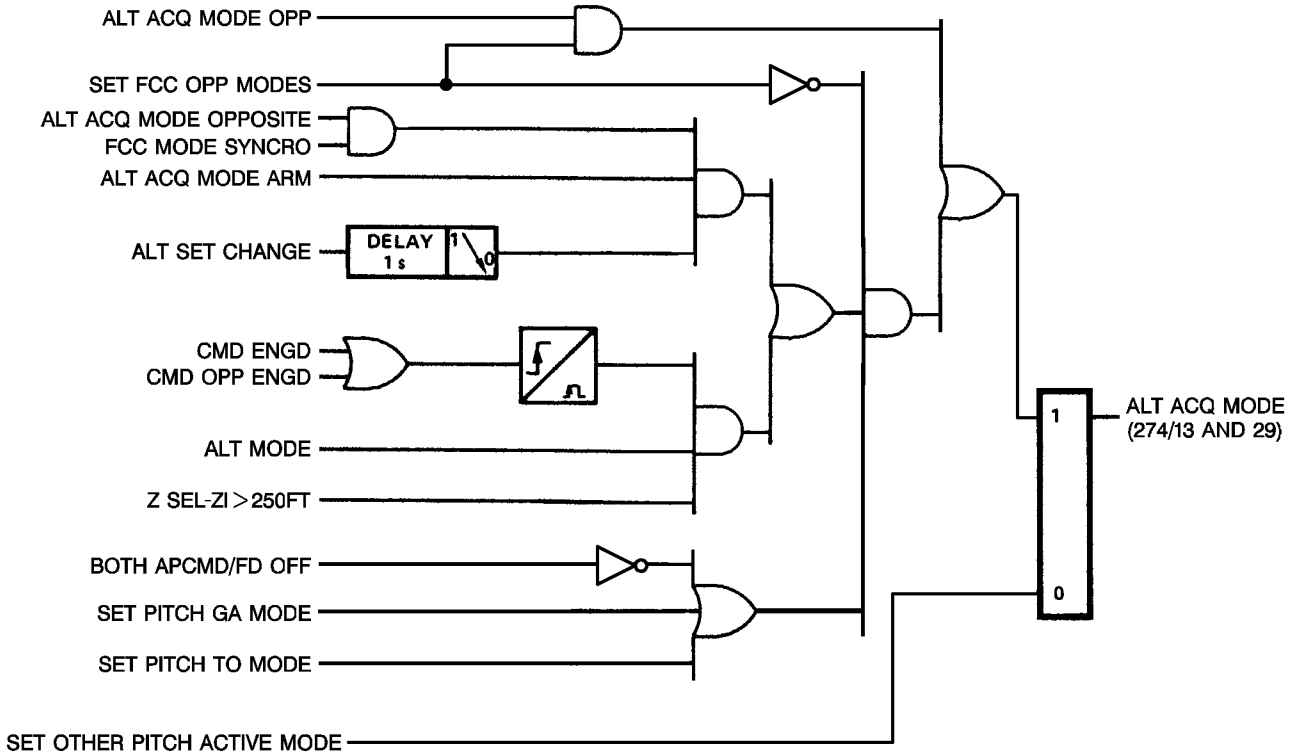
(Ref. Fig. 006)

**ON A/C ALL

C. Command generation
The computation principle of the guidance law in **ALT ACQ** mode is given in figure (Ref. Fig. 010).

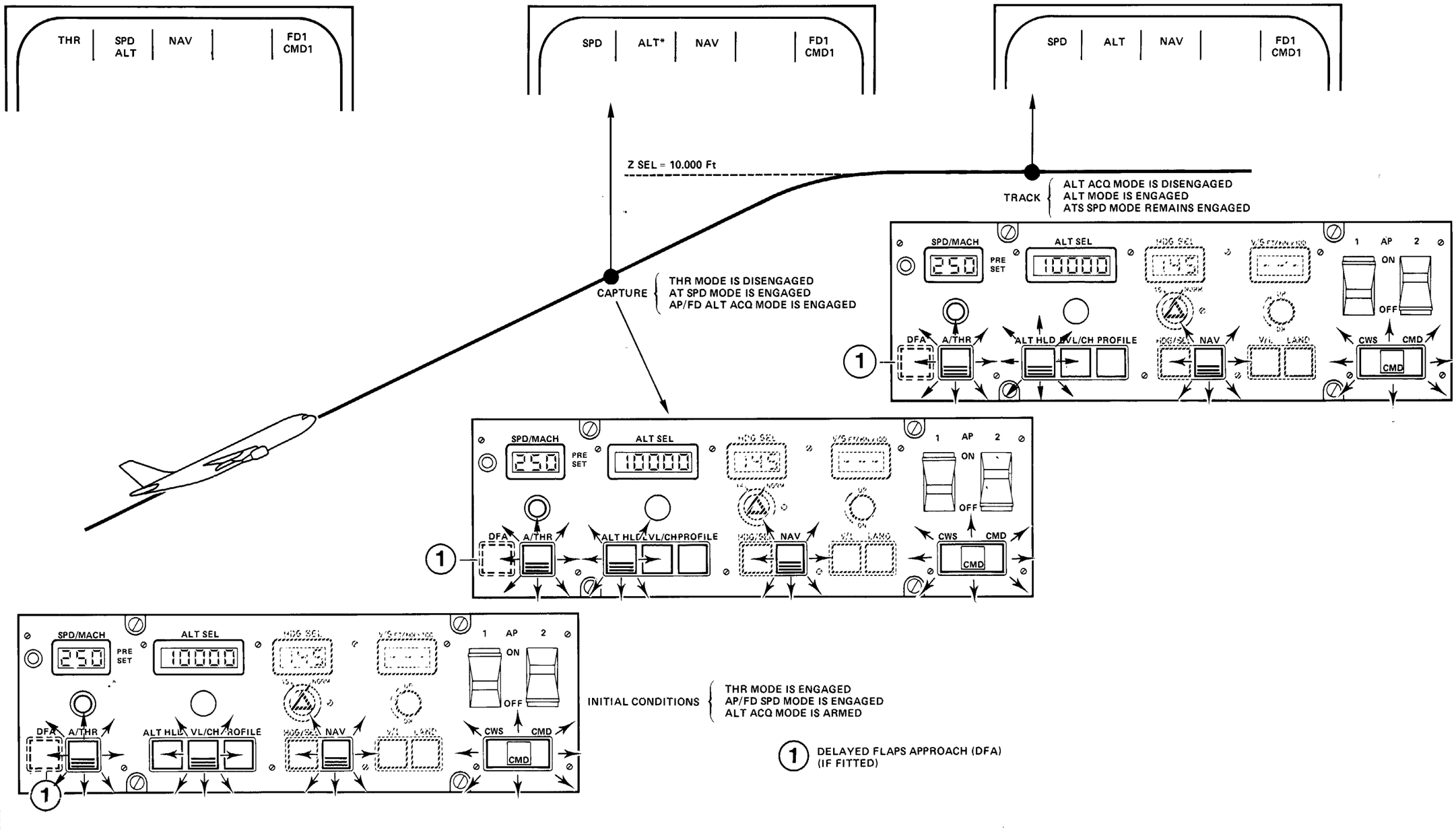
D. Display
- arming of the mode is displayed on the **PFD** by the cyan **ALT** indication
- the capture phase is displayed on the **PFD** by the green **ALT*** indication
- the reference level is permanently indicated on the altitude counter.

(Ref. Fig. 009, 010).



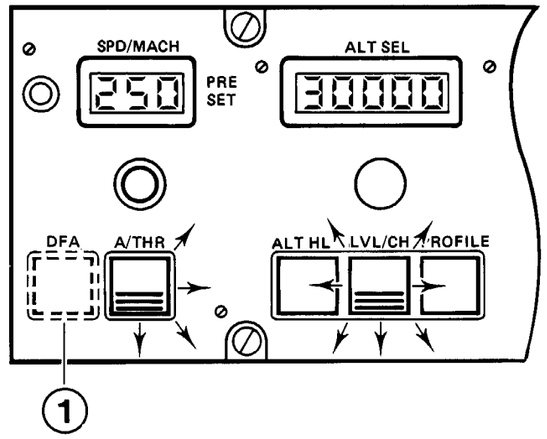
BM5 22 17 10 0 AXV2 00

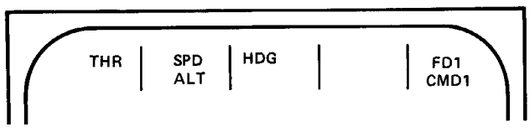
ALT ACQ Mode-Capture Logic Conditions
Figure 006

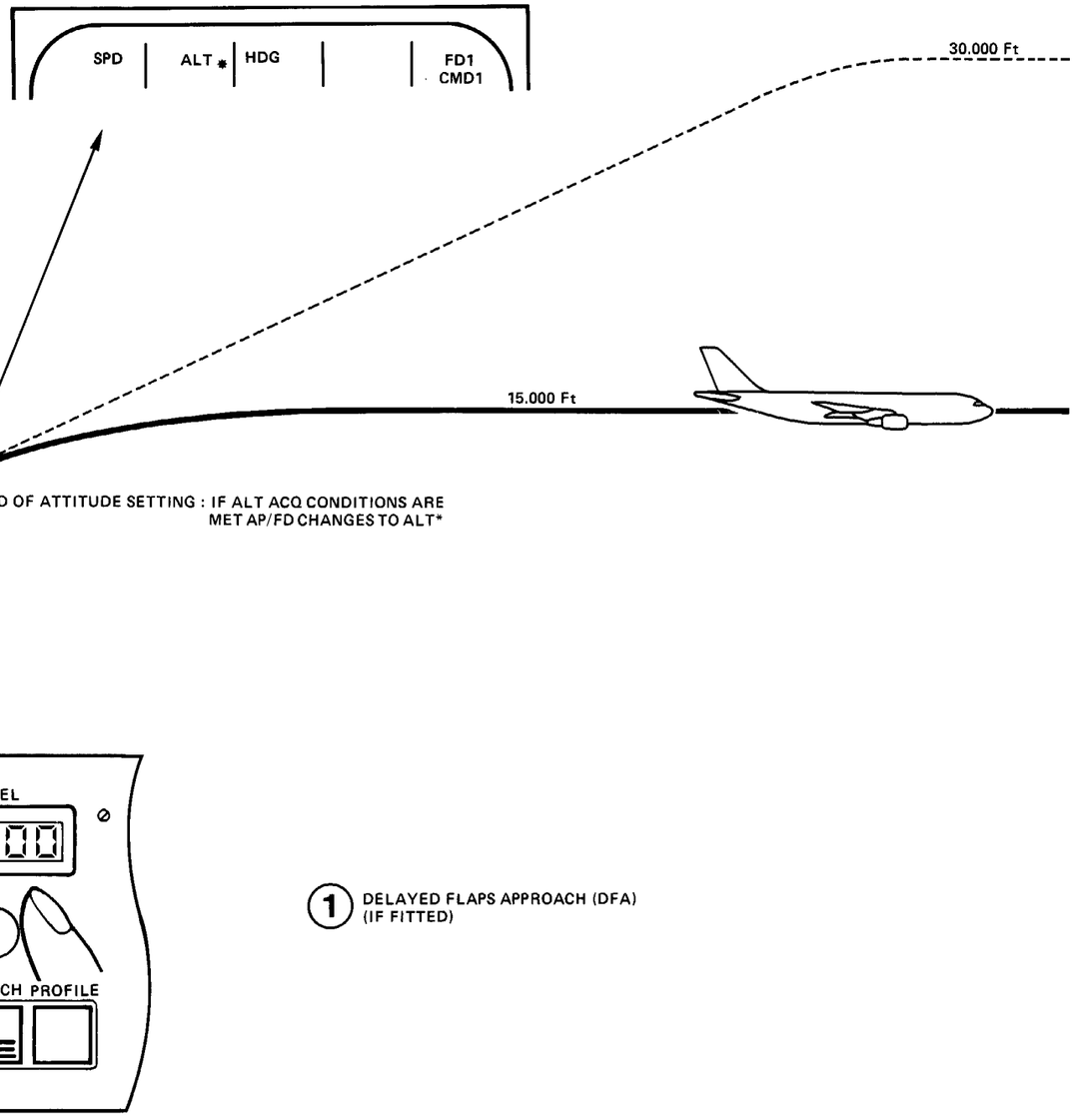


BM5 22 17 10 0 ALM2 24 19
DM5

ALT ACQ Mode - Sequence Operation
Figure 007



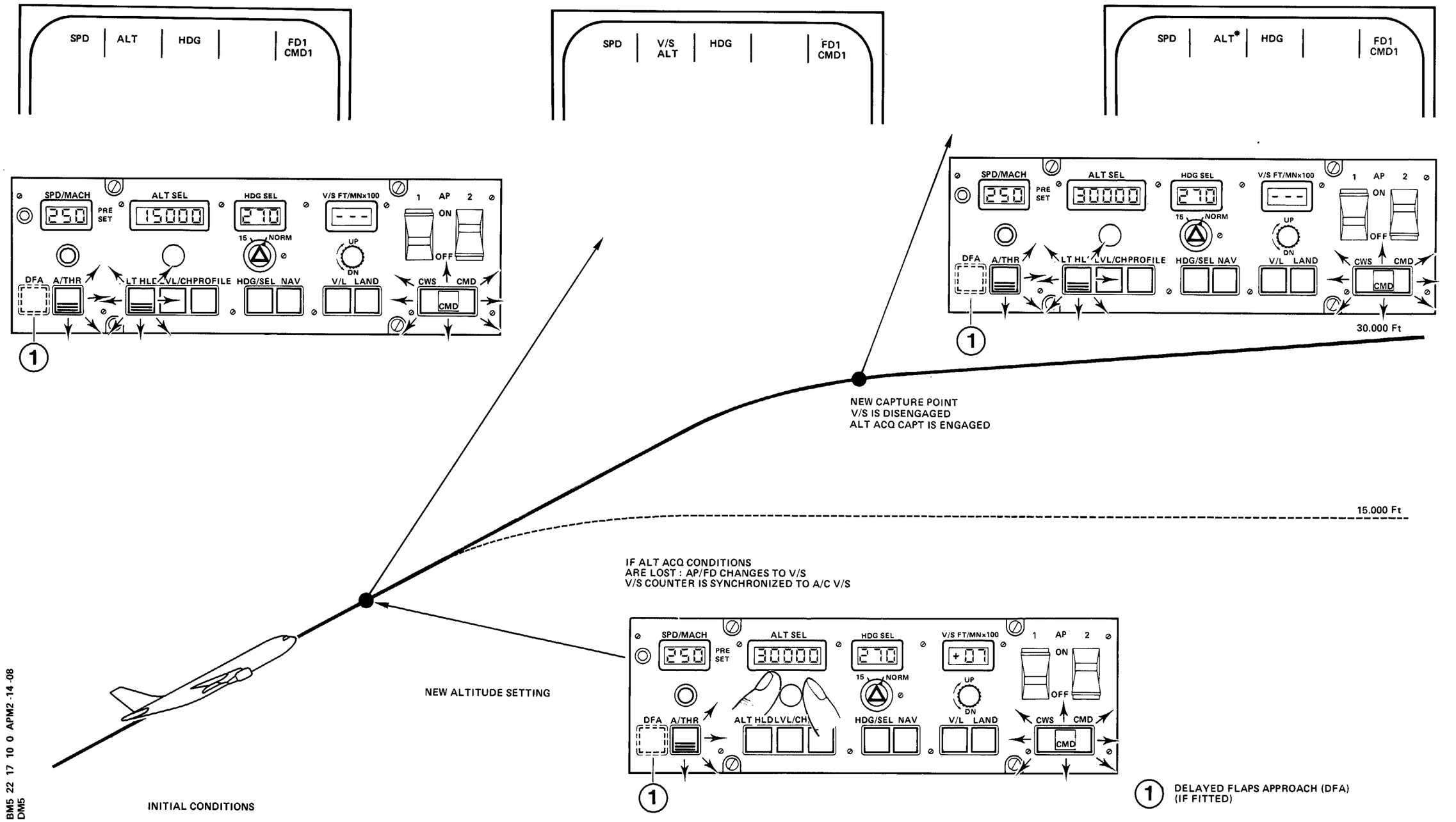




(1) DELAYED FLAPS APPROACH (DFA) (IF FITTED)

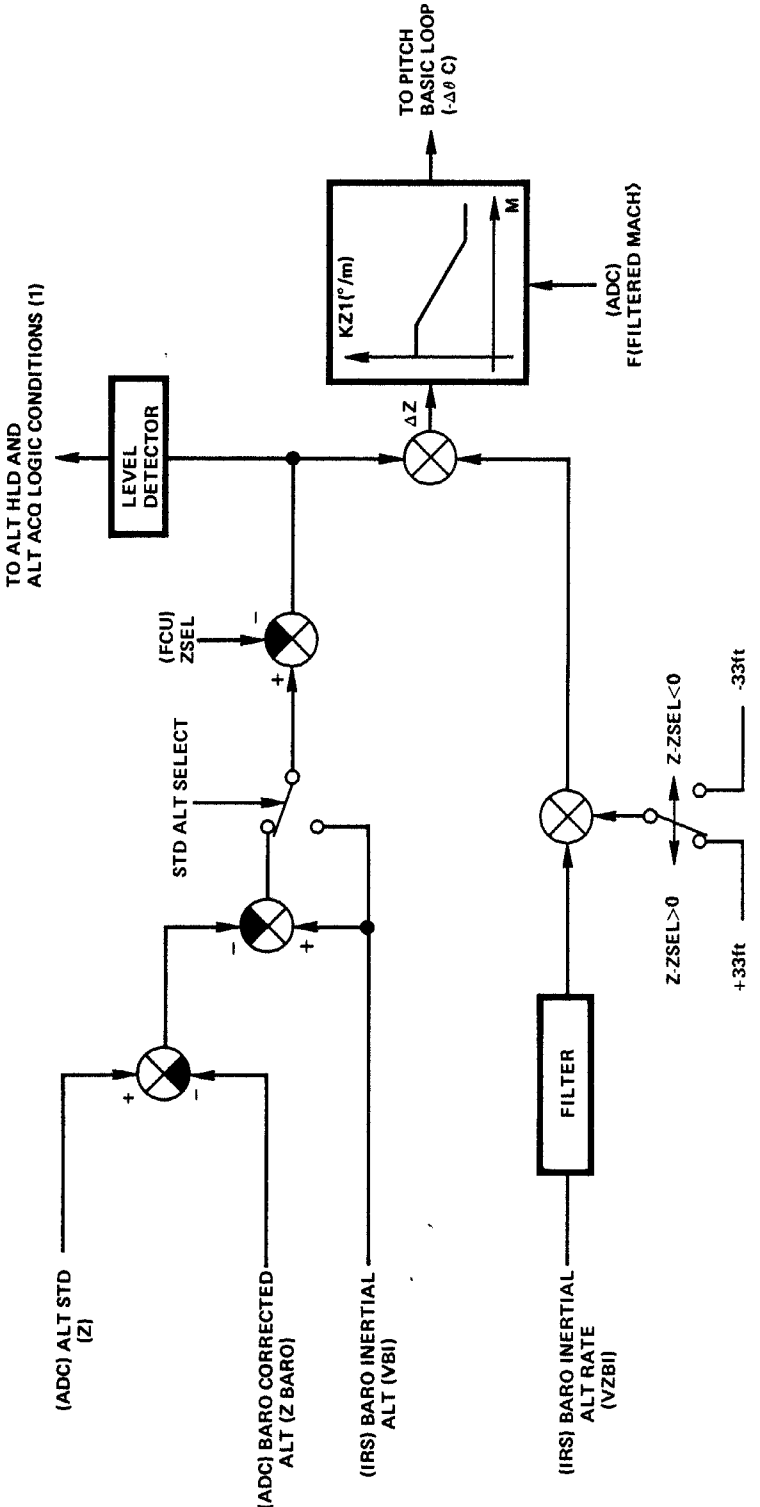
BM5 22 17 10 0 ANM2 14 08
DM5

ALT ACQ Mode - Change of Selected Altitude During Arming Phase
Figure 008



BM5 22 17 10 0 APM2 14 08
DM5

ALT ACQ Mode - Change of Selected Altitude During Capture Phase
Figure 009


TO PITCH BASIC LOOP (-Δθ C)
KZ1(°/m)
M
(ADC) F(FILTERED MACH)
ΔZ
TO ALT HLD AND ALT ACQ LOGIC CONDITIONS (1)
LEVEL DETECTOR
(FCU) ZSEL
STD ALT SELECT
Z-ZSEL<0
-33ft
Z-ZSEL>0
+33ft
FILTER
(ADC) ALT STD (Z)
(ADC) BARO CORRECTED ALT (Z BARO)
(IRS) BARO INERTIAL ALT (VBI)
(IRS) BARO INERTIAL ALT RATE (VZBI)
(1) - ALT COND = 1 IF Z-ZSEL ≤ 2 oft DURING 1.25 SEC
- ALT ACQ COND = 1 IF (Z-ZSEL). Δ Z < 0 + ALT COND = 1

BM5 22 17 10 0 ATM2 24 19
DM5

ALT ACQ Mode Computation
Figure 010

3. Level Change Mode (LVL/CH)

A. Characteristics
The LVL/CH mode enables to perform level changes with reduced crew workload.
Engagement of this mode activates :
- speed or Mach hold (SPD/MACH) modes for the AP/FD
- thrust control for the autothrottle :
  . either THR limit hold in climb configuration
  . or thrust reduction (RETARD) in descent configuration.

B. Operation
(1)Engagement/Disengagement
The following figure shows the engage and disengage conditions of the LVL/CH mode as well as the generation of SPD/MACH mode data (climb or descent) for the autothrottle.
It is to be noted that the loss of both AP/FDs inhibits the LVL/CH mode. This results in a return to the basic mode, except in PITCH TO or PITCH GA mode.
R (Ref. Fig. 011)

R

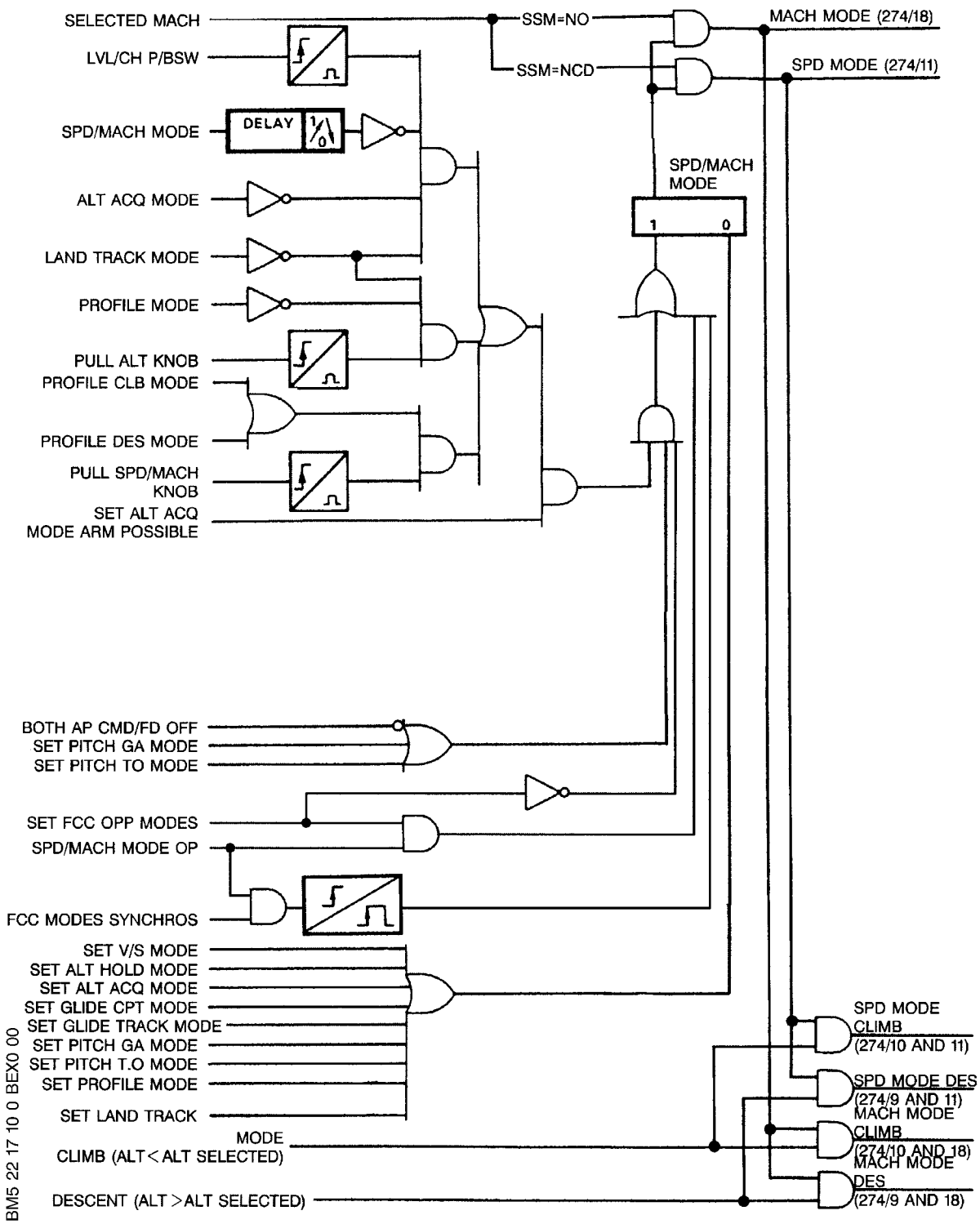
Engagement of this mode has the following results :
(a)If the level selected in the ALT SEL counter on the FCU is higher than the aircraft altitude :
- AP/FD engages in SPD or MACH mode depending on the parameter displayed in the SPD/MACH counter on the FCU (see NOTE 1)
- autothrottle engages in THR mode.

(b)If the level selected in the ALT SEL counter on the FCU is lower than the aircraft altitude :
- AP/FD engages in SPD or MACH mode depending on the parameter displayed in the SPD/MACH counter on the FCU (see NOTE 1)
- autothrottle engages in RETARD mode which causes retard of the throttle control levers to full idle stop and declutching.

However the pilot can reduce the thrust retard by stopping the throttle control levers at the desired position. They are then declutched in this position (Ref. Fig. 012, 013).

NOTE 1 : SPEED or MACH is selected in the SPD/MACH display on the FCU by means of the pushbutton switch located to the left of the display or by automatic switching at determined altitudes.
The switching is made on the aircraft parameter value. The switching of the displayed parameter results in switching of the engaged mode.
In addition, engagement of the LVL/CH mode results in a possible PRESET being taken into account.
Use of the PRESET function is described in 22-15.

C. Command generation
The generation principle of the AP/FD commands in LVL/CH is given in

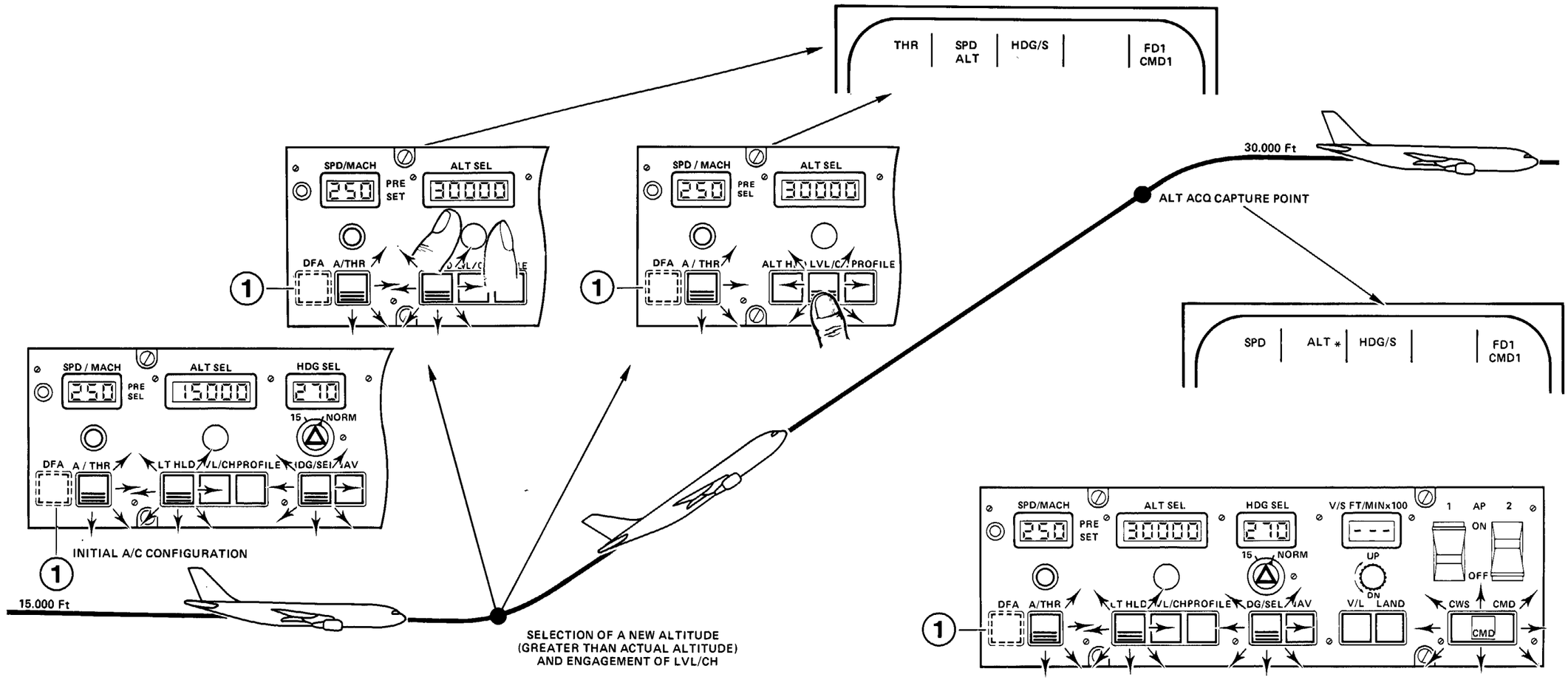


LEVEL CHANGE Mode Logic
Figure 011

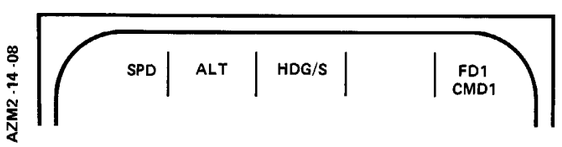
R

figure (Ref. Fig. , 014)
It is to be noted that this mode is protected against underspeed and overspeed conditions. The use of the total slope enables, during acceleration in the climbing phase, or deceleration in the descent phase, to prevent level flight accelerations or decelerations, part of the thrust variation being used in speed change, the other in level change.
In addition, the displacement speed of the throttle control levers is modulated (in climbing or in descent phase) so as to reduce speed variations during transient at LVL/CH engagement.

D. Display
When mode is engaged, the green SPD or MACH indication appears on the PFDs and the LVL/CH pushbutton switch light on the FCU is on.



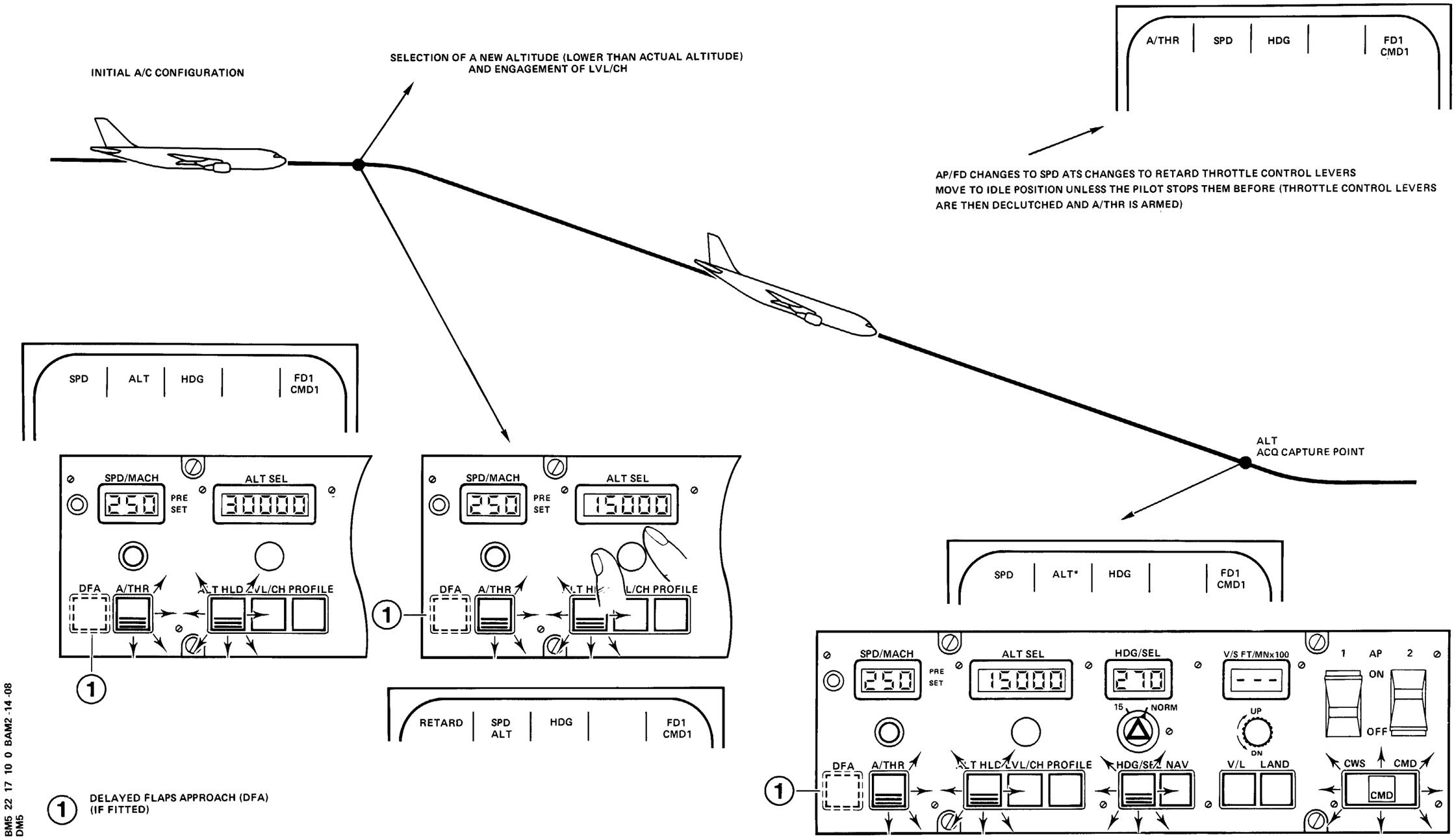
SPD



1 DELAYED - FLAP - APPROACH (DFA) (IF FITTED)

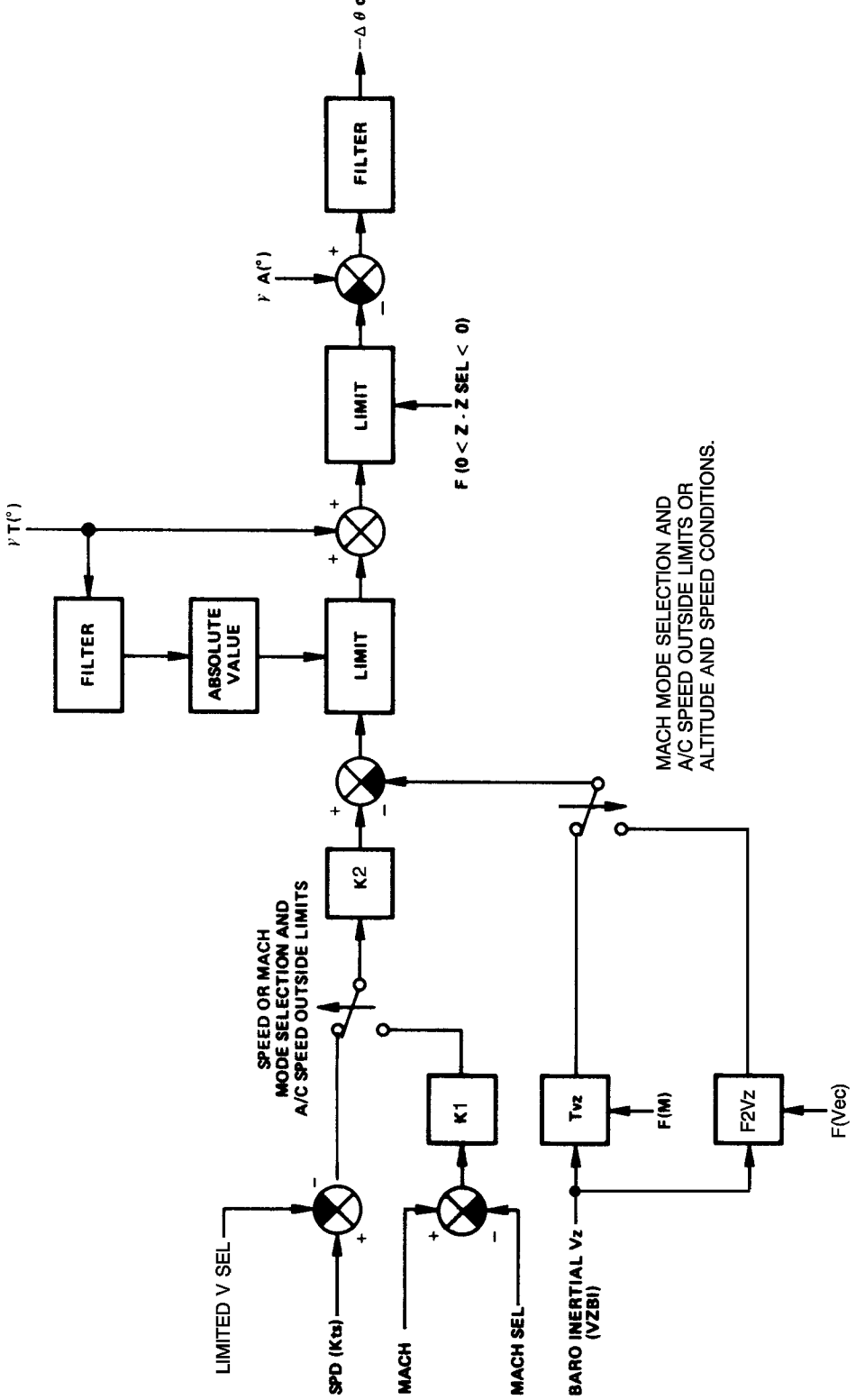
BM5 22 17 10 0 AZM2 14 08
DM5

LVL/CH Mode - Mode Selection when ALT SEL > Actual Altitude
Figure 012



BM5 22 17 10 0 BAM2 14 08
DM5

LVL/CH Mode - Mode Selection when ALT SEL < Actual Altitude
Figure 013



BM5 22 17 10 0 BCP2 00

LEVEL CHANGE Mode Computation
Figure 014

4. PROFILE Mode (Coupling to FMC)

A. Characteristics

The PROFILE mode enables AP/FD and autothrottle coupling to the FMC.
It fulfills the following functions :
- vertical navigation through the AP/FD
- engine thrust control through the autothrottle in relation with targets to be reached.

It is to be noted that the autothrottle can be engaged in PROFILE mode if at least one AP or one FD is engaged. On the contrary, the AP/FD can be engaged in PROFILE mode even if the autothrottle is not available.
The vertical navigation principle is described in the Flight Management Computer System (Ref. 34-63-00). The following description only deals with the PROFILE mode operation in conjunction with the AFS controls.

B. Operation

(1)Engagement/disengagement

The operational logic associated with the PROFILE mode engagement and disengagement is shown in figure (Ref. Fig. 016). The following phase indications (armed or active phases) : P.ALT, P.CLB, P.DES are transmitted directly to the FCC by the FMC (Ref. Fig. 017).
The SPD/MACH display on the FCU shows dashes to clearly indicate that the speed is no longer controlled through the speed selected on the FCU (although in all other modes, the selected speed is always used as a reference for the AP/FD or the autothrottle).
In addition the PROFILE mode engagement has the following results :
- TRP AUTO mode engagement, i.e. automatic management of the limit thrust computation modes by the FMC (CL, CR, etc...) (Ref. 22-30).
- indication of the thrust target in the TARGET display on the TRP. In the event of a TCC failure, the pilot can select display of the thrust computed by the FMC, the PROFILE mode being engaged for the AP/FD (Ref. Fig. 015).

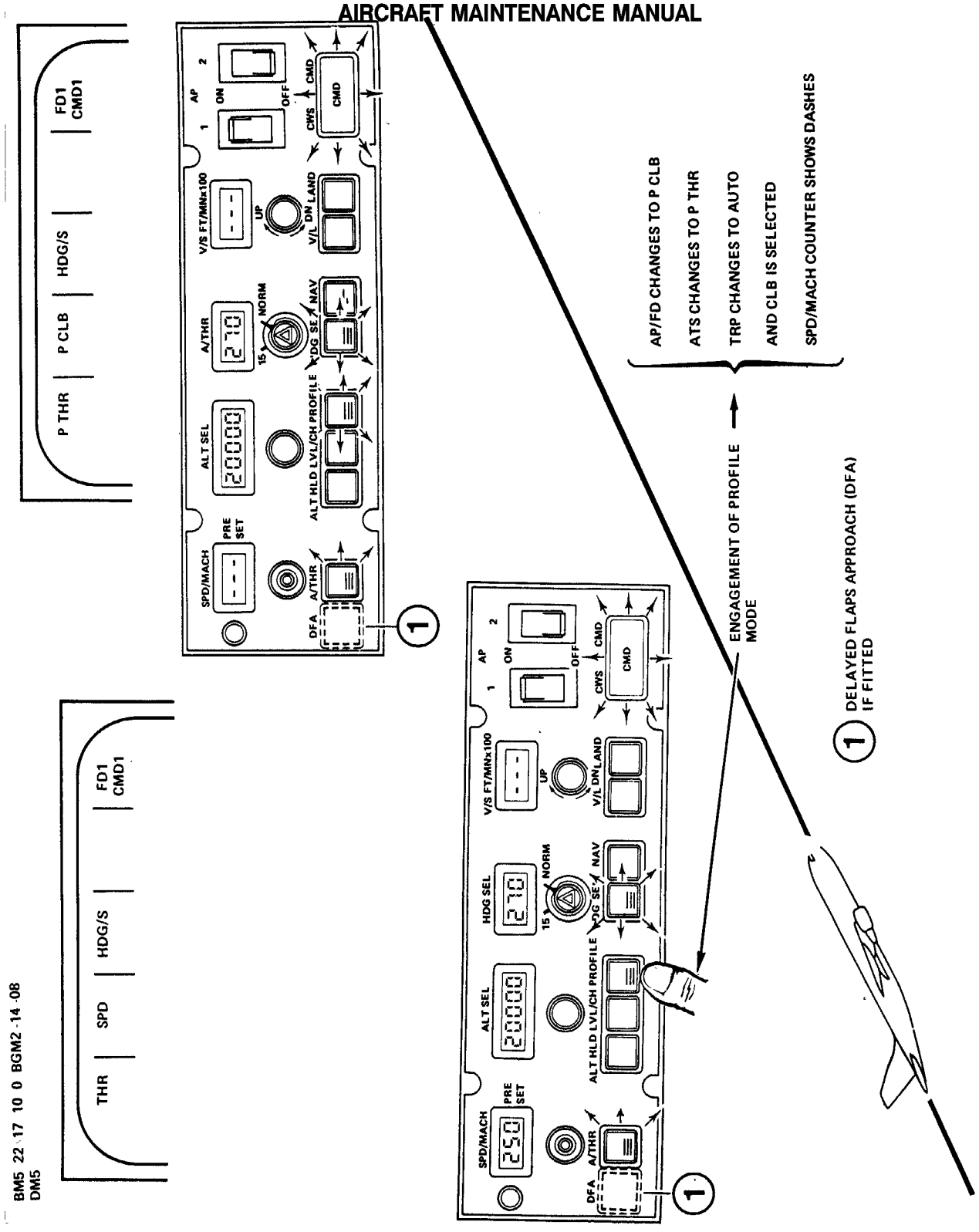
(2)Engagement during takeoff phases (Ref. Fig. 018)

The PROFILE mode can be used just after takeoff to carry out the noise abatement procedures or any other procedure associated with the takeoff airfield.
To this end, the PROFILE mode can be selected on the ground by pressing the PROFILE pushbutton switch.
For safety reasons, the PROFILE mode cannot be active before the aircraft reaches an altitude of 400 ft. and consequently selection of this mode results in its arming, during change to :

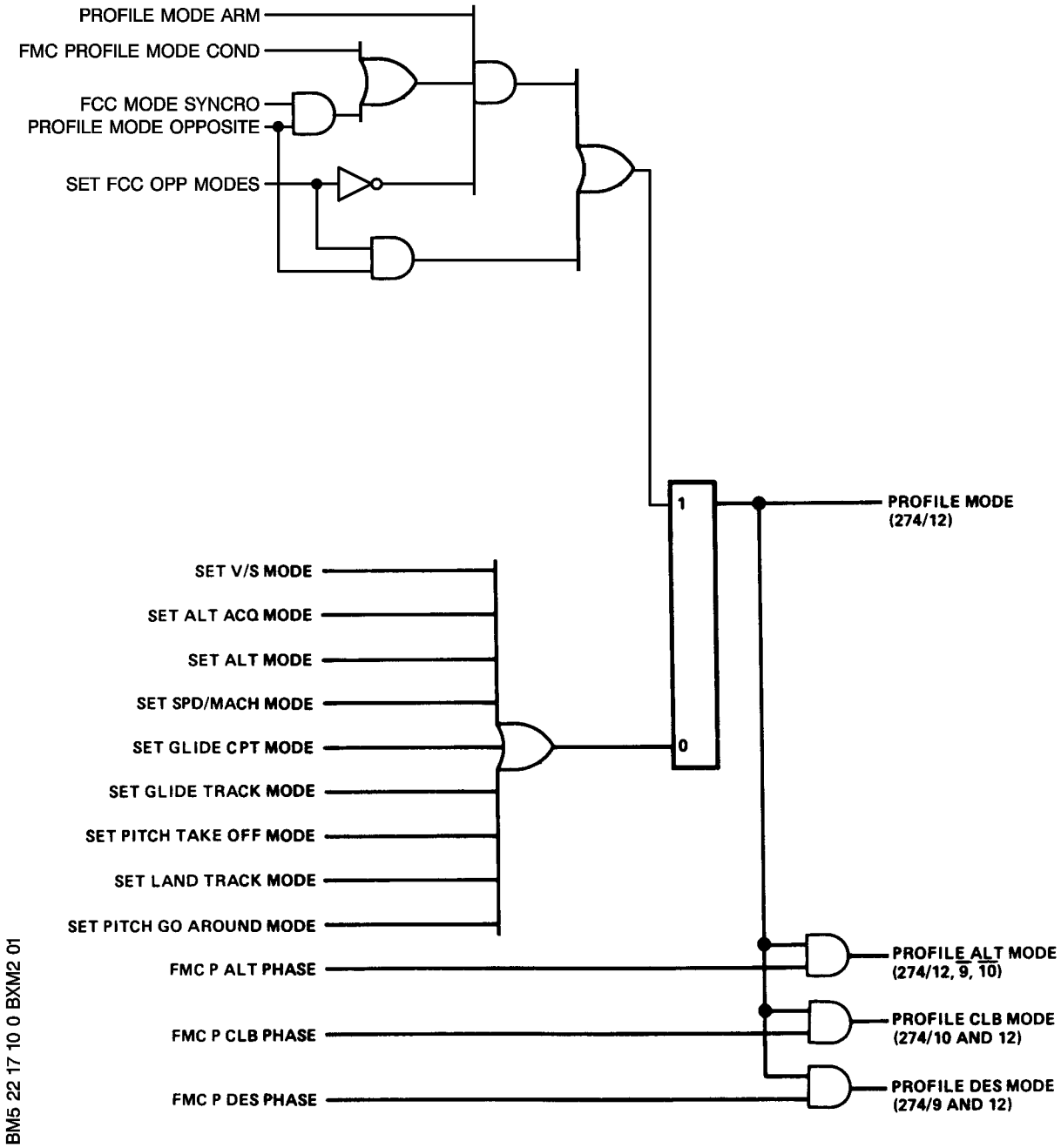
R - ALTITUDE PHASE 4 (value known by the FMC through the data base or through the CDU entry or 1500 ft. in the other cases).
The AUTO mode of the TRP becomes active and the autothrottle becomes active again as soon as a TRP mode other than TO or FLX TO is selected by the FMC.

R - ALTITUDE PHASE 5 (value known by the FMC through the data base or through the CDU entry or 2500 ft. in the other cases).
The PROFILE mode becomes active for the AP/FD.

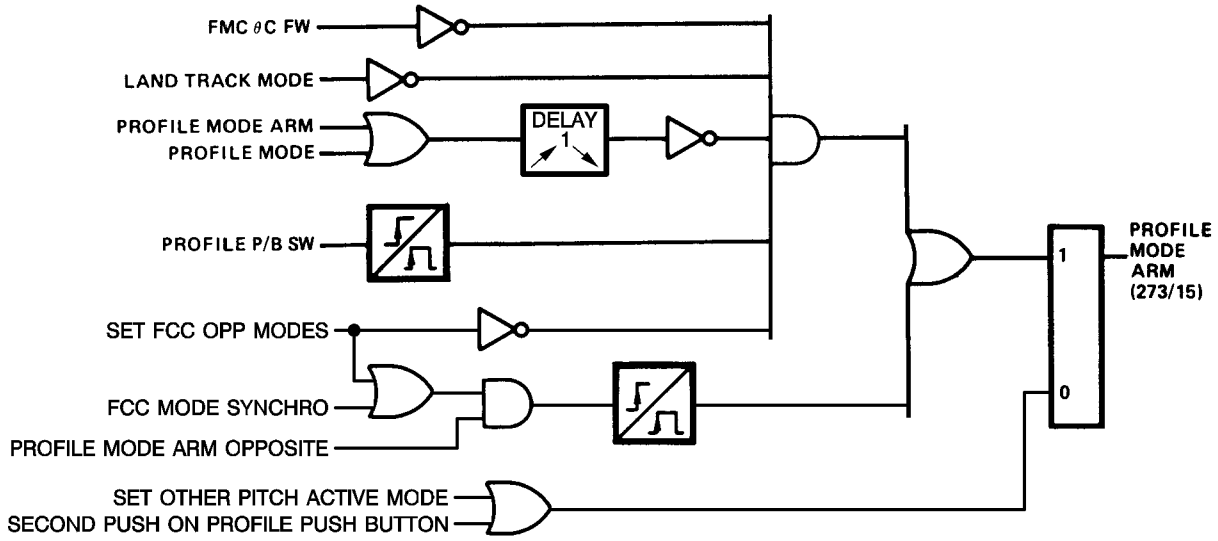
The type of law followed after activation of the PROFILE mode is described in the System Description Note : Flight Management System.

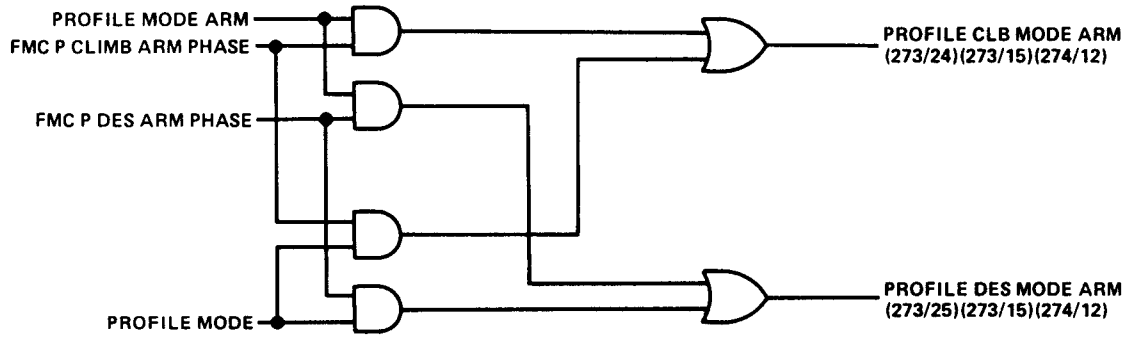


PROFILE Mode Engagement In Climb
Figure 015



PROFILE Mode-Logic
Figure 016




PROFILE MODE ARM
FMC P CLIMB ARM PHASE
FMC P DES ARM PHASE
PROFILE MODE
PROFILE CLB MODE ARM (273/24)(273/15)(274/12)
PROFILE DES MODE ARM (273/25)(273/15)(274/12)

BM5 22 17 10 0 BZM2 01

PROFILE Mode-Logic Arming Conditions
Figure 017

(3)Engagement in cruise mode (Ref. Fig. 019, 020)

The FMC can be used in cruise mode for all the flight phases :

- climb
- level flight
- descent

and can be used as a support mode for the LAND mode (GLIDE ARM phase).

It is to be noted that :

- the altitude displayed on the FCU is always used as a limit
- the FMC cannot perform any level change without initial action by the pilot.

To perform a level change the pilot must :

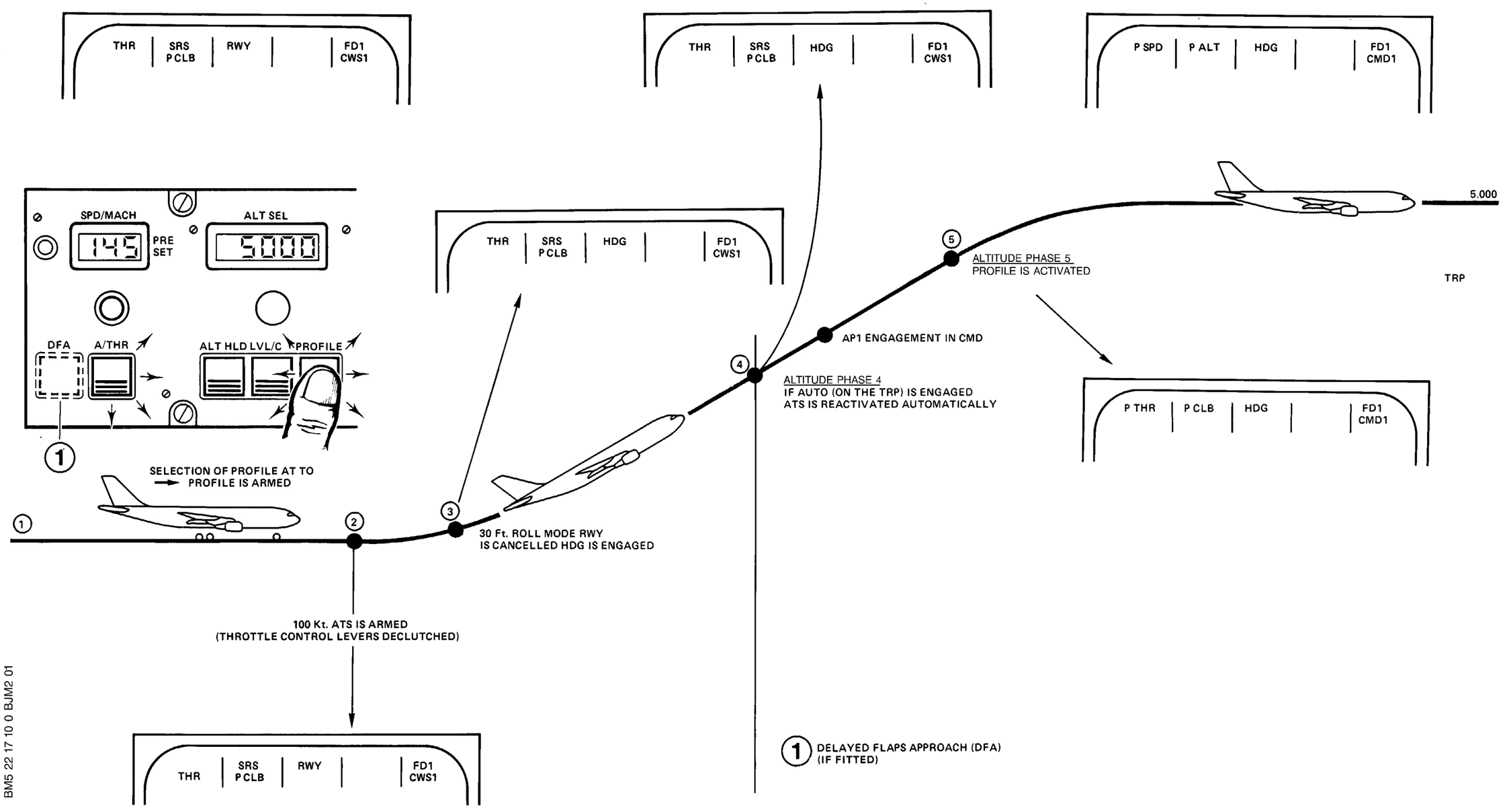
- select the new level on the FCU
- pull the altitude selector knob to authorize acquisition of this level.

After the pilot has pulled the altitude selector knob the new level capture can be :

- either immediate (e.g. in climb configuration)
- or delayed, in particular in descent.
  The FMC will hold the level flight until the top of descent (T/D) is reached.

(4)Engagement in approach

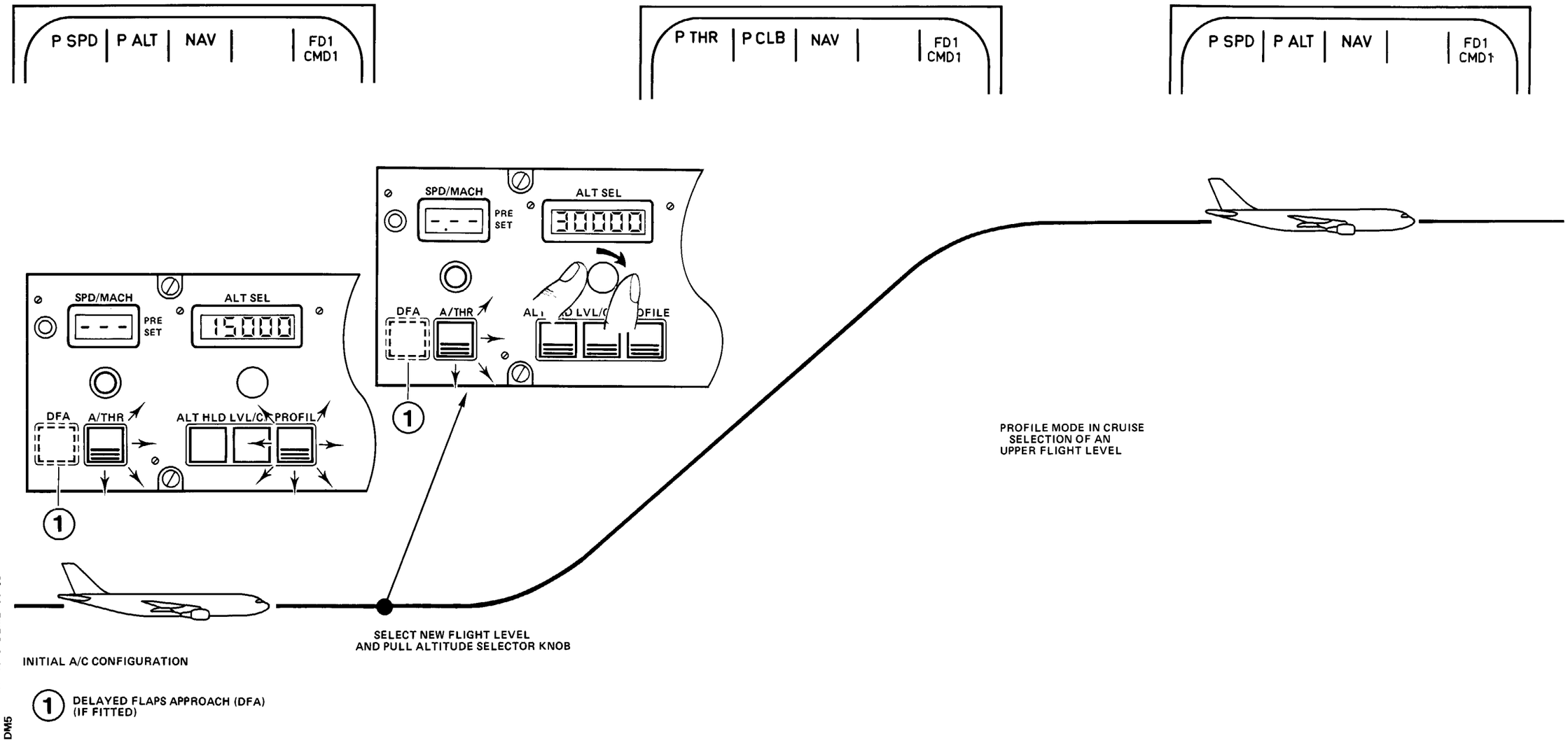
Engagement in approach is the same as engagement in cruise. It is to be noted that, if the pilot decides to go down to the minimum descent altitude (MDA) using the FMC, the FCC sends an altitude alert inhibition signal to the FWC.



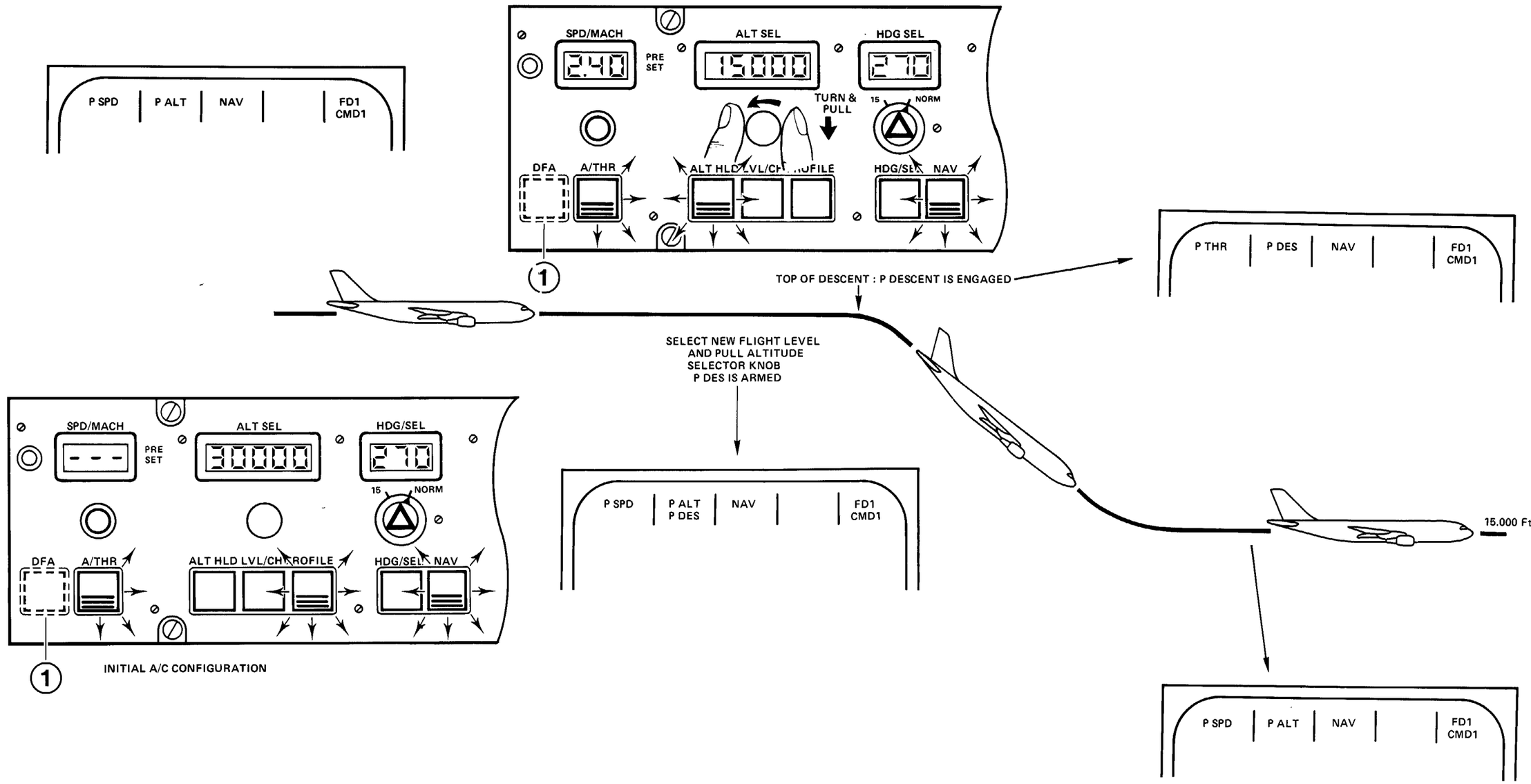
BM5 22 17 10 0 BJM2 01

PROFILE Engagement at Takeoff
Figure 018

R



PROFILE Mode In Cruise - Selection of an Upper Flight Level
Figure 019



BM5 22 17 10 0 BNM2 14-08
DM5

1 DELAYED FLAPS APPROACH (DFA)
(IF FITTED)

PROFILE Mode In Cruise - Selection of a Lower Flight Level
Figure 020

(5)Introduction of speed/Mach and vertical speed constraints
Introduction of Speed/Mach and vertical speed constraints is achieved by the pilot on the corresponding displays of the FCU.
To enter a vertical speed constraint (the V/S display shows dashes when the PROFILE mode is engaged), the pilot must :
- turn the vertical speed selector knob.
  This action results in synchronizing the V/S display with the aircraft vertical speed and authorizing selection of the desired vertical speed.
- engage the V/S mode, by pulling the vertical speed selector knob.

NOTE 1 : If the V/S mode is not engaged, the V/S display shows dashes again 10 seconds after the vertical speed is displayed since the PROFILE mode is still active.

NOTE 2 : Introduction of the vertical speed constraint might have been achieved by :
- returning to V/S mode through second action on the PROFILE pushbutton switch and display of the desired vertical speed. However it is obvious that this action is less rapid and less evident than the action described above (Ref. Fig. 021).

To enter a speed or Mach constraint (the SPD/MACH display shows dashes when the PROFILE mode is engaged) the pilot must :
- turn the speed or Mach selector knob.
  This action results in synchronizing the SPD/MACH display with the aircraft speed and in authorizing selection of the desired speed
- pull the speed or Mach selector knob. This action results in engaging appropriate mode to enable displayed speed constraint to be held :
- in level flight : the AP/FD engages in ALT mode and the autothrottle in SPD/MACH mode
- in climb or in descent configuration : the AP/FD engages in LVL/CH mode and the autothrottle in THR or RETARD mode depending on the configuration (climb or descent).

NOTE 1 : If the selector knob is not pulled, the SPD/MACH display shows dashes again 10 seconds after selection of the speed or Mach value since the PROFILE mode is still active.

NOTE 2 : It is possible to introduce a speed or Mach constraint by using a different procedure requiring :
- engagement of LVL/CH mode in climb or descent configuration
- engagement of V/S mode in level flight (equivalent to ALT engagement)
- selection of speed or Mach value on the SPD/MACH counter of the FCU.

Here again, it is obvious that this action is less rapid and less evident than the action described above.

(Ref. Fig. 022)
(Ref. Fig. 023)

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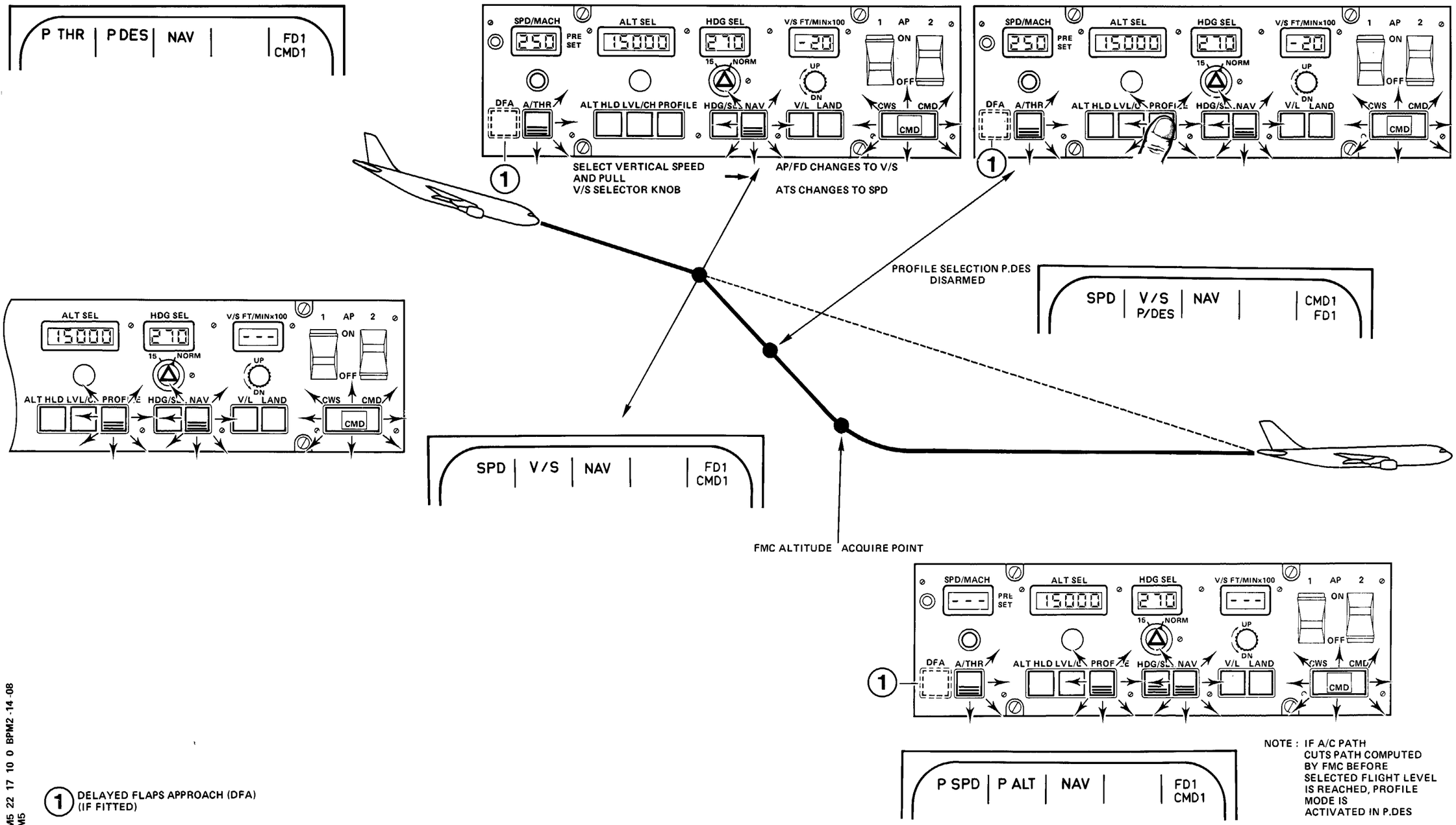

BM5 22 17 10 0 BPM2 -14 -08
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(1) DELAYED FLAPS APPROACH (DFA) (IF FITTED)

Introduction of V/S Constraint
Figure 021

EFFECTIVITY: ALL

R

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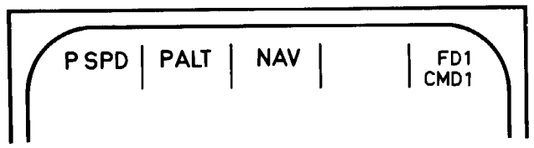
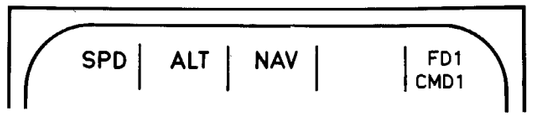
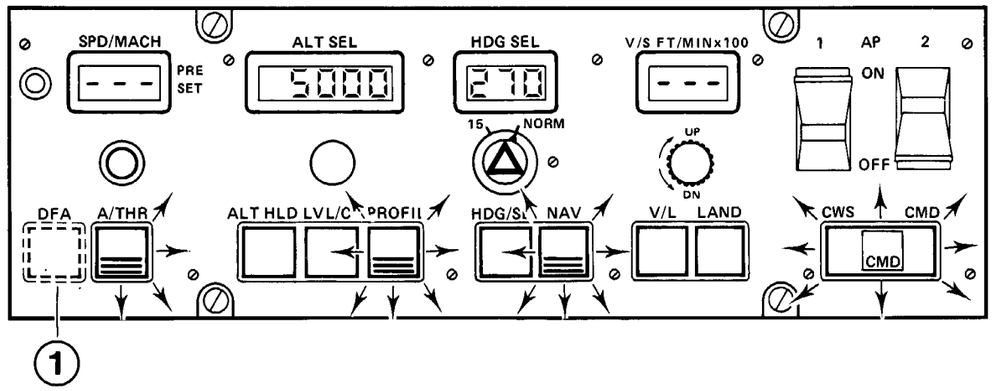
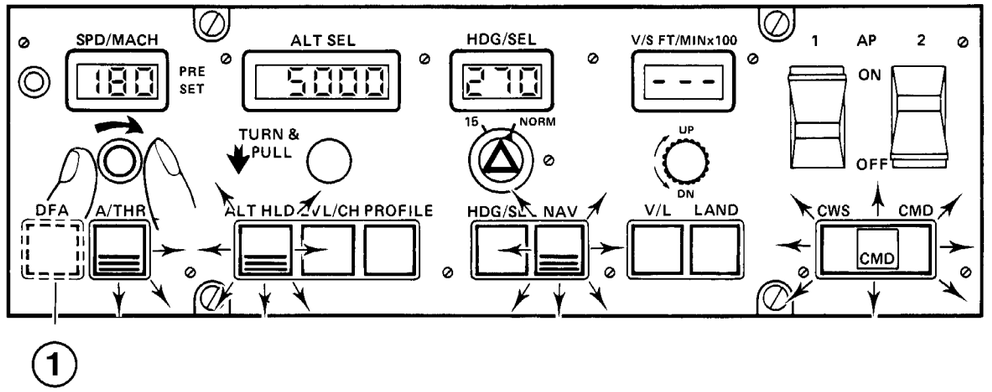
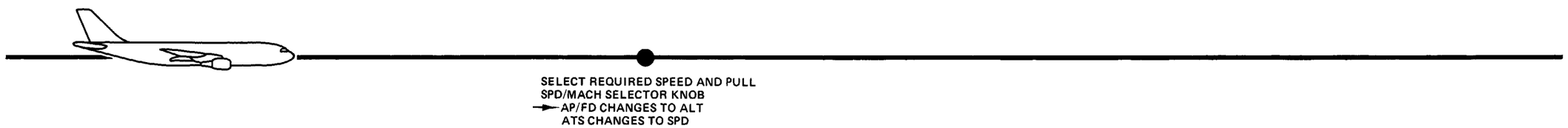








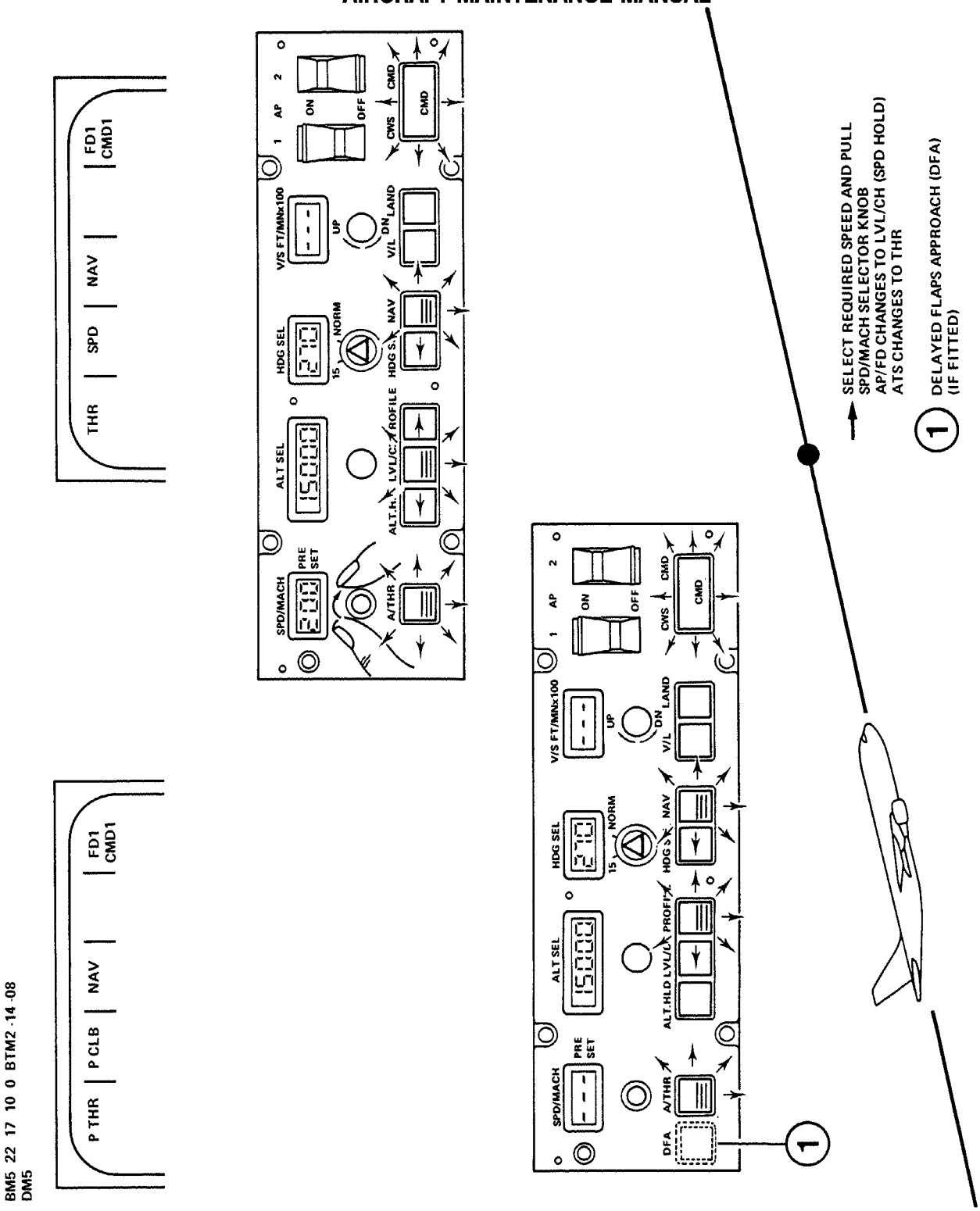


INITIAL A/C CONFIGURATION

(1) DELAYED FLAPS APPROACH (DFA) (IF FITTED)

BM5 22 17 10 0 BRM2-14-08
DM5

Introduction of a Speed Constraint in Flight Level when PROFILE is Engaged
Figure 022



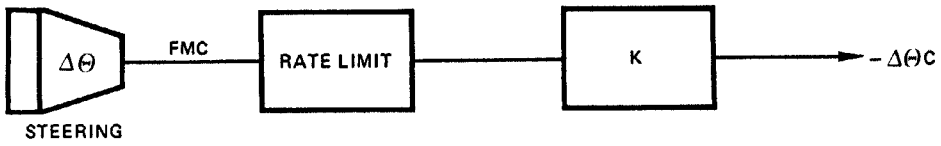
BM5 22 17 10 0 BTM2 14 08
DM5

Introduction of a Speed Constraint in Climb when PROFILE is Engaged
Figure 023

C. Command generation
The computation principle of the AP/FD commands in PROFILE mode is shown in figure (Ref. Fig. 024).
In fact, the FMC sends a steering signal to the FCC which only shapes the signal transmitted by the FMC.

D. Display
Selection of the PROFILE mode is indicated :
- on the FCU by illumination of the PROFILE mode selection pushbutton switch
- on the PFD :
  . in active phase, by illumination of the following green indications :
    P ALT in level flight
    P CLB in climbing phase
    P DES in descent phase
  . in armed phase, by illumination of the following cyan indications :
    P CLB : PROFILE mode arming at takeoff or following a level change after having pulled the altitude selector knob on the FCU
    P DES : descent phase arming after having selected a level lower than aircraft level and pulled the altitude selector knob.

The cyan P DES indication appears at the same time as the green P ALT indication. The cyan P CLB indication appears independently of or at the same time as the green P ALT indication.



BM5 22 17 10 0 BVM2 14
DM5

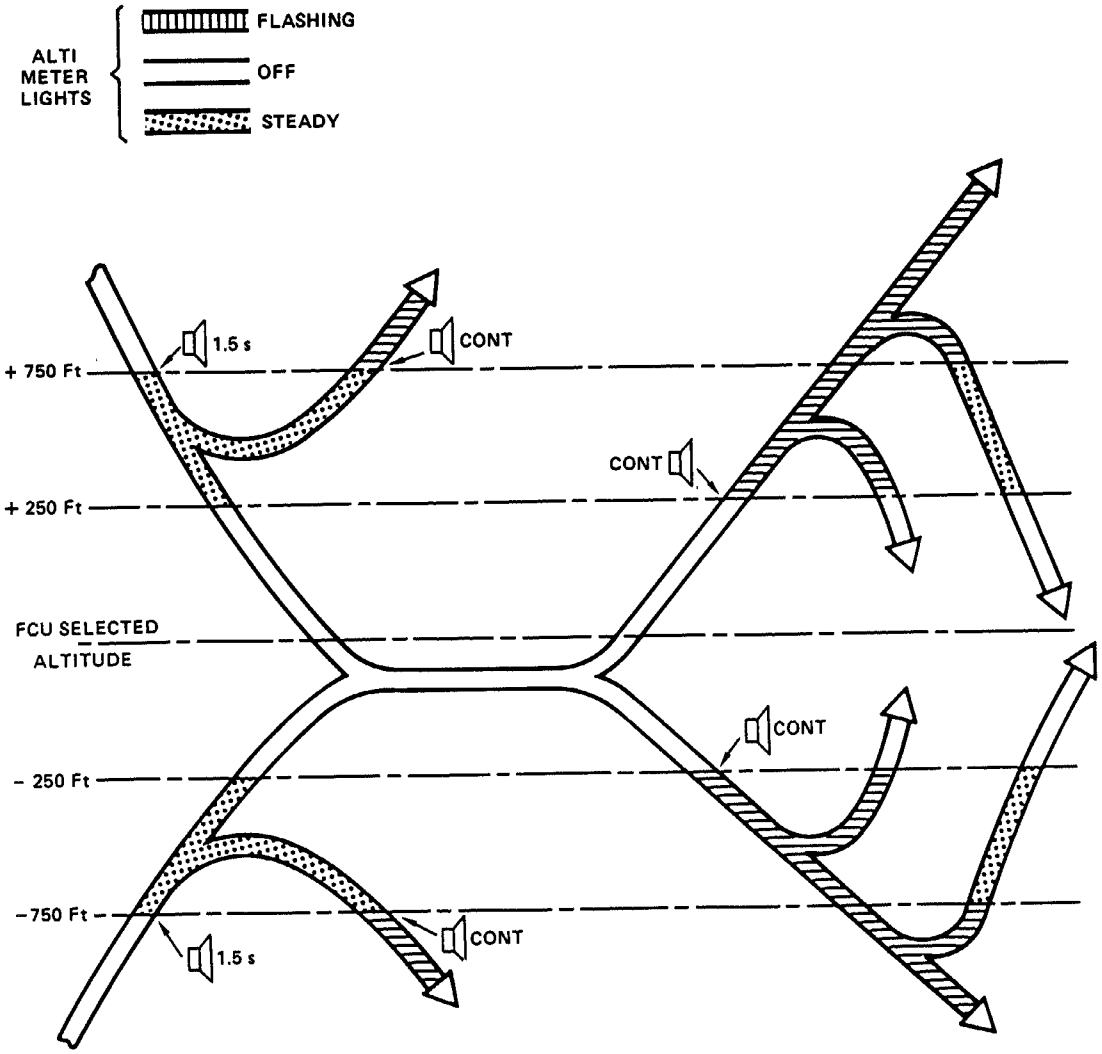
PROFILE Mode Computation
Figure 024

5. Altitude Alert

The Flight Warning Computers generate an altitude alert warning (**ALTITUDE ALERT**) from altitude data corresponding to the value displayed on the **FCU**.
This warning can be activated in manual or automatic flight.
This warning consists in an aural and visual warning (amber warning light on Captain and First Officer altimeters).
Warning is inhibited :

- during selection of a new altitude on the **FCU**
- at glide capture or track
- at **LAND** track
- when **A/C** is not in clean configuration and landing gear down
- when **PROFILE** mode is engaged with one **AP** in **CMD** or **FD** and altitude alert inhibition from **FMC** active.

The activation conditions for this warning are shown in following figure. (Ref. Fig. 025)



1.5 S C.CHORD IS INHIBITED WHEN AP IS IN CMD
CONTINUOUS C.CHORD IS CANCELLED BY NEW ALTITUDE SELECTION
OR BY EMERGENCY AUDIO CANCEL.FLASHING LIGHT GOES OFF
WITH NEW ALTITUDE SELECTION.

BM5 22 17 10 0 HTM2 14 08
DM5

Altitude Alert
Figure 025

1. Heading Selection Hold Mode (HDG SEL)

A. Characteristics

This mode acquires and holds the heading selected in the HDG SEL display on the FCU (Ref. Fig. 001).

B. Operation

(1) Engagement/disengagement

The engage and disengage conditions of the HDG SEL mode are given in figure (Ref. Fig. 003).

C. Command generation

The generation principle of the AP/FD commands in HDG SEL mode is shown in figure (Ref. Fig. 002).

D. Display

The mode engagement is indicated by :

- illumination of the HDG SEL pushbutton switch on the FCU
- illumination of the cyan HDG/S indication on the PFDs in the arming phase of the mode at takeoff
- illumination of the green HDG/S indication on the PFDs when the mode is active.

In addition, the selected heading information is repeated on the Navigation Displays (ND).

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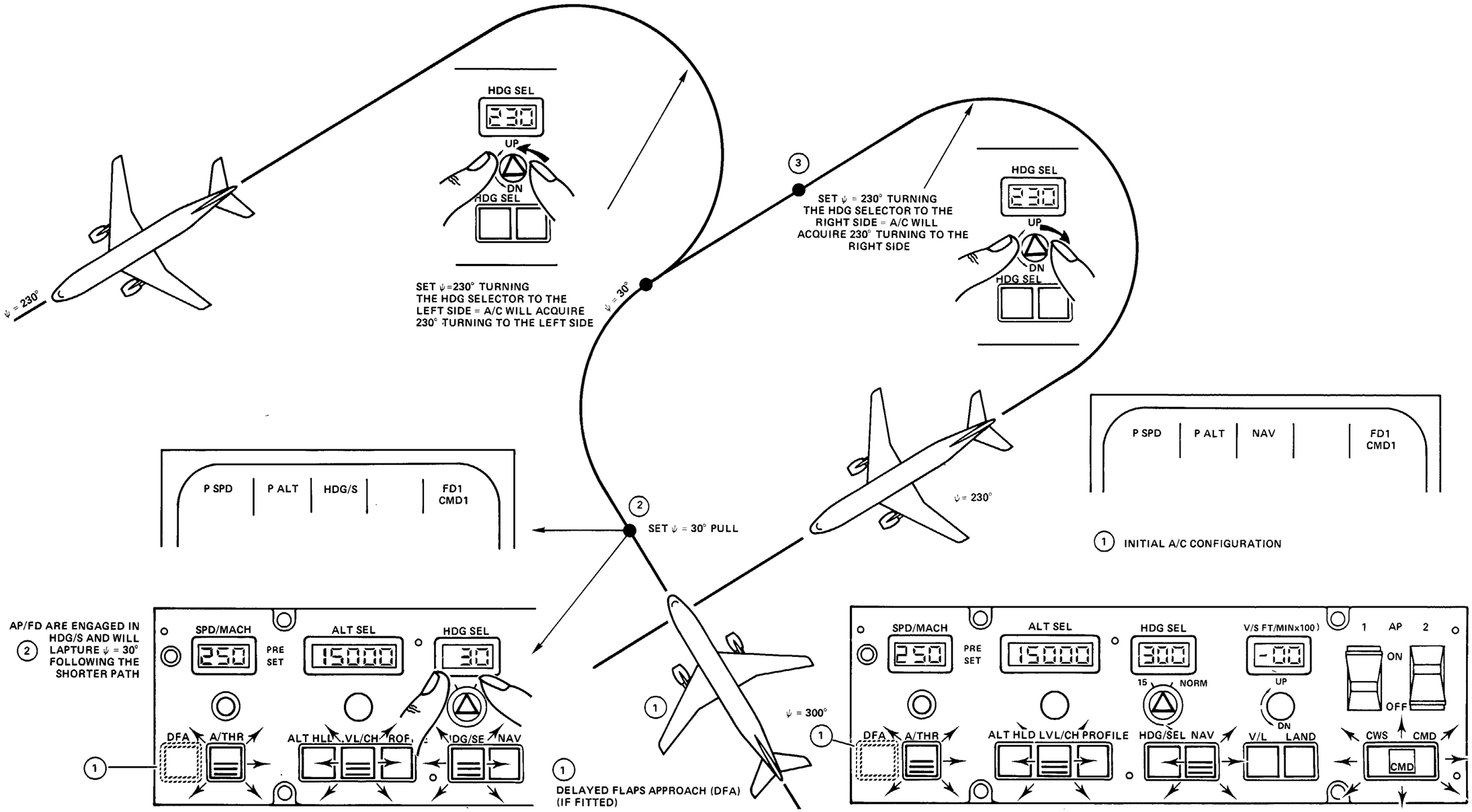

HDG SEL Mode
Figure 001

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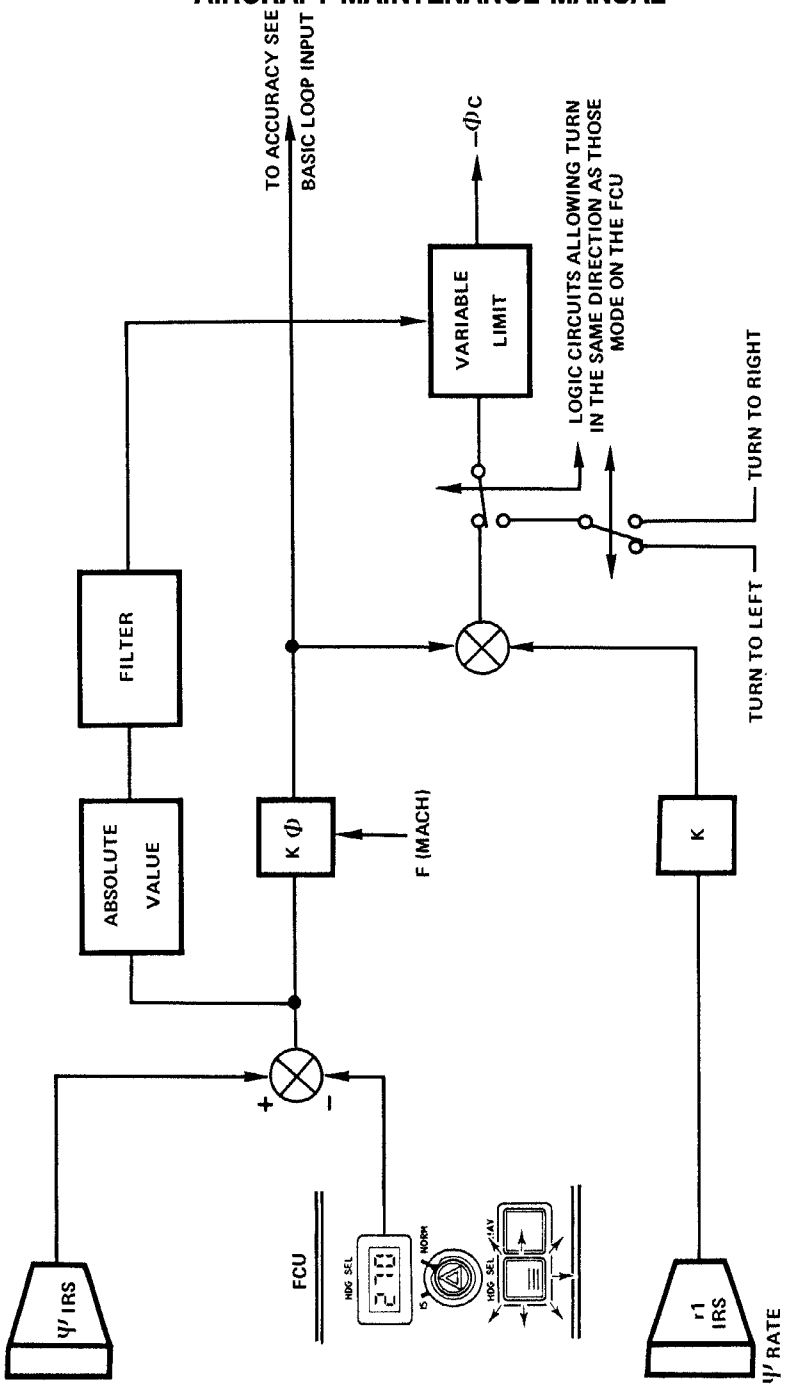

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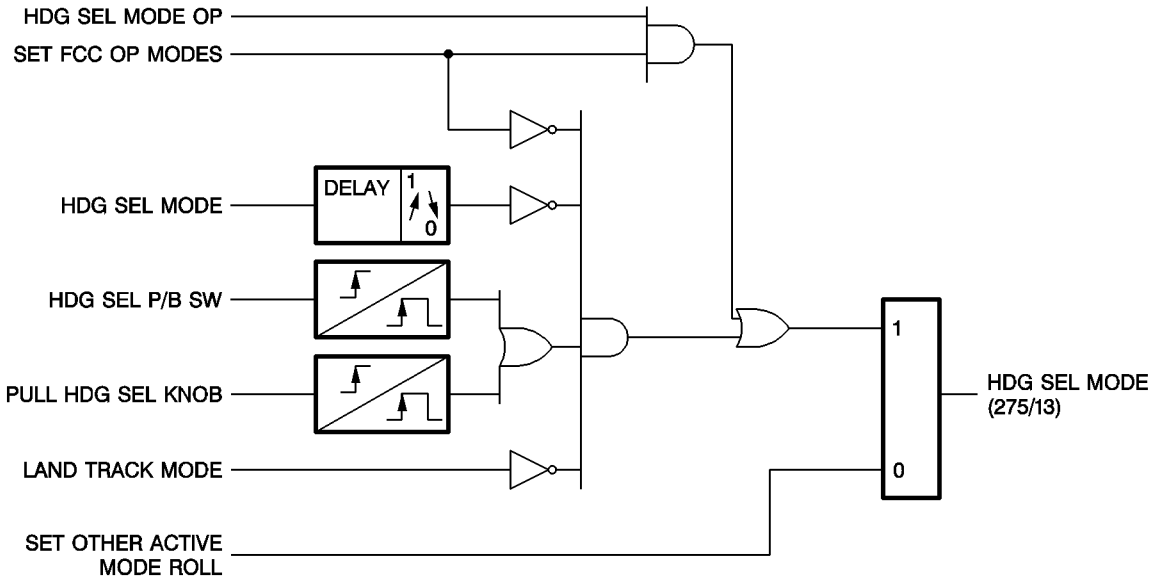
HDG SEL Mode Computation
Figure 002



BM5 22 17 20 0 CEM2 01

HDG SEL Mode Logic
Figure 003

R

2. Navigation Mode (NAV)

A. Characteristics

The NAV mode enables the aircraft to be controlled in the horizontal plane through the AP/FD coupling to the FMC.
The possibilities of navigation in horizontal plane are described in System Description Note : Flight Management Computer System.
The following description only deals with the NAV mode operation in conjunction with the AFS controls.

B. Operation

(1)Engagement/disengagement

The arming and engage conditions of the NAV mode are given in figures (Ref. Fig. 007, 008)
This mode includes :
- an arming phase, the support modes can be the HDG basic mode, HDG SEL mode, RWY takeoff mode, VOR mode in active phase.

(2)Engagement at takeoff

Selection of NAV mode on the ground results in its arming. Effective engagement of this mode will occur after an altitude of 30 ft. has been reached (Ref. Fig. 005).

C. Command generation

The computation principle of the AP/FD commands in NAV mode is given in figure (Ref. Fig. 006).
In fact the FMC sends a steering signal to the FCC which only shapes this information signal.

D. Display

Selection of the NAV mode is indicated :
- on the FCU by illumination of the NAV mode selection pushbutton switch
- on the PFD by :
  . illumination of the cyan NAV indication in arming phase
  . illumination of the green NAV indication in active phase.

During the arming phase, the cyan NAV indication is accompanied by the green indication of the associated support mode (Ref. Fig. 004, 005).

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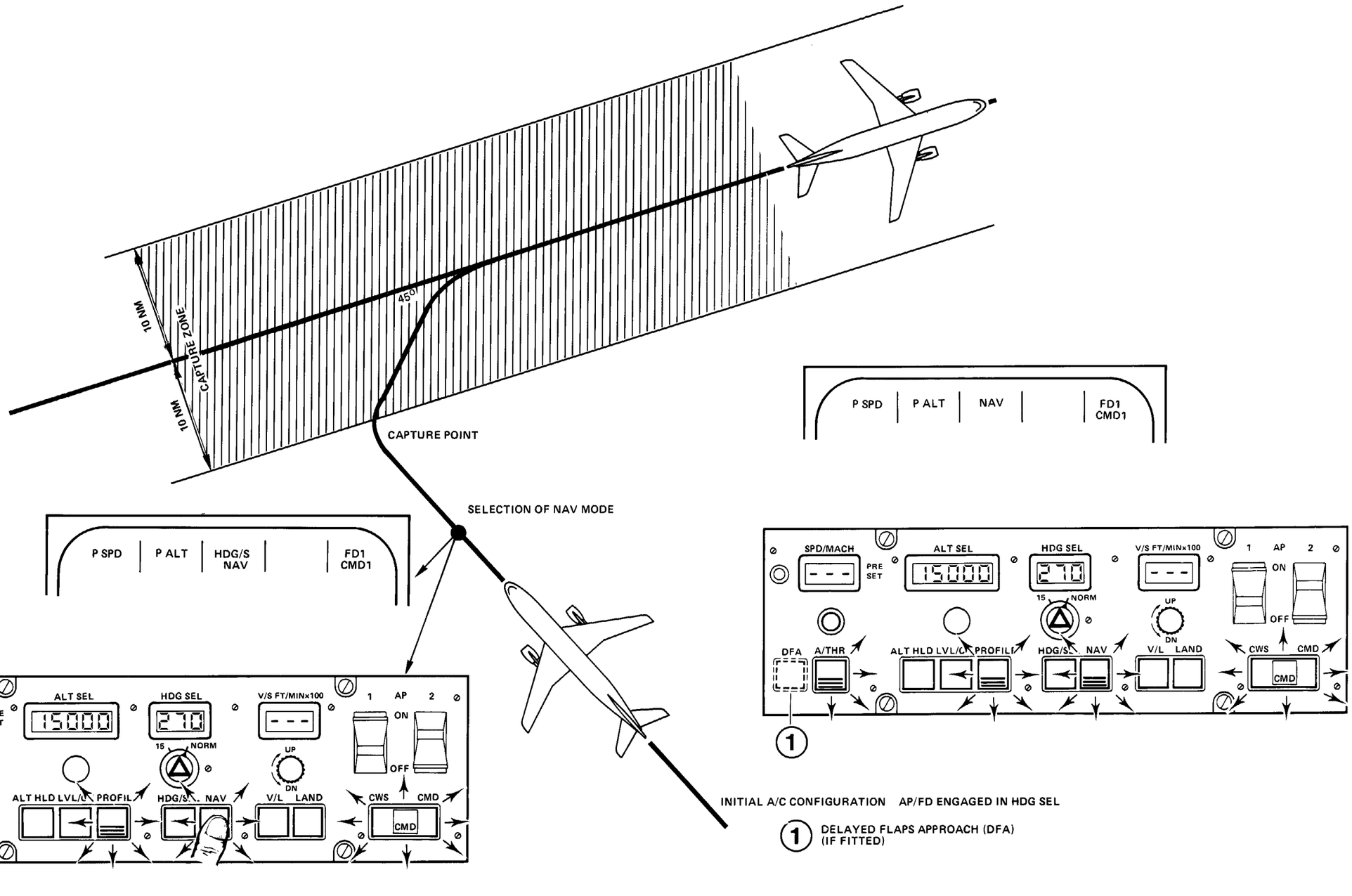

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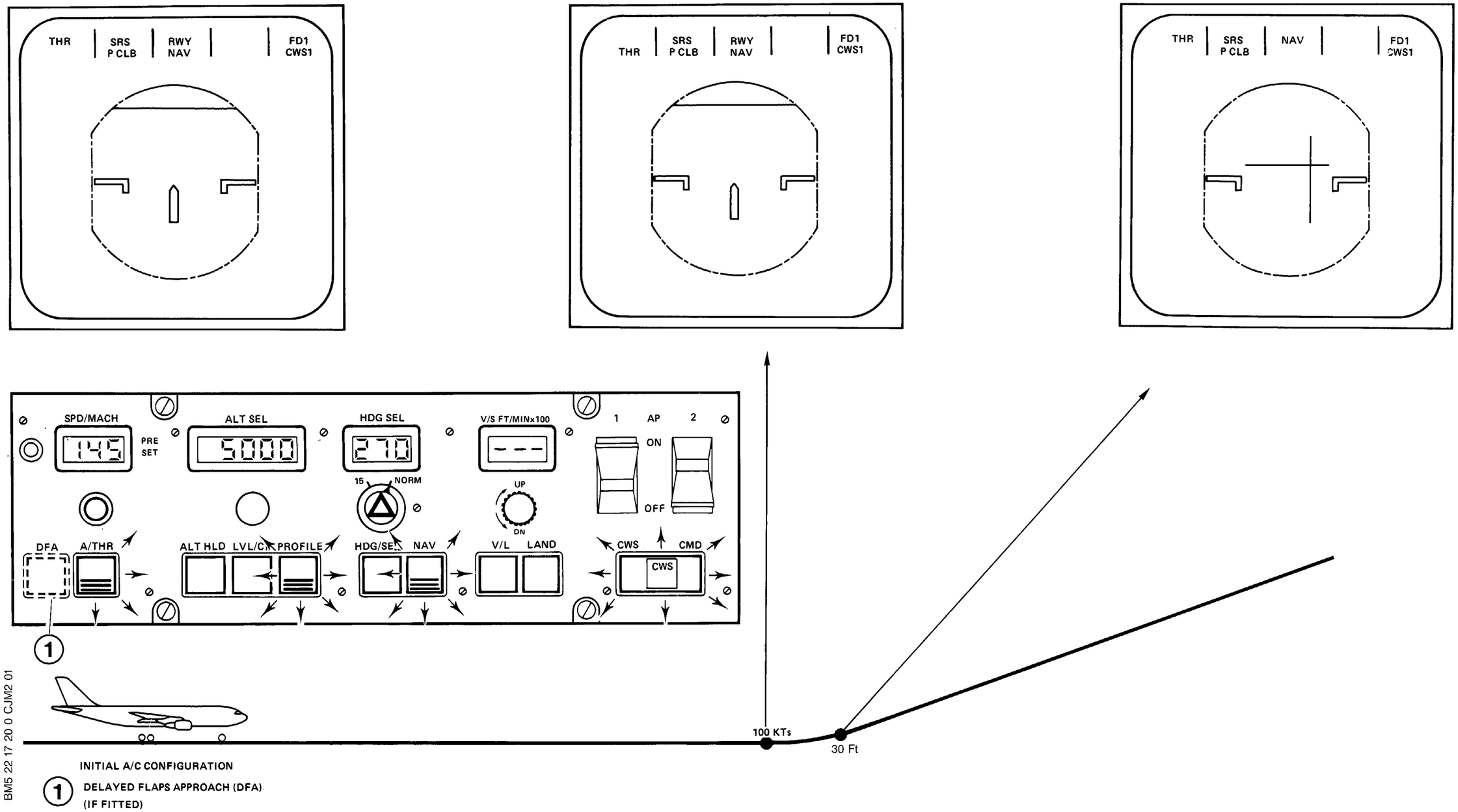
NAV Mode Engagement
Figure 004



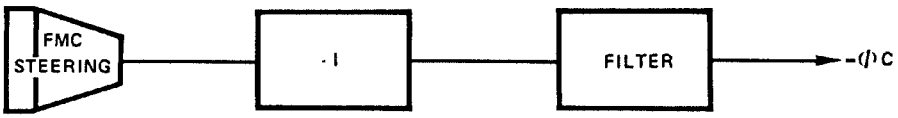
INITIAL A/C CONFIGURATION

(1) DELAYED FLAPS APPROACH (DFA) (IF FITTED)

BM5 22 17 20 0 CJM2 01

NAV Mode Engagement at Takeoff
Figure 005

R



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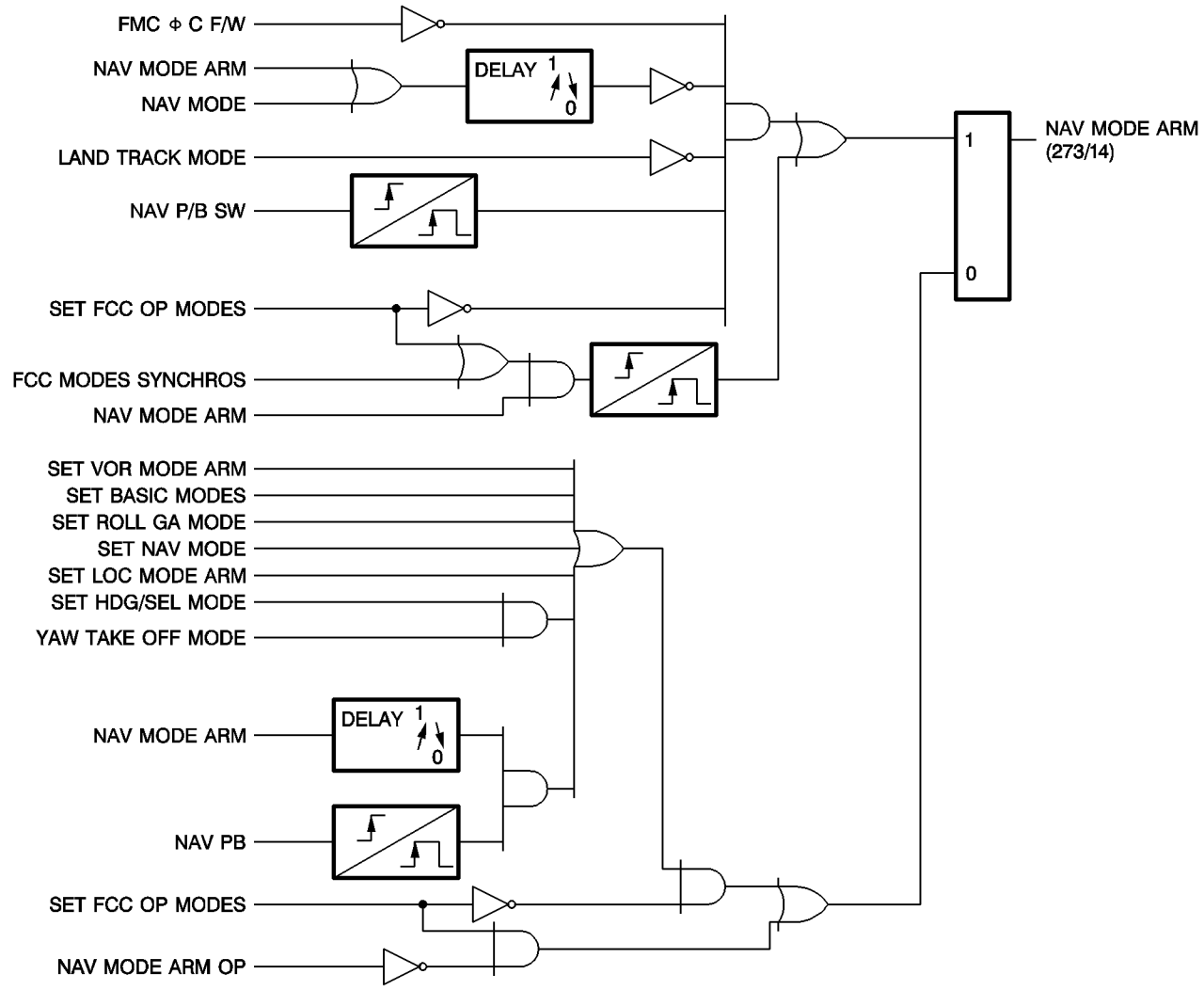
NAV Mode Computation
Figure 006



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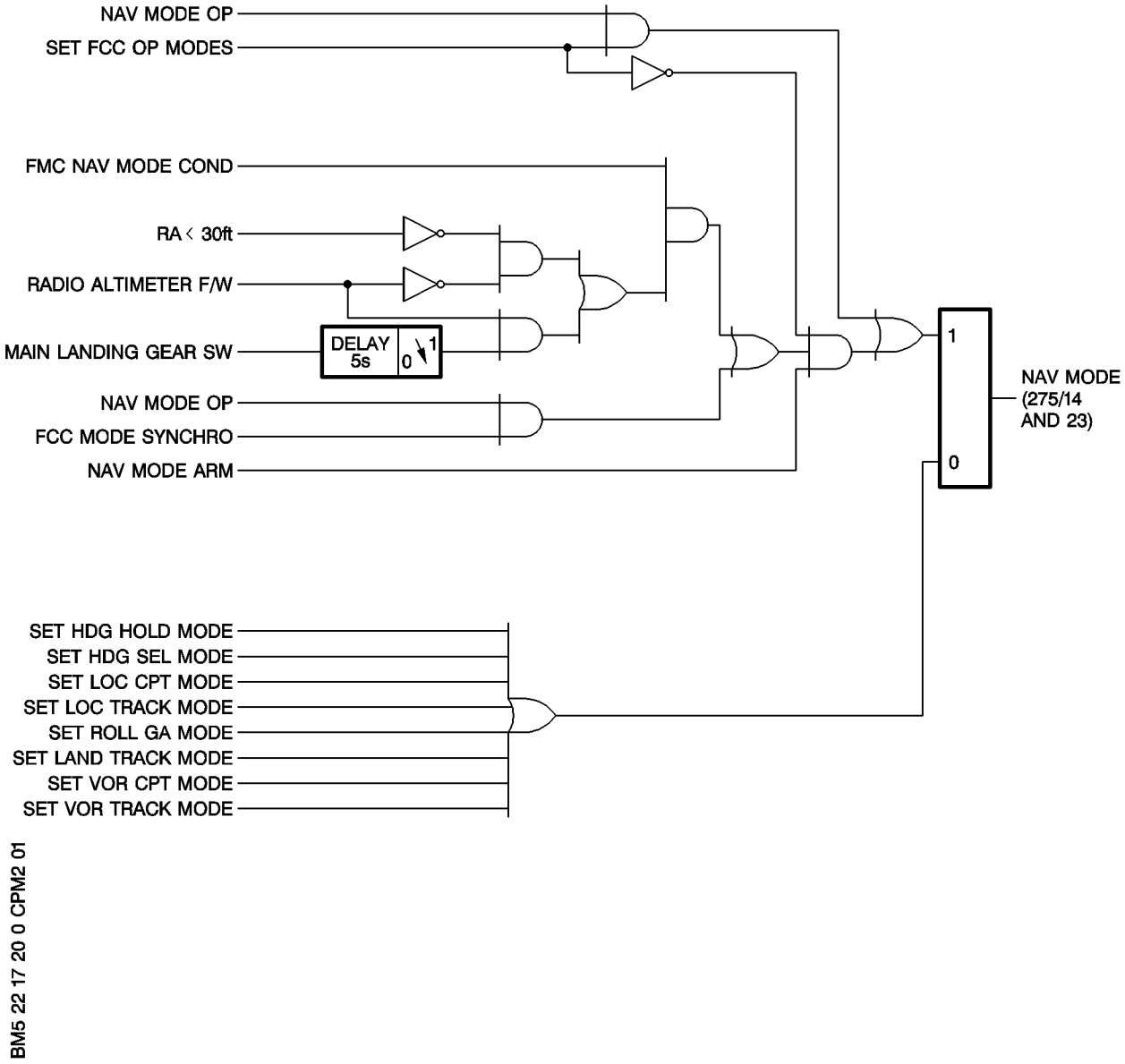
NAV Mode Logic Arming Conditions
Figure 007

R



NAV Mode Logic Engagement Conditions
Figure 008

R

3. VOR Mode

A. Characteristics

The VOR mode captures and holds the VOR beam. It also allows flying over the VOR station (passing through the confusion cone) following the arrival course or a modified course.

This mode includes :

. an arming phase
. a capture phase
. a track phase.

During the arming phase, the HDG, HDG SEL and NAV modes can be used as support modes.

During the capture and track phases, the CWS is available (supervisory override function).

B. Operation (Ref. Fig. 009)

(1) Engagement/disengagement

The arming capture and track conditions of the VOR mode are given in figures (Ref. Fig. 011, 012, 013)

Engagement of : both FDs
both FDs and one AP in CMD
both APs in CMD (with LAND selected)

and the fact of having one pushbutton switch common to VOR and LOC mode engagement have the following consequences :

(2) Simultaneous engagement of both FDs

- before arming, if one VOR/NAV/ILS mode selector switch is placed in VOR position and the other is selected LOC (ILS position), priority is given to the LOC mode when V/L is selected. Consequently, the FD associated with the selector switch in VOR position, disengages. If LOC is selected, the FD will then engage in LOC mode.
- with VOR mode armed or active for both FDs and if LOC is selected (ILS position) on the VOR/NAV/ILS mode selector switch, the associated FD disengages (a 5 second delay is provided).

Before arming, if the two VOR/NAV/ILS mode selector switches are in NAV position, V/L selection causes no change.

But if one VOR/NAV/ILS mode selector switch is in VOR position and the other in NAV position, at V/L selection VOR mode is armed on the two FDs.

(3) Engagement of one AP in CMD and both FDs

Priority is given to AP engagement.

Before the V/L mode is selected, if the VOR/NAV/ILS mode selector switch associated with the AP is in VOR position and if the opposite VOR/NAV/ILS mode selector switch is selected LOC (ILS position), at V/L selection AP (and associated FD) engages in VOR mode and opposite FD disengages (it will engage again in VOR mode if VOR is selected by means of the VOR/NAV/ILS mode selector switch) and vice versa.

After engagement of the VOR mode and with LOC selected by means of the VOR/NAV/ILS mode selector switch, the associated systems disengage (a 5-second delay is provided).

(4)Engagement of both APs in CMD
The VOR mode being active (and LAND mode armed), LOC selection on one VOR/NAV/ILS mode selector switch causes disengagement of the system associated with the selector switch.
The LAND mode being armed, action on the V/L pushbutton switch is not taken into account if the VOR/NAV/ILS mode selector switches are in VOR position.
The LAND mode being armed, action on the V/L pushbutton switch is taken into account if the VOR/NAV/ILS mode selector switches are selected LOC (ILS position). If a VOR/NAV/ILS mode selector switch is in VOR position, the associated AP/FD disengages after action on the V/L pushbutton switch. The logic conditions associated with AP/FD engagement in V/L mode are given in 22-16, paragraph 2.C.(1).

C. Command generation
The computation principle of the guidance law in VOR mode is given in figure (Ref. Fig. 010).

D. Display
- selection of the VOR mode is indicated on the FCU by illumination of the V/L mode selection pushbutton switch.
- arming of the VOR mode is indicated on the PFD by illumination of the cyan VOR indication.
- engagement of the capture phase is indicated on the PFD by illumination of the green VOR* indication.
- change to the track phase is indicated on the PFD by illumination of the green VOR indication.

It is to be noted that during the arming phase the cyan VOR indication is accompanied by a green indication representing the engaged support. (Ref. Fig. 009)

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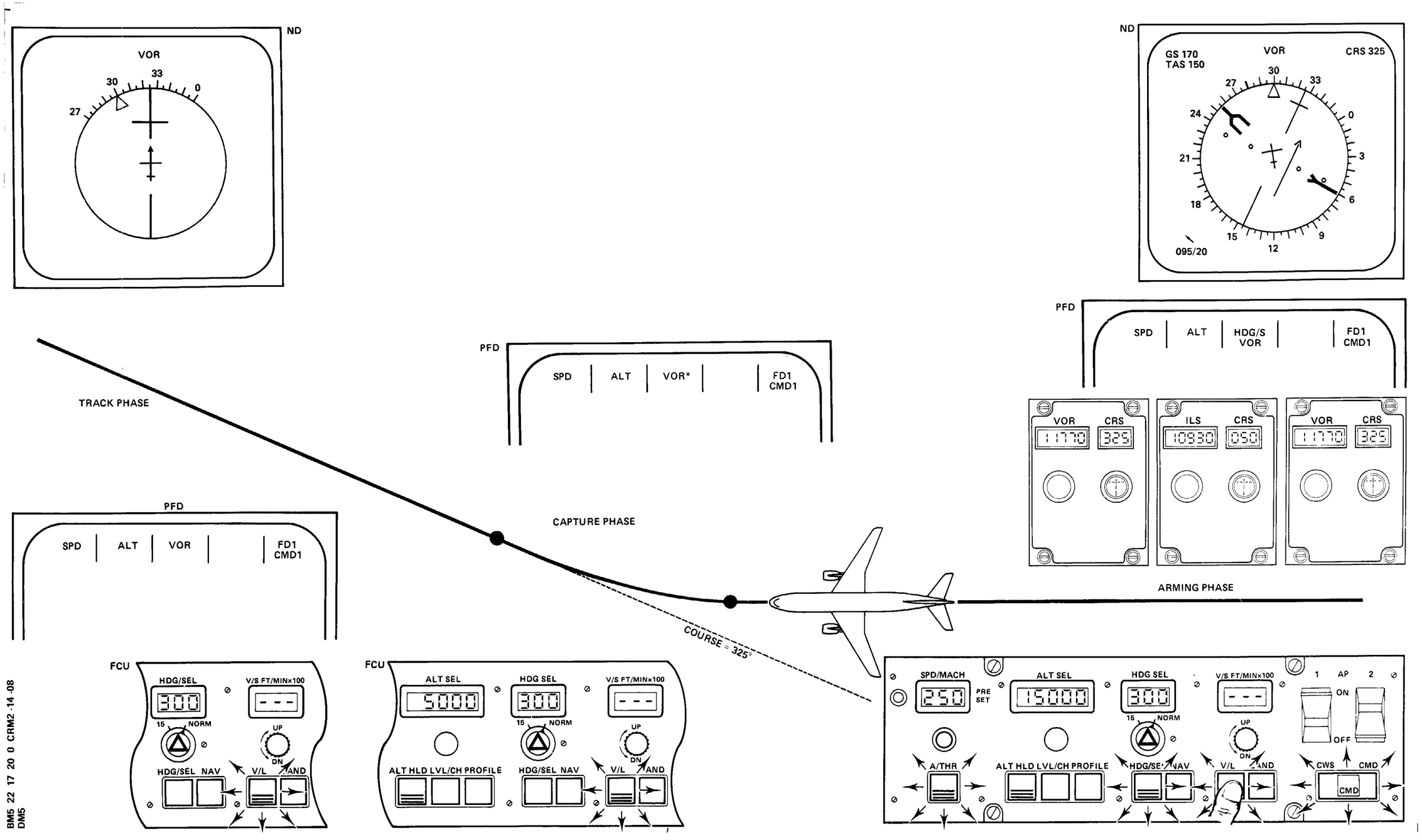

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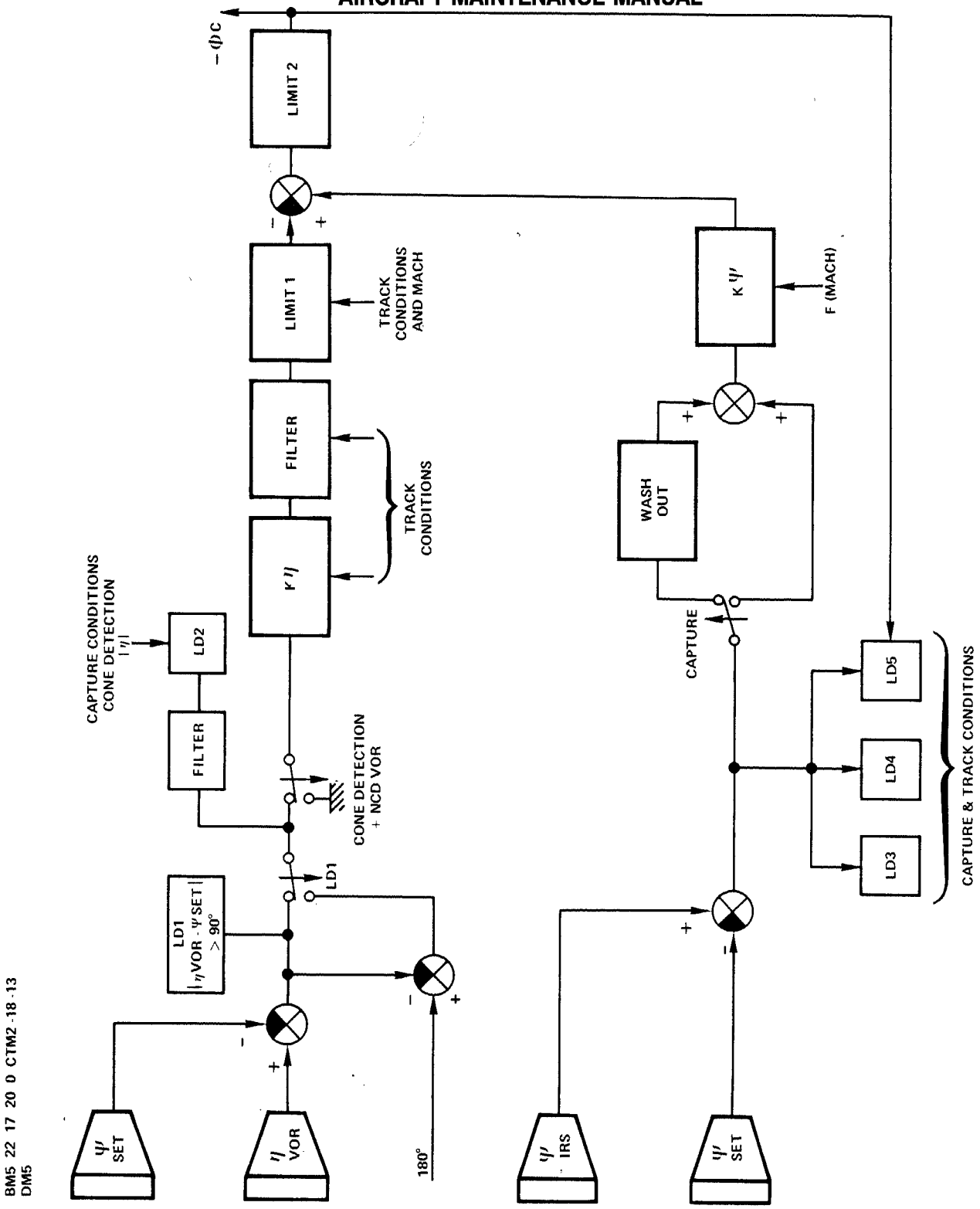
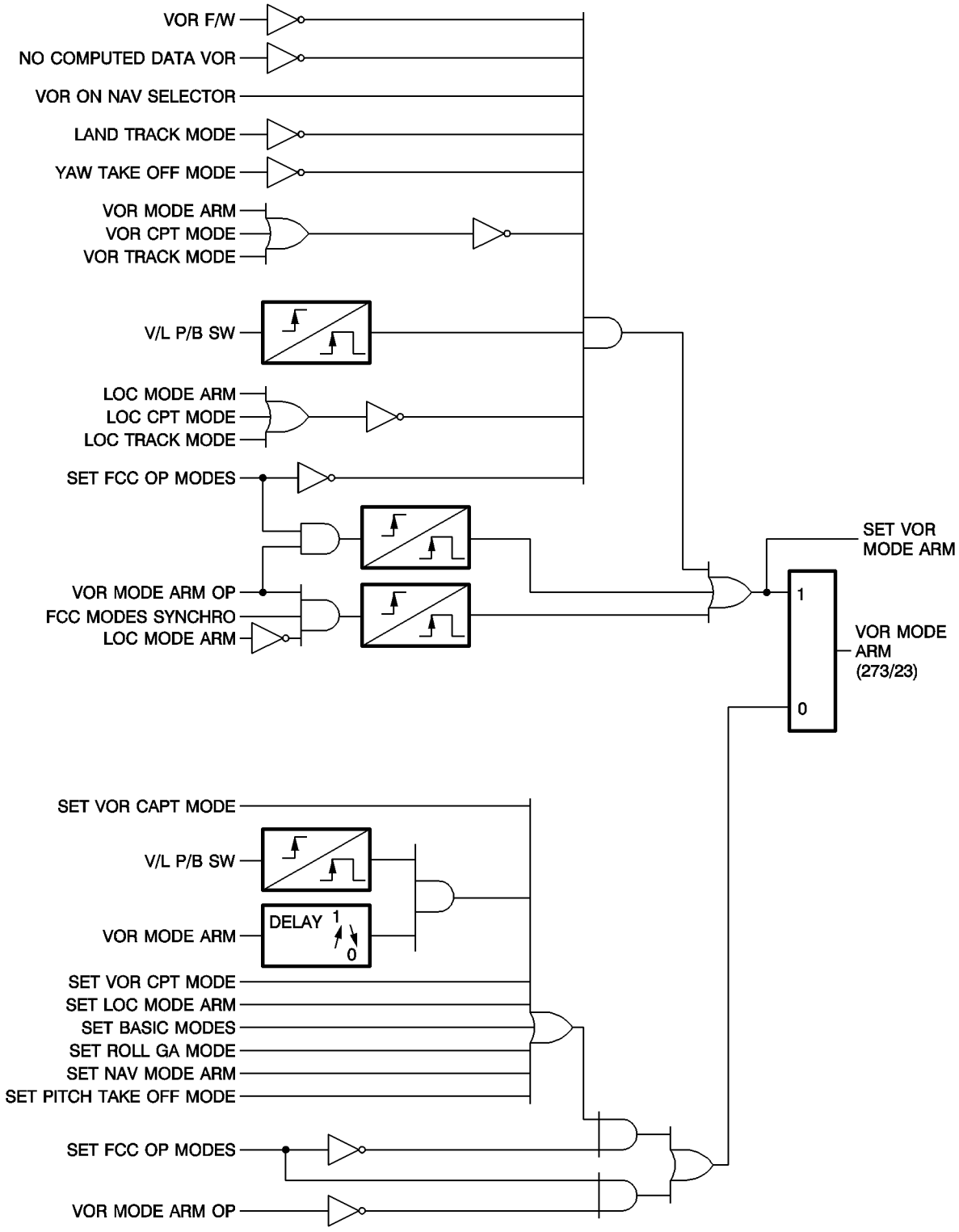
VOR Mode
Figure 009

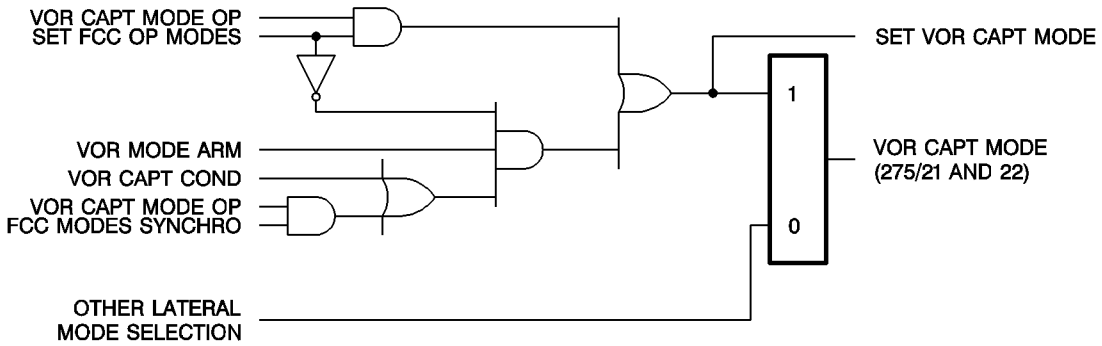


VOR Mode Computation
Figure 010



VOR Mode Logic Arming Conditions
Figure 011

R



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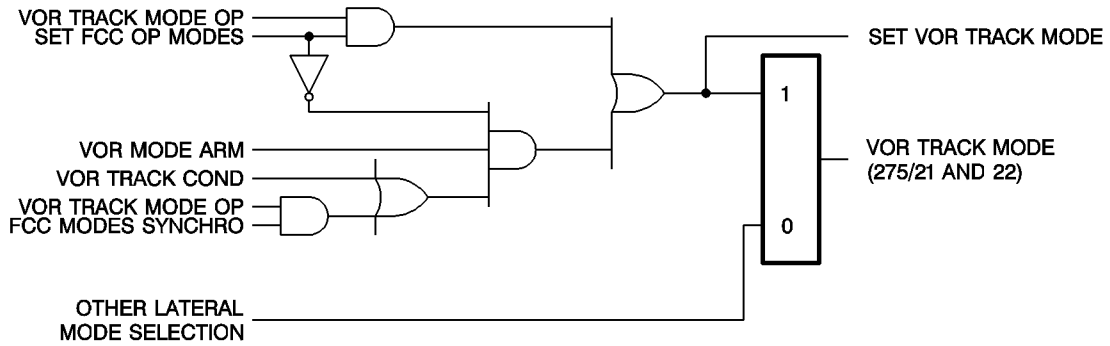
VOR Mode Logic Capture Conditions
Figure 012

R



BM5 22 17 20 0 CZM2 01

R VOR Mode Logic Track Conditions
Figure 013

4. LOC Mode

A. Characteristics

The LOC mode captures and tracks a LOCALIZER beam independently of the GLIDE beam. This mode is used :

- on the airfields not provided with glide transmitter
- in the event of very noisy ILS beams (Cat. 1 beams for example).

This mode includes :

- an arming phase
- a capture phase
- a track phase.

During the arming phase, the HDG, HDG/S and NAV modes can be used as support modes.

During the capture phase, the CWS is available (supervisory override function).

B. Operation (Ref. Fig. 014)

(1) Engagement/disengagement

The arming, capture and track conditions of the LOC are given in figures (Ref. Fig. 015, 016, 017).

Engagement of :

. both FDs
. both FDs and one AP in CMD
. both APs in CMD (with LAND selected)

and the fact of having one pushbutton switch common to VOR and LOC mode engagement have the following consequences :

(2) Simultaneous engagement of both FDs

- before arming, if a VOR/DME control unit displays an ILS frequency and the other displays a VOR frequency, priority is given to the LOC mode when V/L is selected. Consequently, the FD associated with the control unit displaying a VOL frequency, disengages (ref. paragraph C.(5) VOR mode).
- with LOC mode armed or active for both FDs and if a VOR frequency is displayed on the VOR/DME control unit, the associated FD disengages (a 5-second delay is provided to enable frequency change).

(3) Engagement of one AP in CMD and both FDs

Priority is given to AP engagement. Before the V/L mode is selected and if the VOR/DME control unit associated with the AP displays a LOC frequency and if the opposite VOR/DME control unit displays a VOR frequency, at V/L selection AP (and associated FD) engages in LOC mode and opposite FD disengages (it will engage again in LOC mode if a LOC frequency is selected on the VOR/DME control unit). After engagement of the LOC mode and if a VOR frequency is selected on the VOR/DME control unit(s), the associated system(s) disengages.

(4) Engagement of both APs in CMD

LAND mode being armed or active (and as long as altitude is greater than 400 ft.) selection of the LOC mode (with ILS frequencies displayed on the VOR/DME control units) is taken into account. Below 400 ft. this selection has no effect if the LAND mode is active.

The logic conditions associated with AP/FD engagement in V/L mode are given in 22-16, paragraph 2.C.(1).

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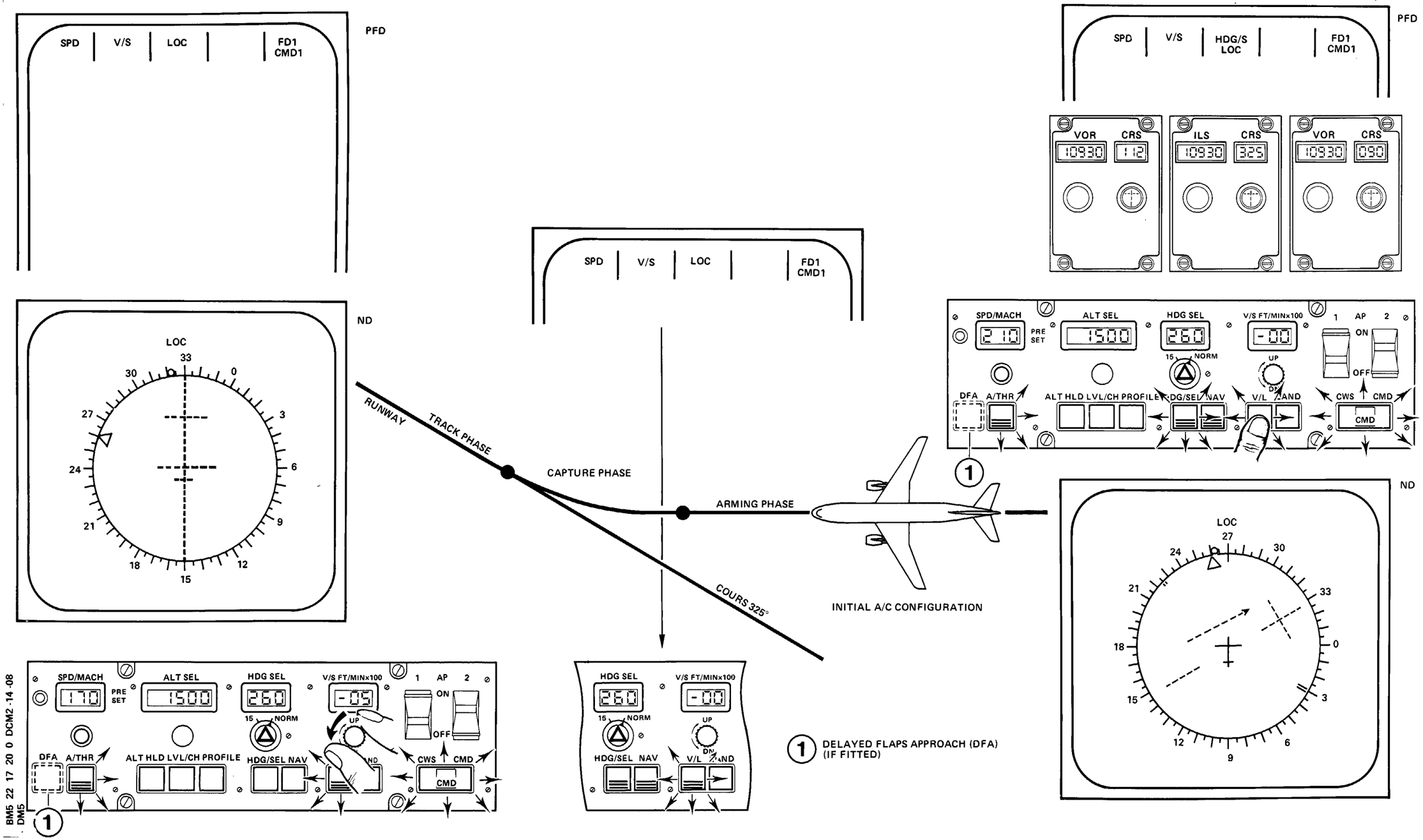

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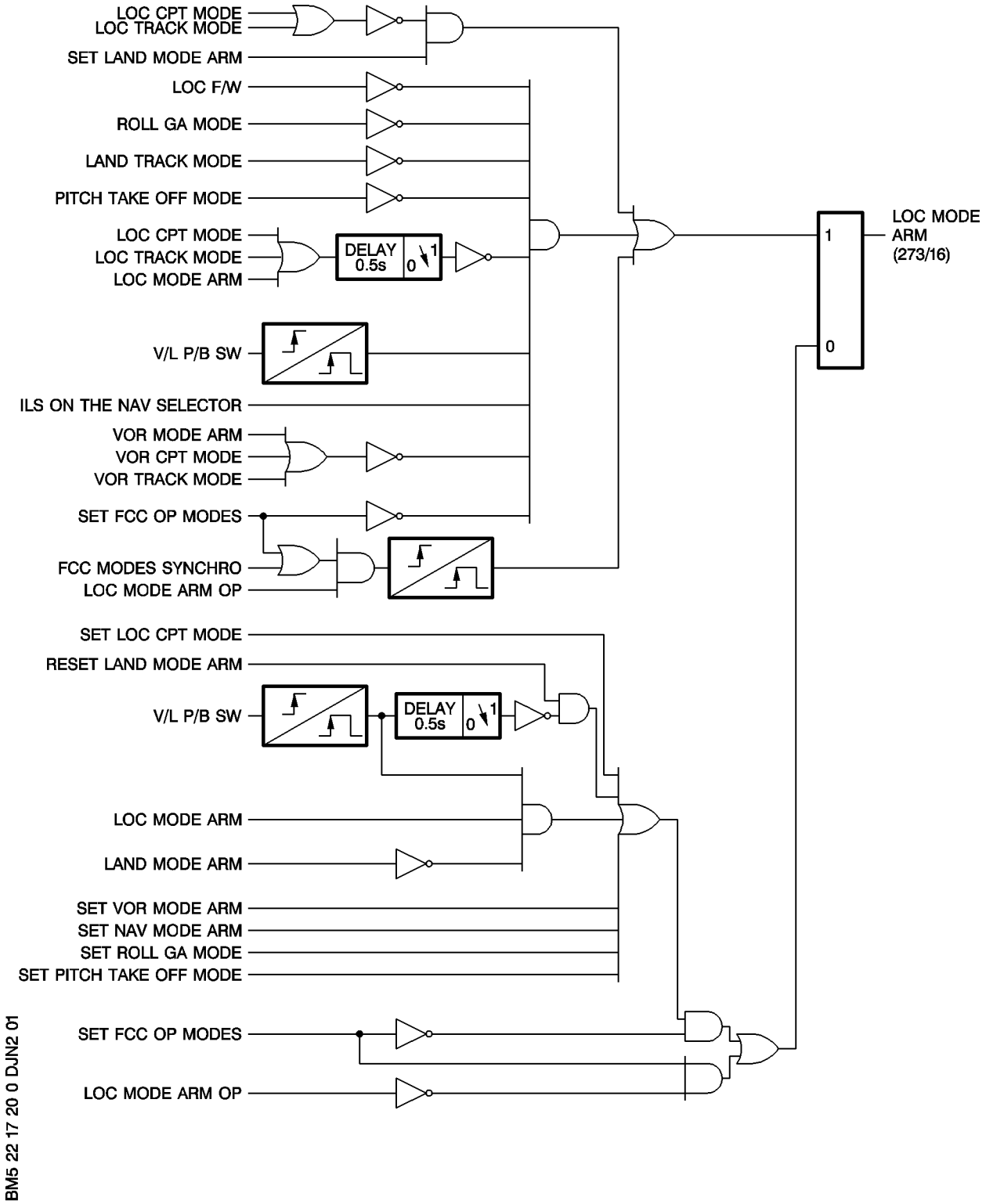
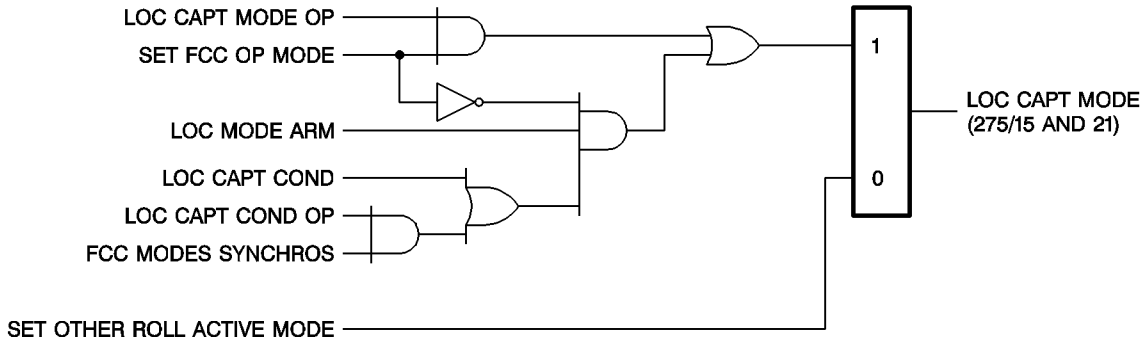
LOC Mode
Figure 014



LOC Mode Logic Arming Conditions
Figure 015

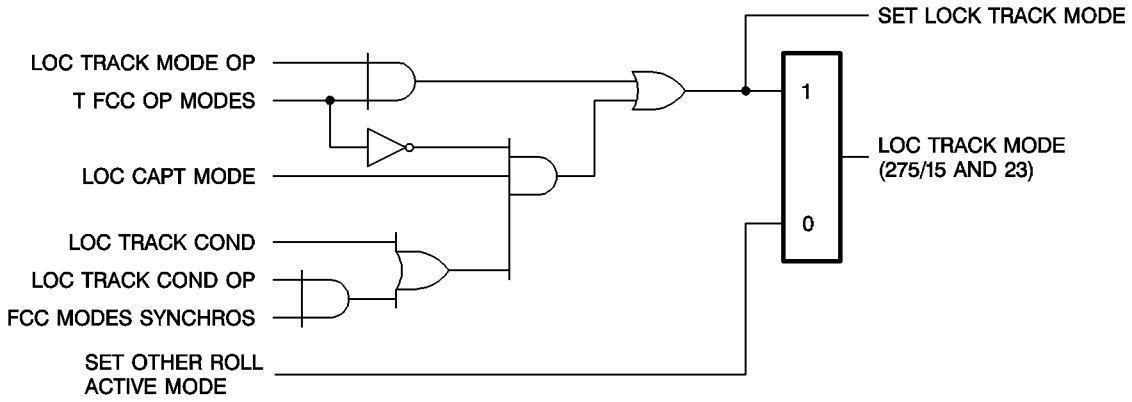
R



BM5 22 17 20 0 DLM2 01

LOC Mode Logic Capture Conditions
Figure 016

R



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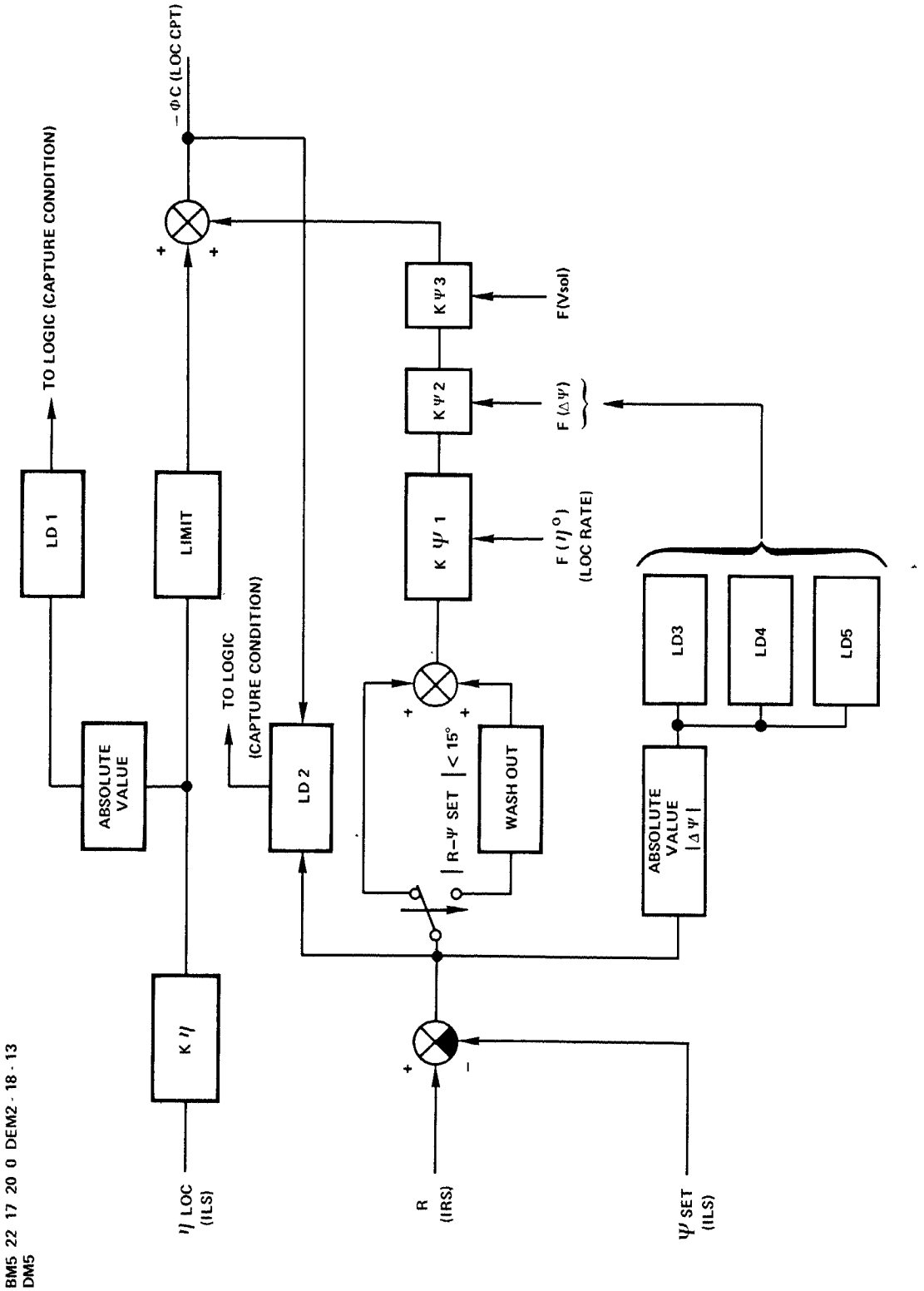
LOC Mode Track Logic Conditions
Figure 017

R

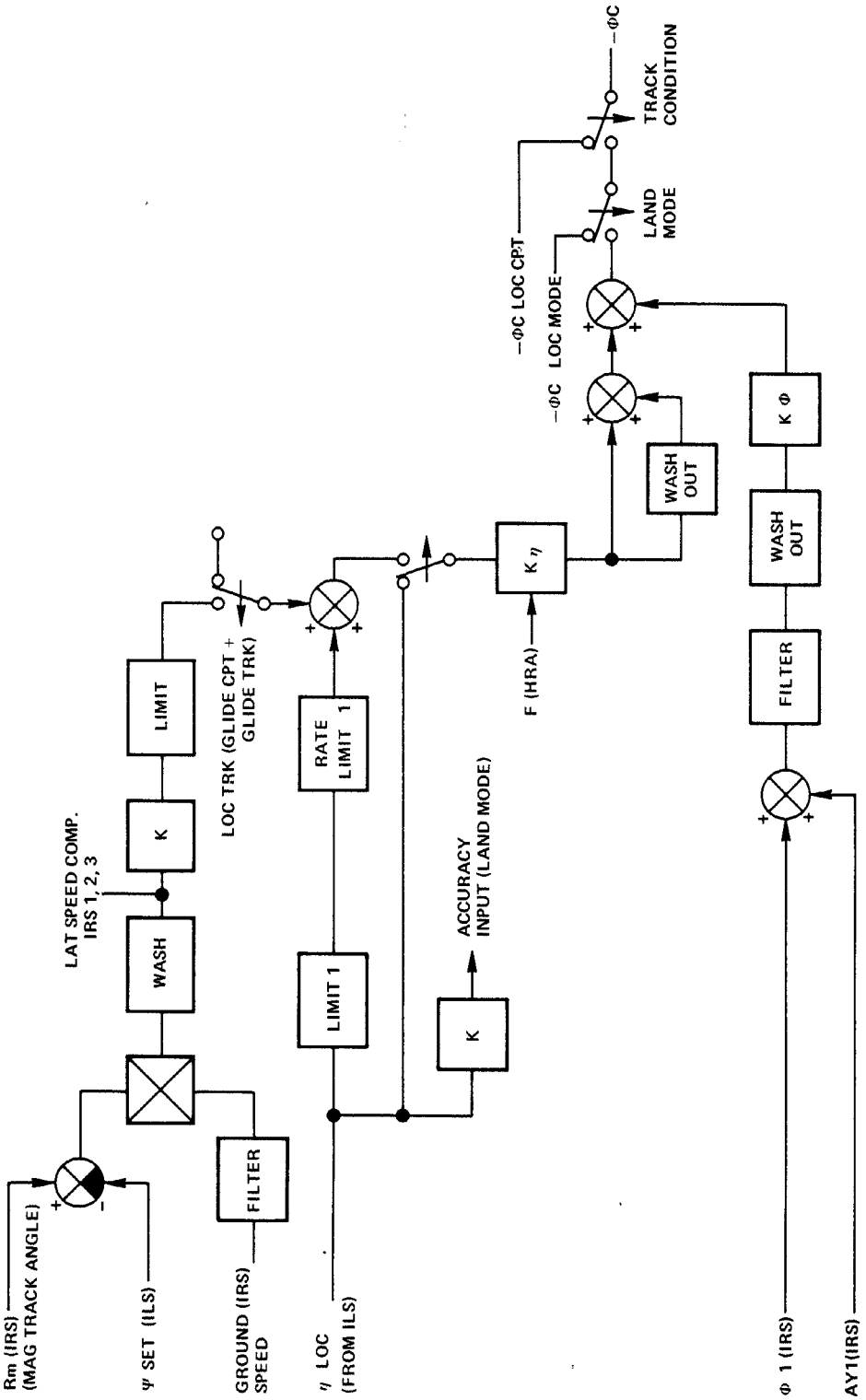
C. Command generation
The computation principle of the guidance law in LOC mode for the capture and track phases is given in following figures.
(Ref. Fig. 018).
(Ref. Fig. 019).

D. Display
- Selection of the LOC mode is indicated by illumination of the V/L mode selection pushbutton switch.
- Arming of the LOC mode is indicated on the PFD by illumination of the cyan LOC indication.
- Engagement of the capture phase is indicated on the PFD by illumination of the green LOC * indication.
- Change to the track phase is indicated on the PFD by illumination of the green LOC indication.

It is to be noted that during the arming phase the cyan LOC indication is accompanied by a green indication corresponding to the support mode engagement.



LOC TRACK Computation
Figure 018



LOC TRACK Computation
Figure 019

R

R

The AP/FD modes common to the longitudinal and lateral axes are :
TAKEOFF (TO)
LAND
GO AROUND (GA)
Modes engage on both axes. However it is possible to leave TO and GA modes one axis after the other.
Selection of the LAND or GA mode authorizes engagement of both APs in CMD.

1. Take off Mode (TO)

A. Characteristics
This mode provides a guidance function, at takeoff, on the runway axis by means of the LOC beam and by following an optimum longitudinal flight path after rotation.
This mode is available for the FD (on the ground the AP can be engaged in CWS mode only).
After takeoff, 4 seconds after the extension of the main landing gear shock absorber, the AP can be engaged in CMD and provides guidance of the aircraft with the same laws as those used for the FD.

B. Operation (Ref. Fig. 001)
(1)Engagement/disengagement.
When TO mode is selected, the FD commands are indicated by the pitch and yaw bars. The roll bar does not appear as long as the yaw bar is visible.
In addition, selection of TO results in the coupling of the autothrottle (the A/THR function is activated and the THR mode engages).
On the ground, it is possible to arm the PROFILE mode on the pitch axis and the NAV or HDG SEL mode on the roll axis.
After takeoff the PROFILE mode will be taken into account at altitude 2 (ref. 22-17-10, paragraph 4.).
The armed lateral mode will be taken into account when the FD bar disappears.
The engage conditions of the TO mode are given in the following figures.
(Ref. Fig. 001)
(Ref. Fig. 002)
(Ref. Fig. 003)

C. Command generation
(1)Pitch axis
The guidance law enables to hold V2 + 10 kt with both engines running (V2 is displayed on the FCU before takeoff).
With one engine running the guidance law enables to hold :
- the aircraft speed if it is greater than V2 at the time of the engine failure
- V2 if the aircraft speed is lower than V2 at the time of the engine failure.

In addition, the guidance law includes :
- an attitude protection to reduce aircraft nose-up attitude during

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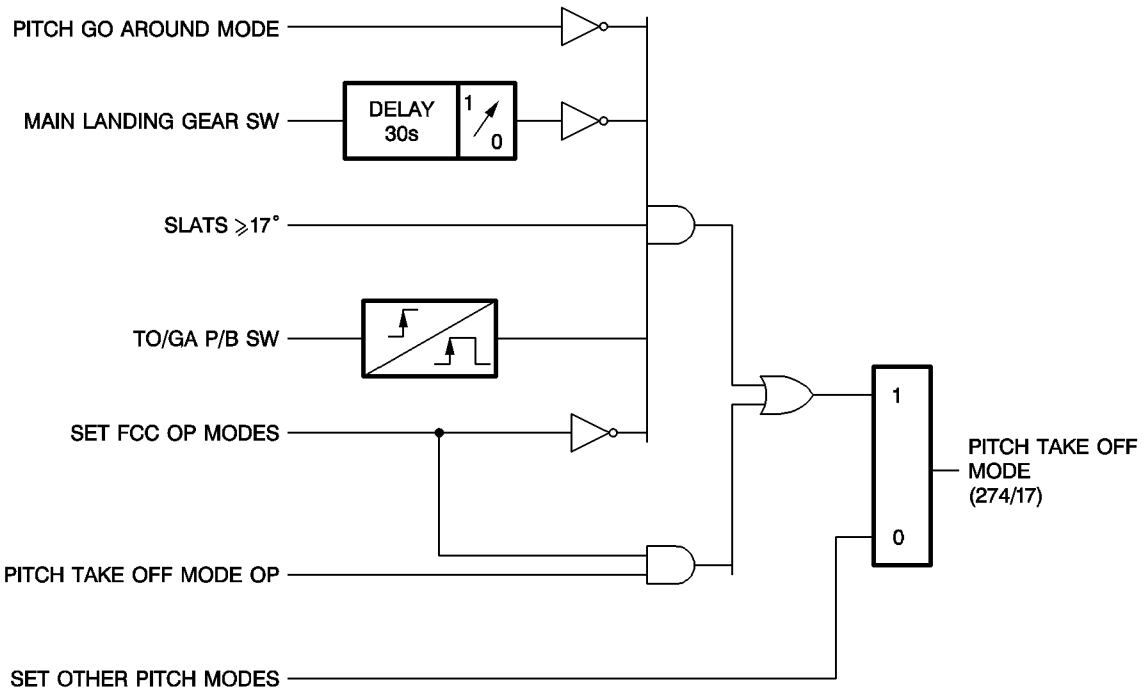
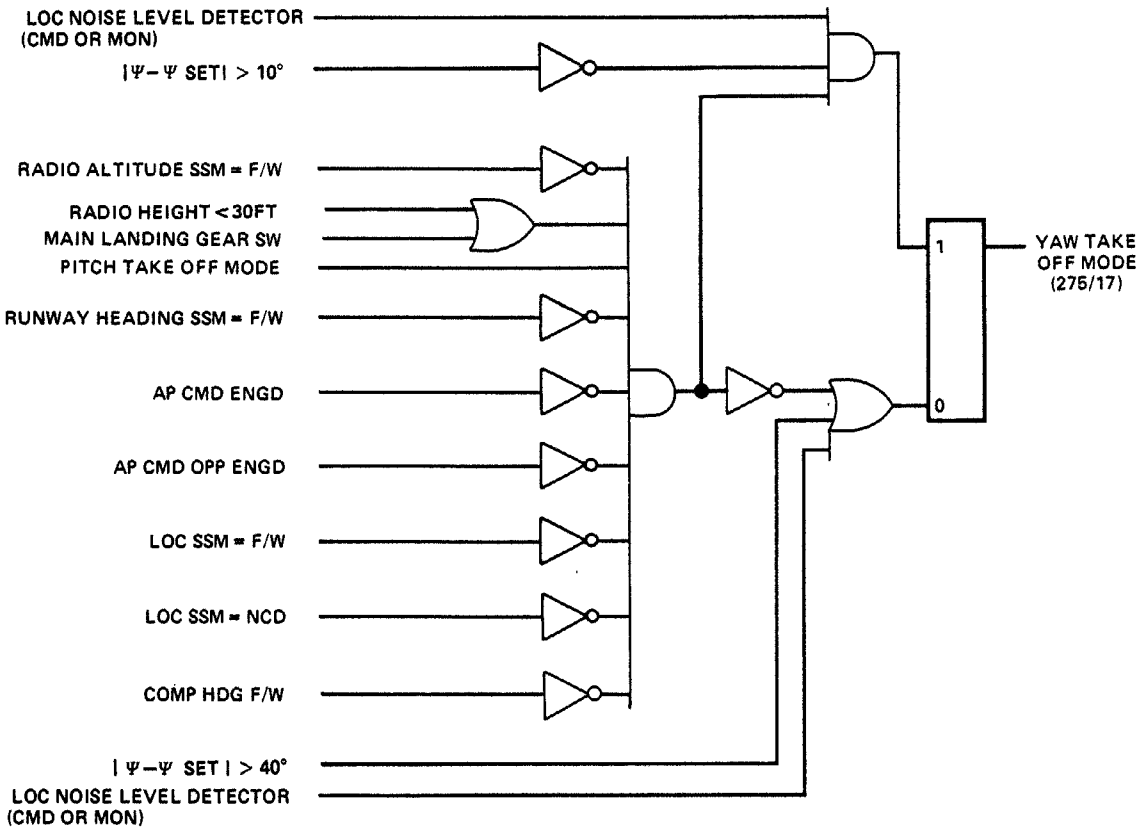
Takeoff Mode Operation
Figure 001



BM5 22 17 30 0 HAM2 01

Takeoff Mode Logic - Pitch Axis
Figure 002

R



BM5 22 17 30 0 DZT2 24

Takeoff Mode Logic - Lateral Axis
Figure 003

this phase
- a vertical speed protection to ensure a minimum climbing rate.
  NOTE : An attitude pre-command is provided to initialize aircraft rotation when this law is used in GA mode.
- a protection against windshear (detected by the FAC's) for maintaining a minimum climbing rate to the detriment of speed, and then to the detriment of altitude when the safety speed is reached (approx. 1.1 VS). The safety speed can be greater than 1.1 VS to maintain a sufficient margin with respect to stall when leaving the disturbance. It is to be noted that the SRS guidance law remains unchanged as long as the aircraft performances are not affected by windshear.

The computation principle of the guidance law in SRS mode is given in the following figure.
(Ref. Fig. 004)

(2)Lateral axis
The computation principle of the takeoff run law is identical with the roll out law.
(Ref. Fig. 005)

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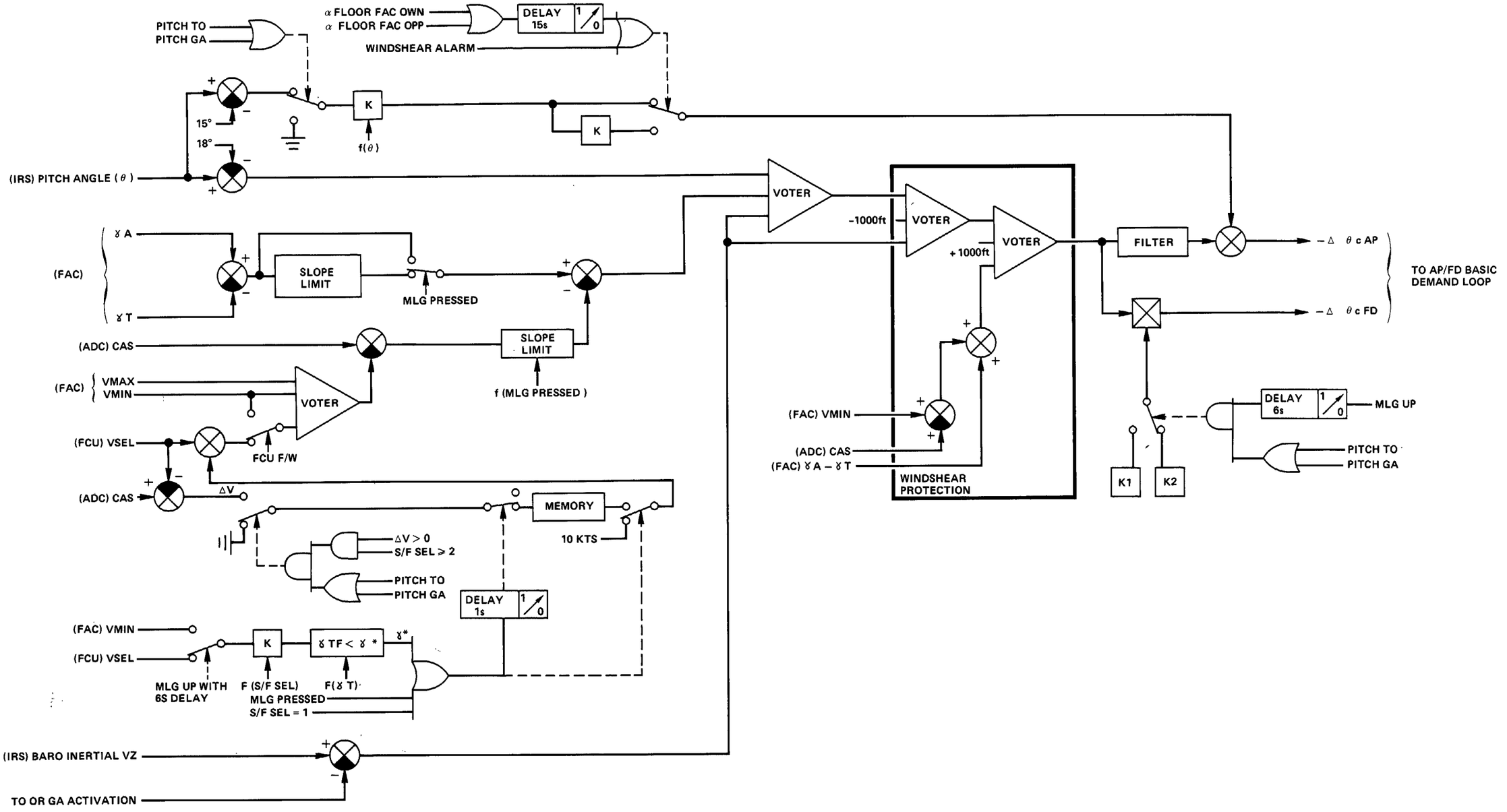

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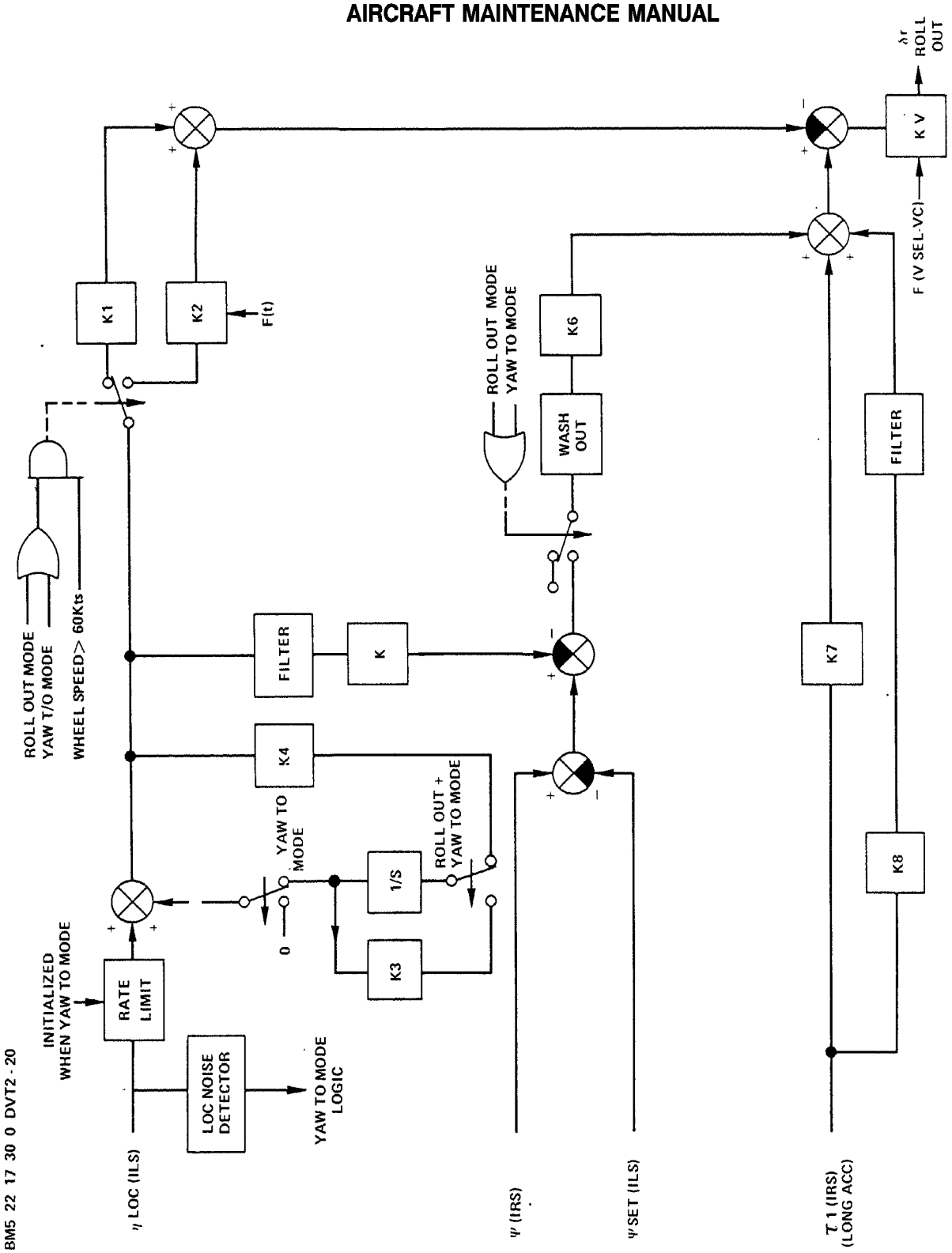
Takeoff or Go Around Simplified Control Law
Figure 004



Roll Out/Takeoff Mode Computation
Figure 005

D. Display

Engagement of the TO mode is shown by the following green indications :

. SRS : on the PFD in the section corresponding to the AP/FD longitudinal modes.

. RWY : on the PFD in the section corresponding to the AP/FD lateral modes.

When the autothrottle engagement in THR mode is forced by the TO engagement, it is shown by the green THR indication in the PFD section corresponding to the ATS modes and by illumination of the A/THR pushbutton switch on the FCU (Ref. Fig. 001).

After 30 ft., the green RWY indication is replaced by HDG if no lateral mode has been armed on the ground or by the green indication corresponding to the lateral mode armed on the ground (e.g. HDG/S or NAV).

2. LAND Mode

A. Characteristics
The LAND mode provides the capture and track of the ILS beam (LOC and GLIDE) as well as the flare, alignment and roll out functions to enable automatic landing and guidance on the ground after touchdown.
This mode is available for the AP and FD (although the flare phase is not certified for the FD). As this mode is provided also for category 3 landings, selection of second AP is authorized.

B. Operation (Ref. Fig. 006)
(1)Engagement/disengagement
As soon as the LAND mode is selected, engagement of both APs is authorized.
Since the LAND mode uses an arming phase, the lateral and longitudinal support modes remain engaged until the capture of the LOC and GLIDE beams respectively and control the aircraft along a path convergent with the LOC and GLIDE

In addition the pilot must select the ILS transmitter frequency and the runway heading on the ILS control box.
Below 400 ft. the LAND mode is latched so that it is not possible to leave the mode by engaging a mode other than the GO AROUND mode.
In addition, for safety reasons, actions on the FCU (e.g. CMD-CWS switching) are no longer taken into account after an altitude of 400 ft. has been reached with both APs engaged.
(2)LAND mode sub-phases
The engage logic conditions of the LAND mode sub-phases are shown in the following figures.
(Ref. Fig. 007)
(Ref. Fig. 008)
(Ref. Fig. 009)
(Ref. Fig. 010)
(3)Controls associated with LAND mode selection
Selection of the LAND mode entails the following controls on systems external to the AFS :
- busbar splitting
- radio altimeter test inhibition
- ILS test inhibition.
The logic conditions associated with AP/FD engagement in LAND mode are given in 22-16 paragraph 2.C.(1).
(Ref. Fig. 011)

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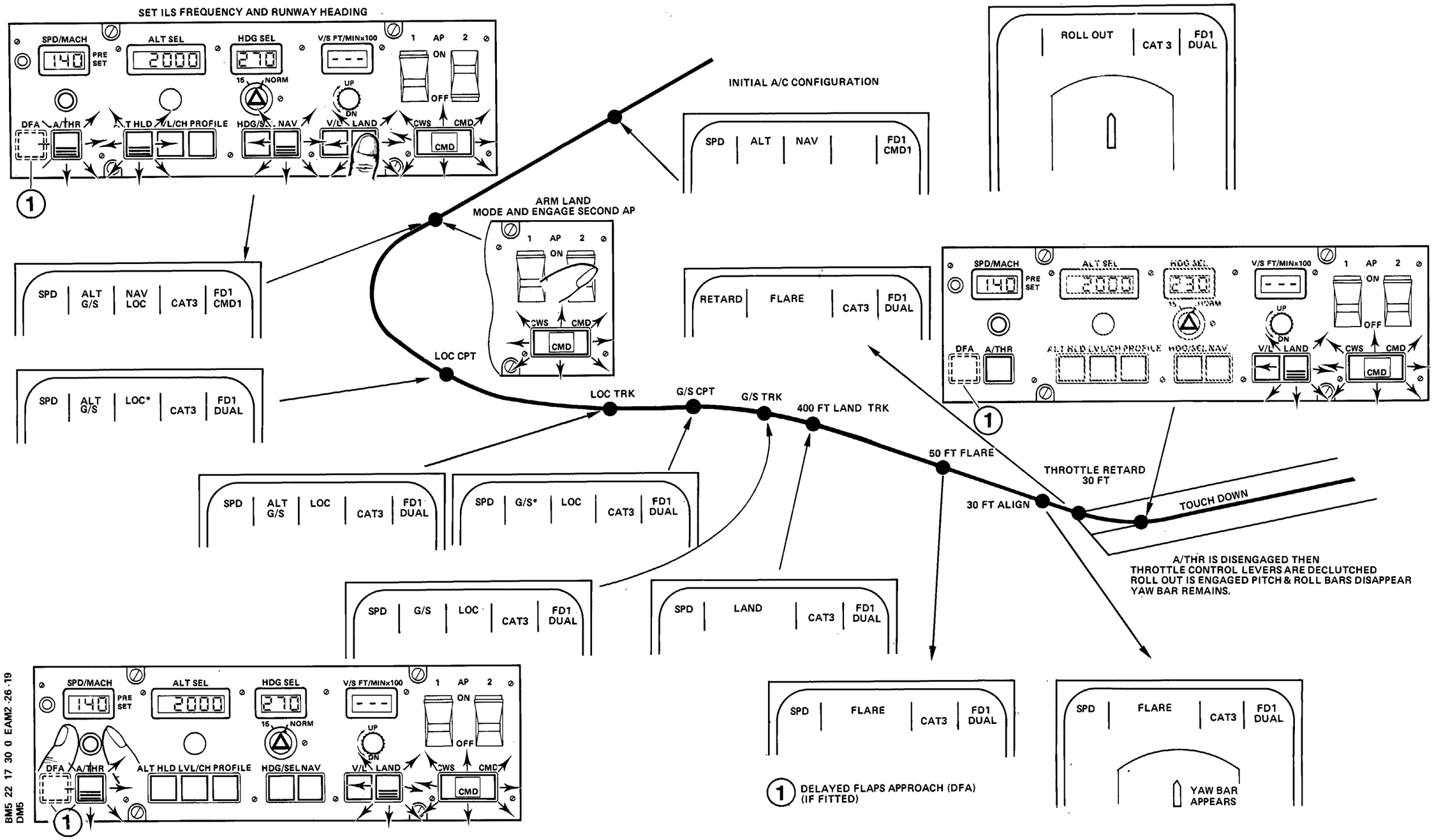
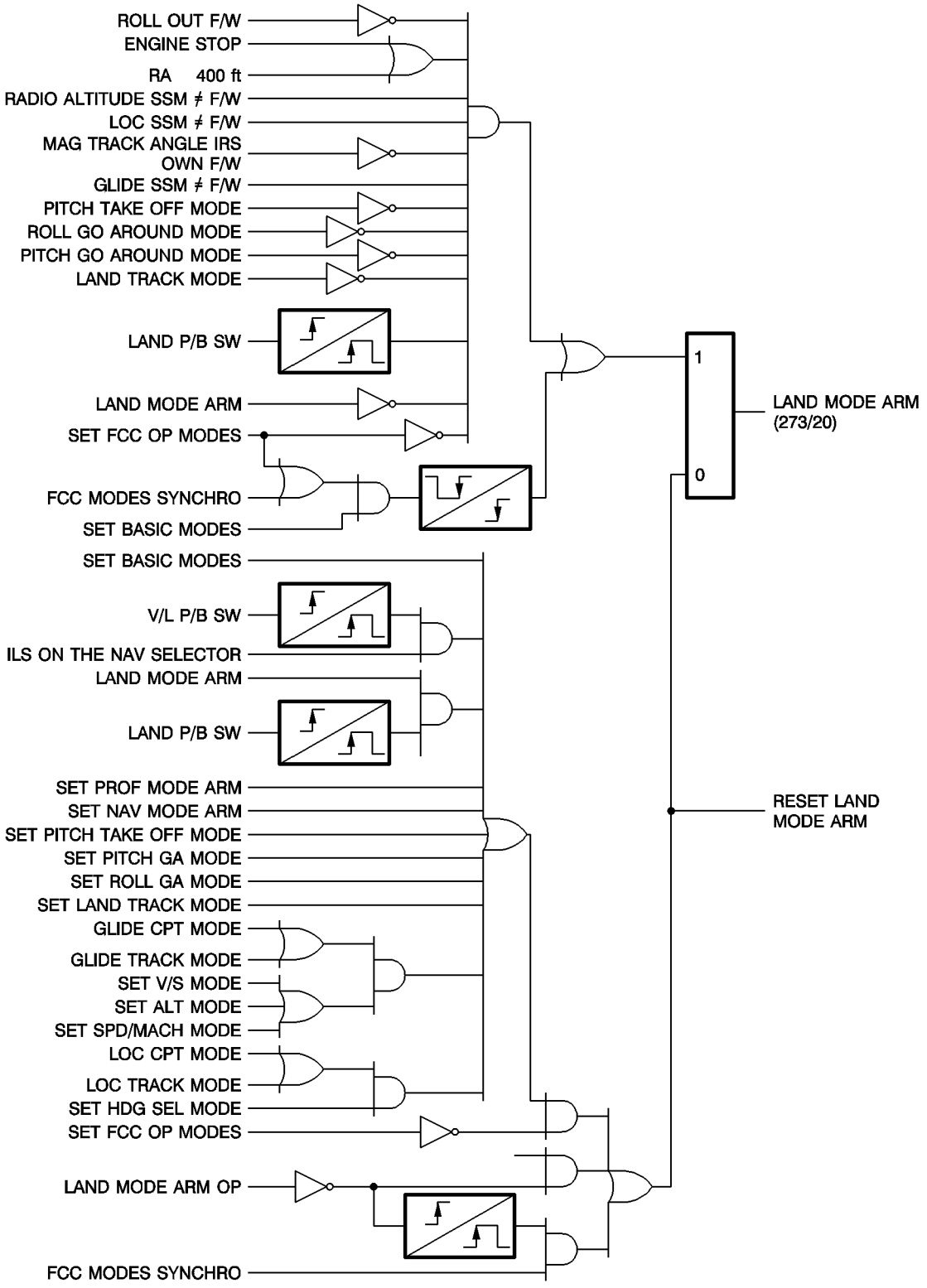

SET ILS FREQUENCY AND RUNWAY HEADING
SPD/MACH
ALT SEL
HDG SEL
V/S FT/MINx100
1 AP 2
ON
OFF
PRE SET
NORM
UP
DN
DFA
A/THR
ALT HLD
LVL/CH PROFILE
HDG/SEL
NAV
V/L
LAND
CWS
CMD
INITIAL A/C CONFIGURATION
SPD | ALT | NAV | FD1 CMD1
ROLL OUT | CAT 3 | FD1 DUAL
ARM LAND MODE AND ENGAGE SECOND AP
SPD | ALT G/S | NAV LOC | CAT3 | FD1 CMD1
SPD | ALT G/S | LOC* | CAT3 | FD1 DUAL
SPD | ALT G/S | LOC | CAT3 | FD1 DUAL
SPD | G/S* | LOC | CAT3 | FD1 DUAL
SPD | G/S | LOC | CAT3 | FD1 DUAL
SPD | LAND | CAT3 | FD1 DUAL
RETARD | FLARE | CAT3 | FD1 DUAL
SPD | FLARE | CAT3 | FD1 DUAL
SPD | FLARE | CAT3 | FD1 DUAL
YAW BAR APPEARS
LOC CPT
LOC TRK
G/S CPT
G/S TRK
400 FT LAND TRK
50 FT FLARE
THROTTLE RETARD 30 FT
30 FT ALIGN
TOUCH DOWN
A/THR IS DISENGAGED THEN THROTTLE CONTROL LEVERS ARE DECLUTCHED ROLL OUT IS ENGAGED PITCH & ROLL BARS DISAPPEAR YAW BAR REMAINS.
1 DELAYED FLAPS APPROACH (DFA) (IF FITTED)
BM5 22 17 30 0 EAM2 26 19 DM5

LAND Mode
Figure 006



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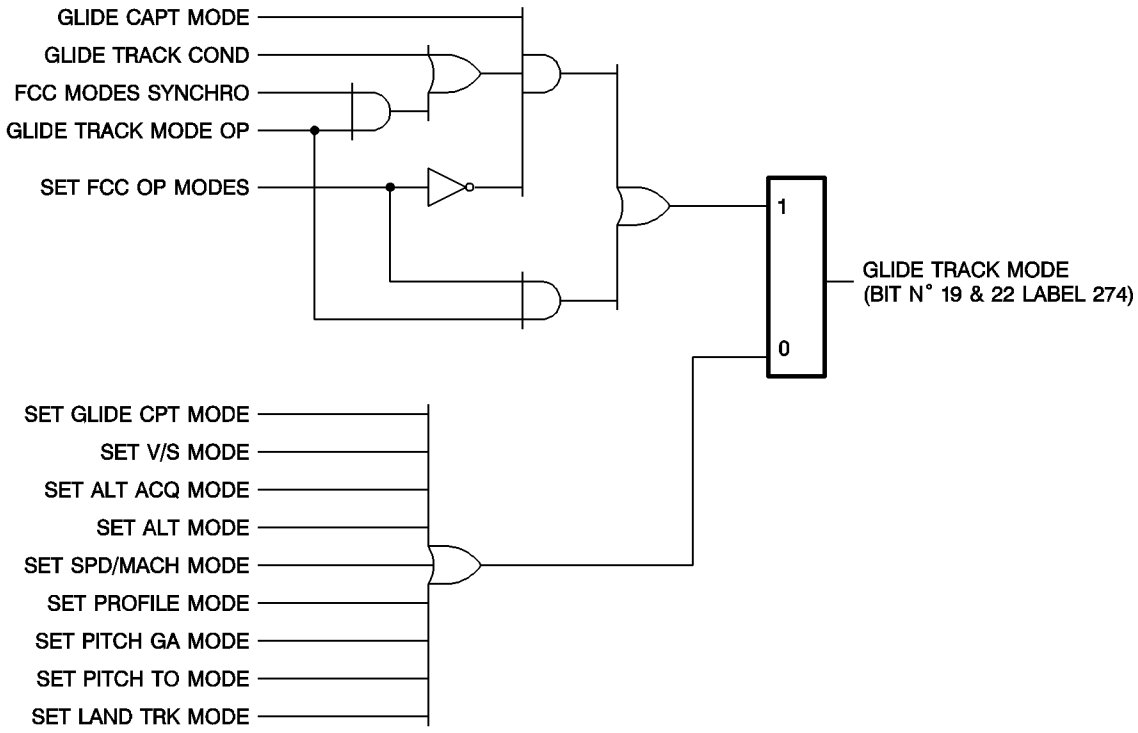
LAND Mode - Arming Logic Conditions
Figure 007

R



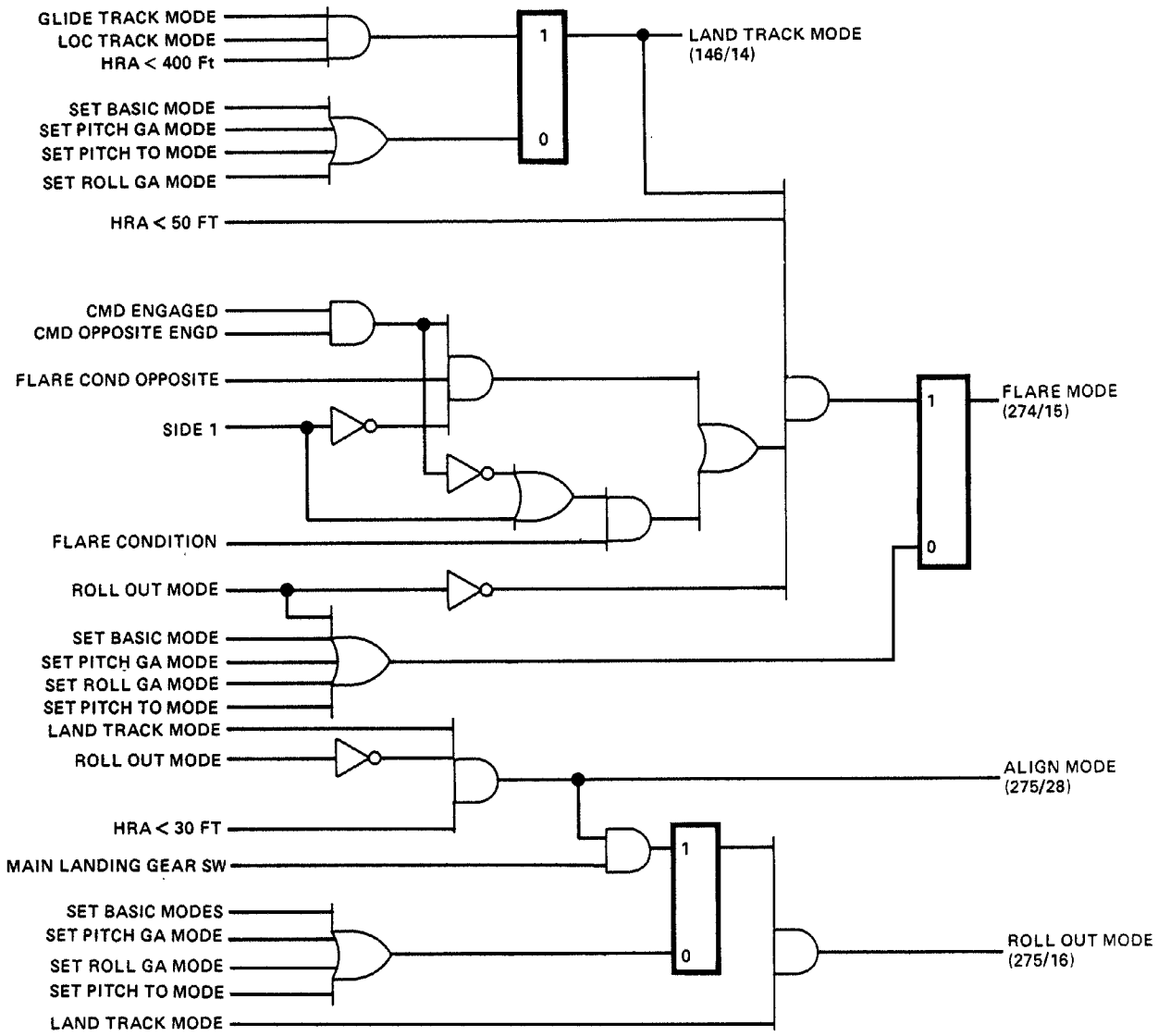
BM5 22 17 30 0 ETM2 01

LAND Mode - Glide Arm, Capture and Track Logic Conditions
Figure 008

R

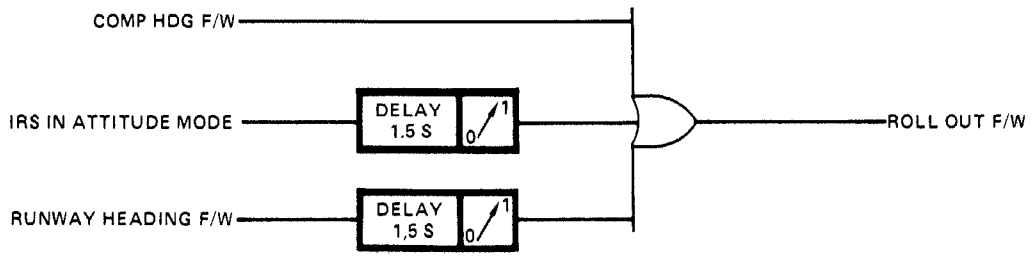
EFFECTIVITY: ALL

PIA



BM5 22 17 30 0 EXR2 - 00

LAND Mode - LAND Track, Flare, Align, Roll Out Logic Conditions
Figure 009



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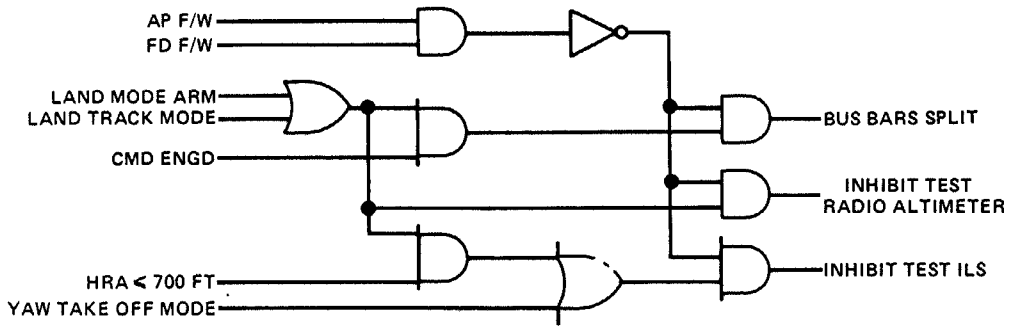
LAND Mode - Roll Out F/W
Figure 010



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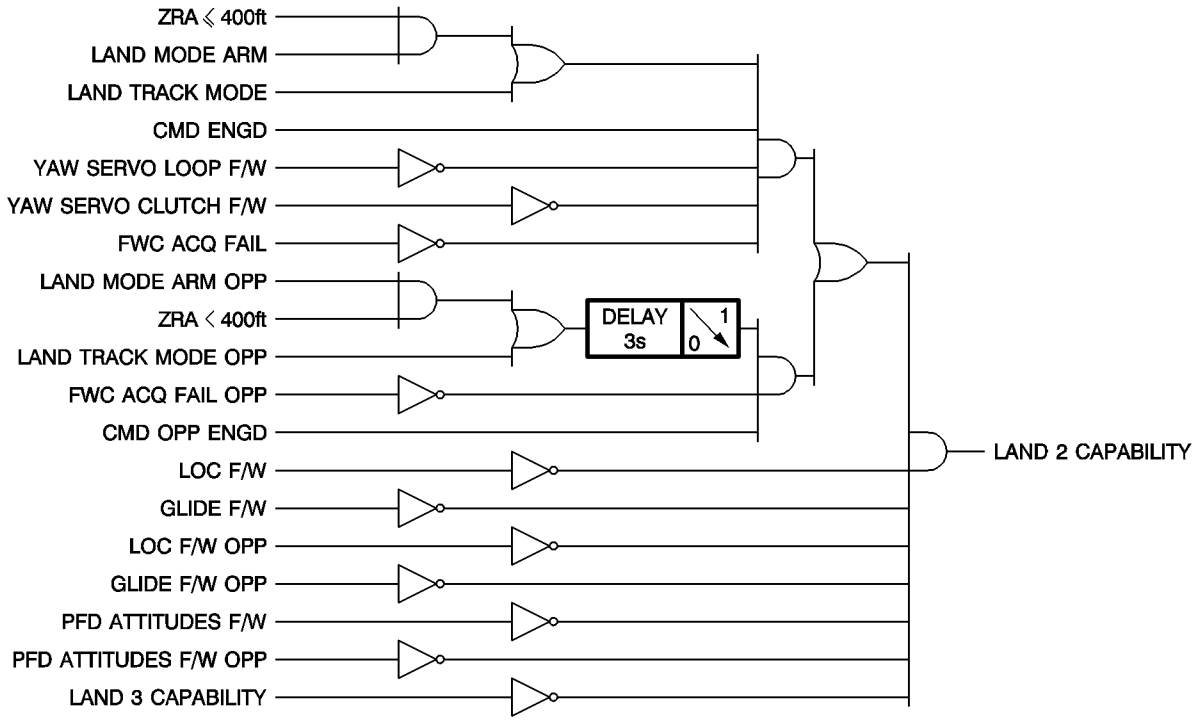
Orders Associated with LAND Mode Selection
Figure 011

(4)Landing capabilities
As soon as the LAND mode is selected, the FCCs transmit the available landing capability signal to the PFDs. The logic conditions associated with the capability display are given in following figures.
The loss of the LAND3 capability is displayed on the ECAM display units.

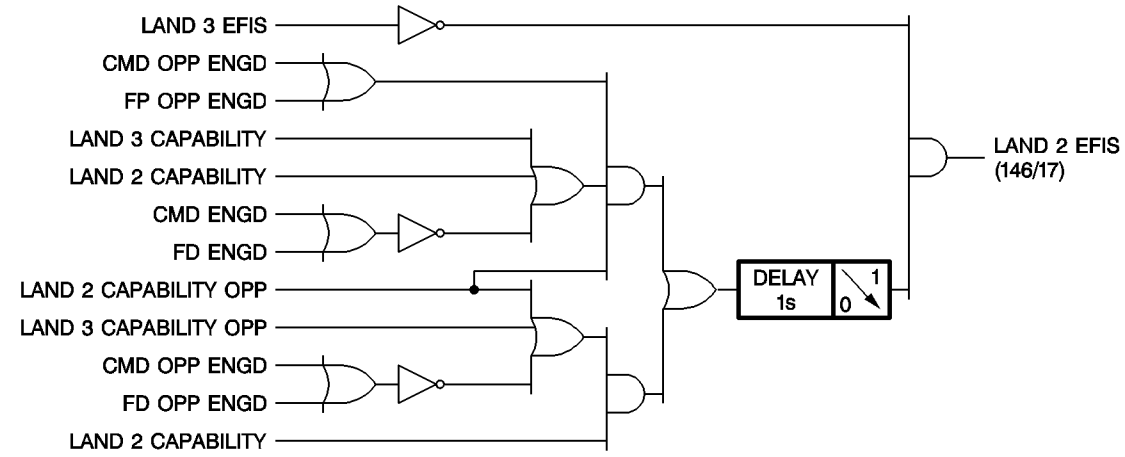
R NOTE : When the LAND pushbutton switch is pressed, the landing
R capability is displayed on the PFDs and the LAND 3 INOP warning
R goes off.

The generation principle of the LAND3 F/W condition is shown in following figure.
(Ref. Fig. 012)
(Ref. Fig. 013)
(Ref. Fig. 014)
(Ref. Fig. 015)

(5)Not applicable.



CAT 2 LOGIC CONDITIONS IN EACH FCC



CAT 2 INDICATION SENT TO PFD

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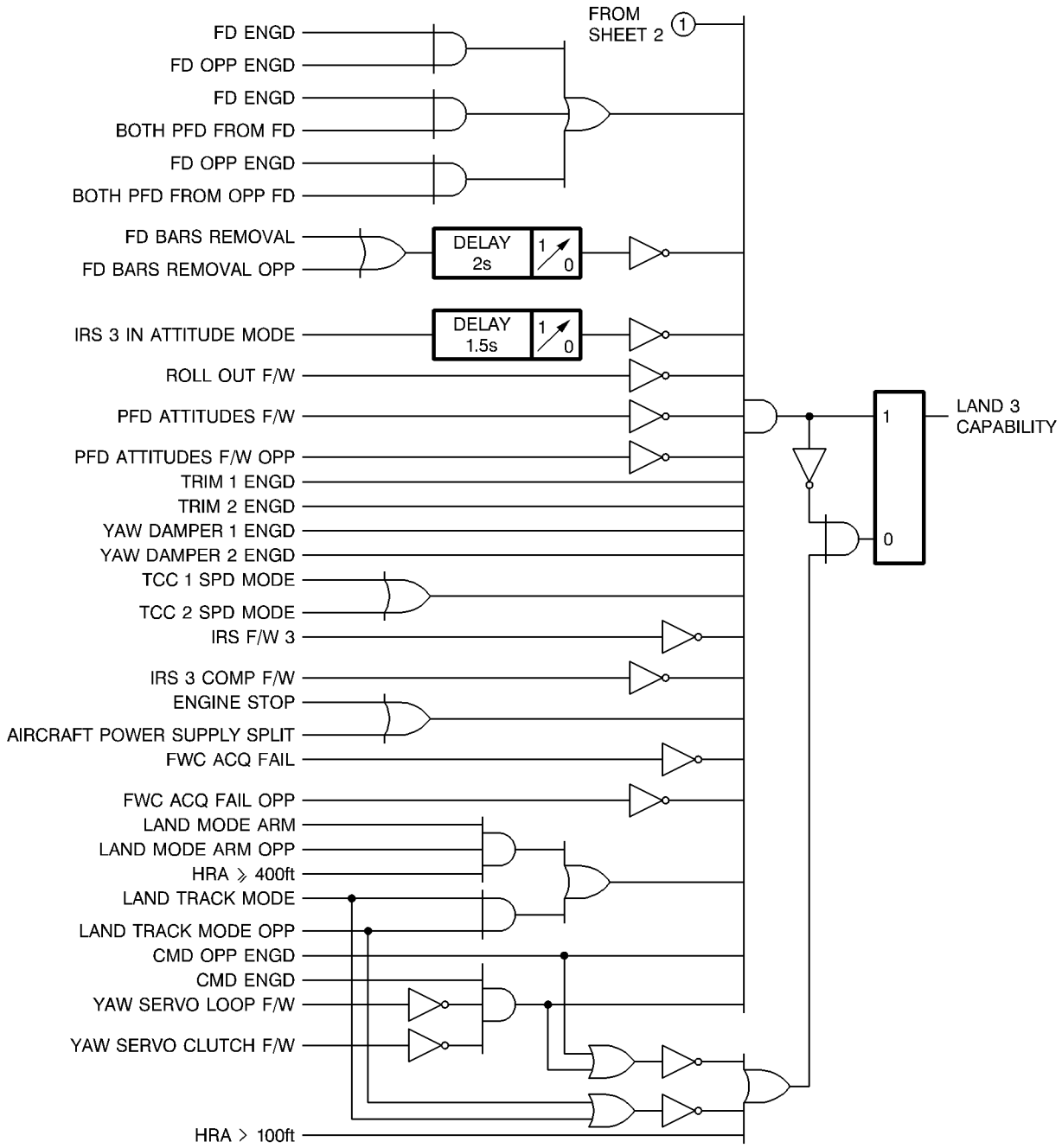
CAT2 Capability Logic Conditions
Figure 012

R



CAT 3 LOGIC CONDITION IN EACH FCC

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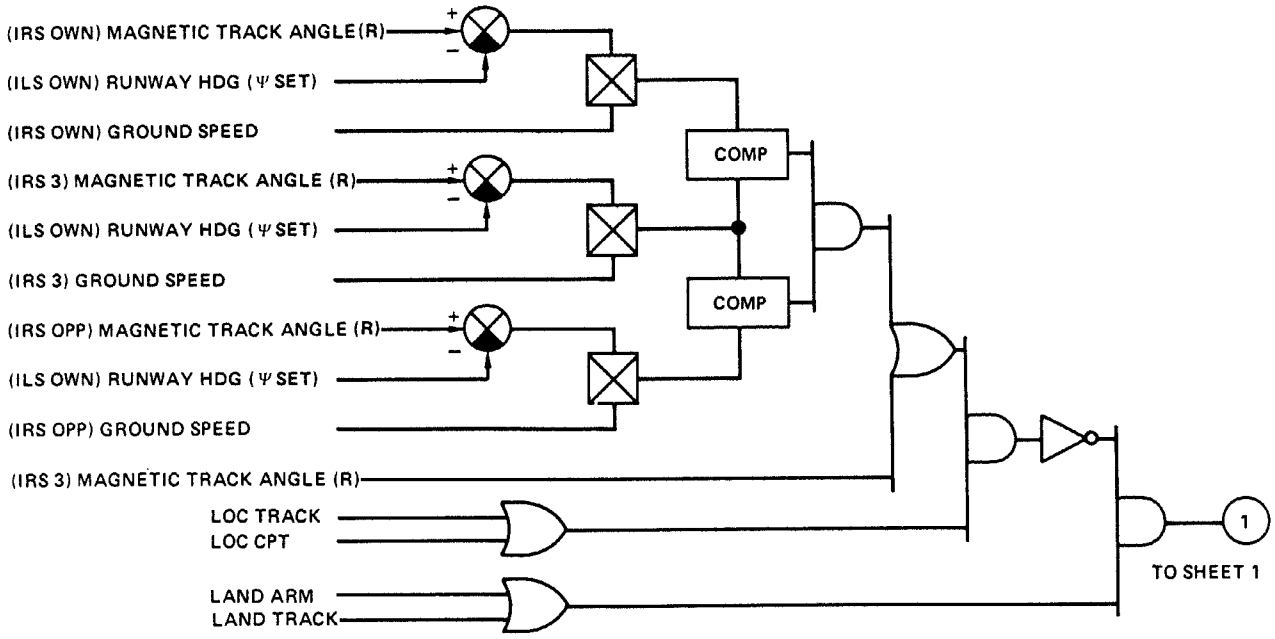
CAT3 Capability Logic Conditions
(In Each FCC) (Sheet 1/2)
Figure 013

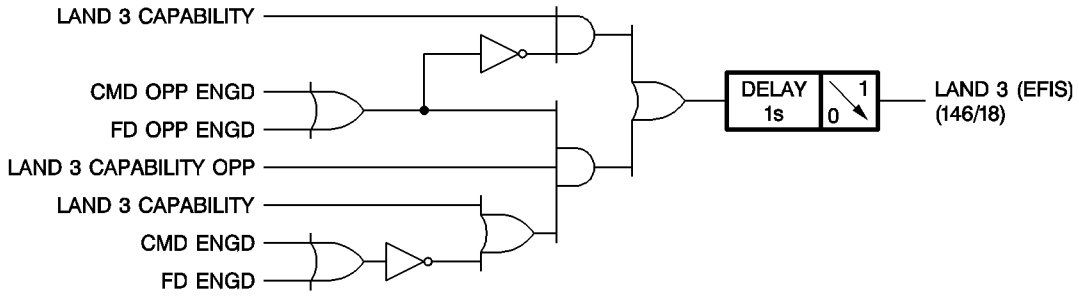
R



BM5 22 17 30 0 FJPM 00

CAT3 Capability Logic Conditions
(In Each FCC) (Sheet 2/2)
Figure 013

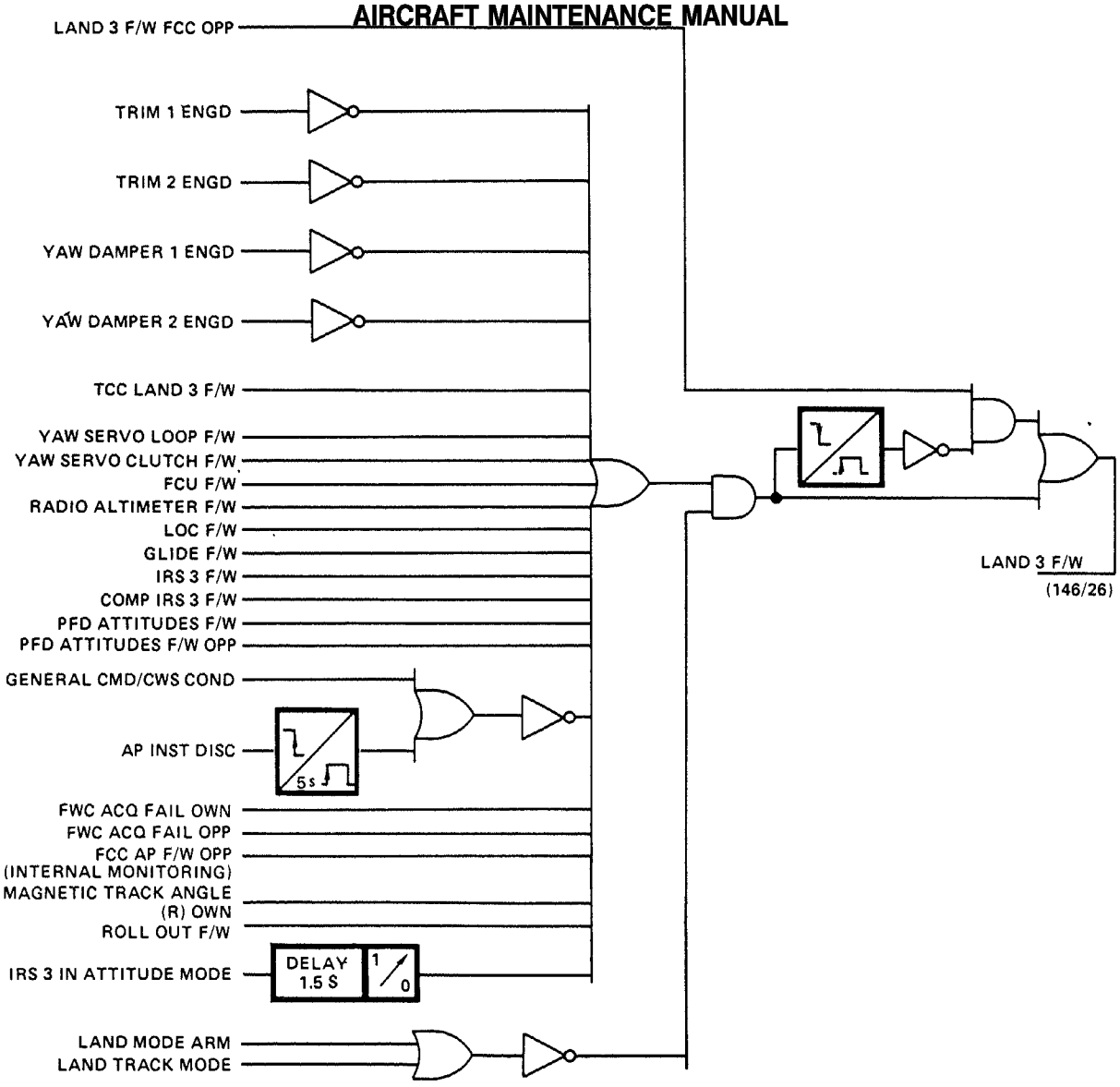
R



CAT 3 INDICATION SENT TO PFD

BM5 22 17 30 0 FJT2 01

CAT3 Indication Sent to PFD
Figure 014



BM5 22 17 30 0 FNR2-18-10
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LAND3 F/W Logic Conditions
Figure 015

(6)Warnings associated with automatic landing

An **AUTOLAND** warning light located on the glareshield in front of each pilot indicates to the pilots that a normal take over is required in the event of any abnormal automatic approach.

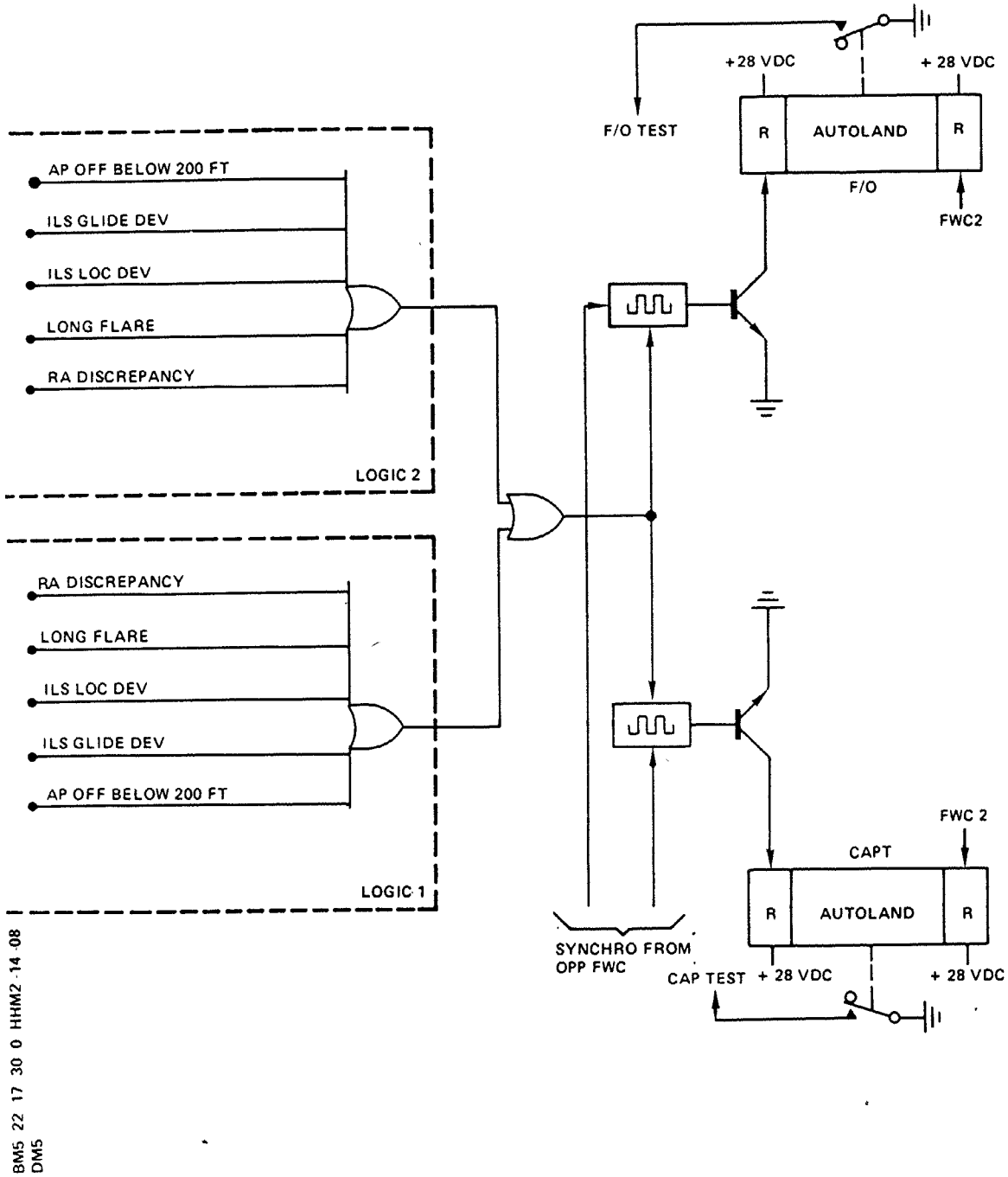
The **AUTOLAND** warning lights are red flashing lights which can be tested by pressing one of the lights.

The **AUTOLAND** warning is activated if :

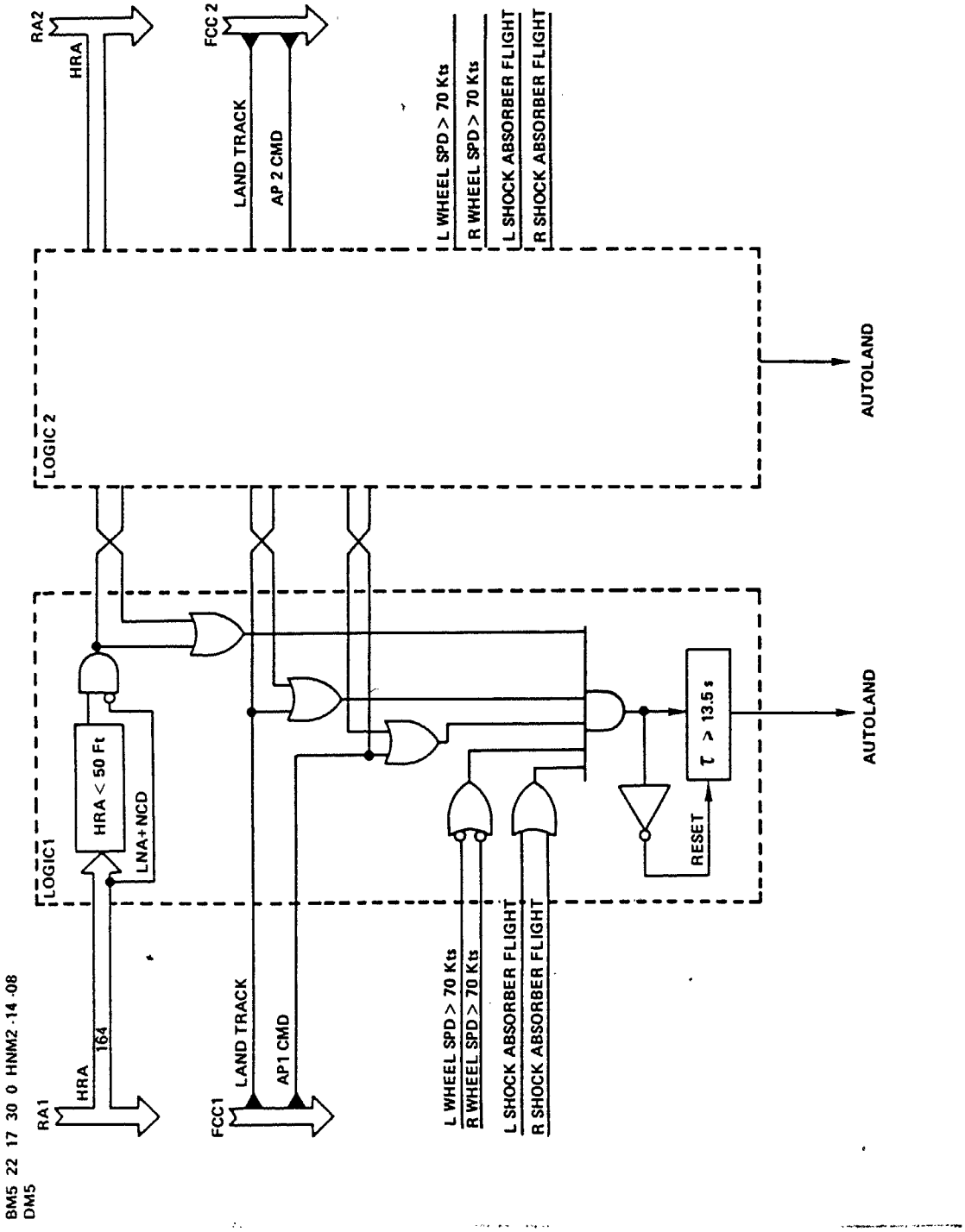
- **AP OFF** warning is activated
- **RA** discrepancy
- **LONG FLARE** warning is activated
- **LOC** or **GLIDE** excessive deviation warning is activated

The conditions for activating the **AUTOLAND** warning are shown in the following figures.

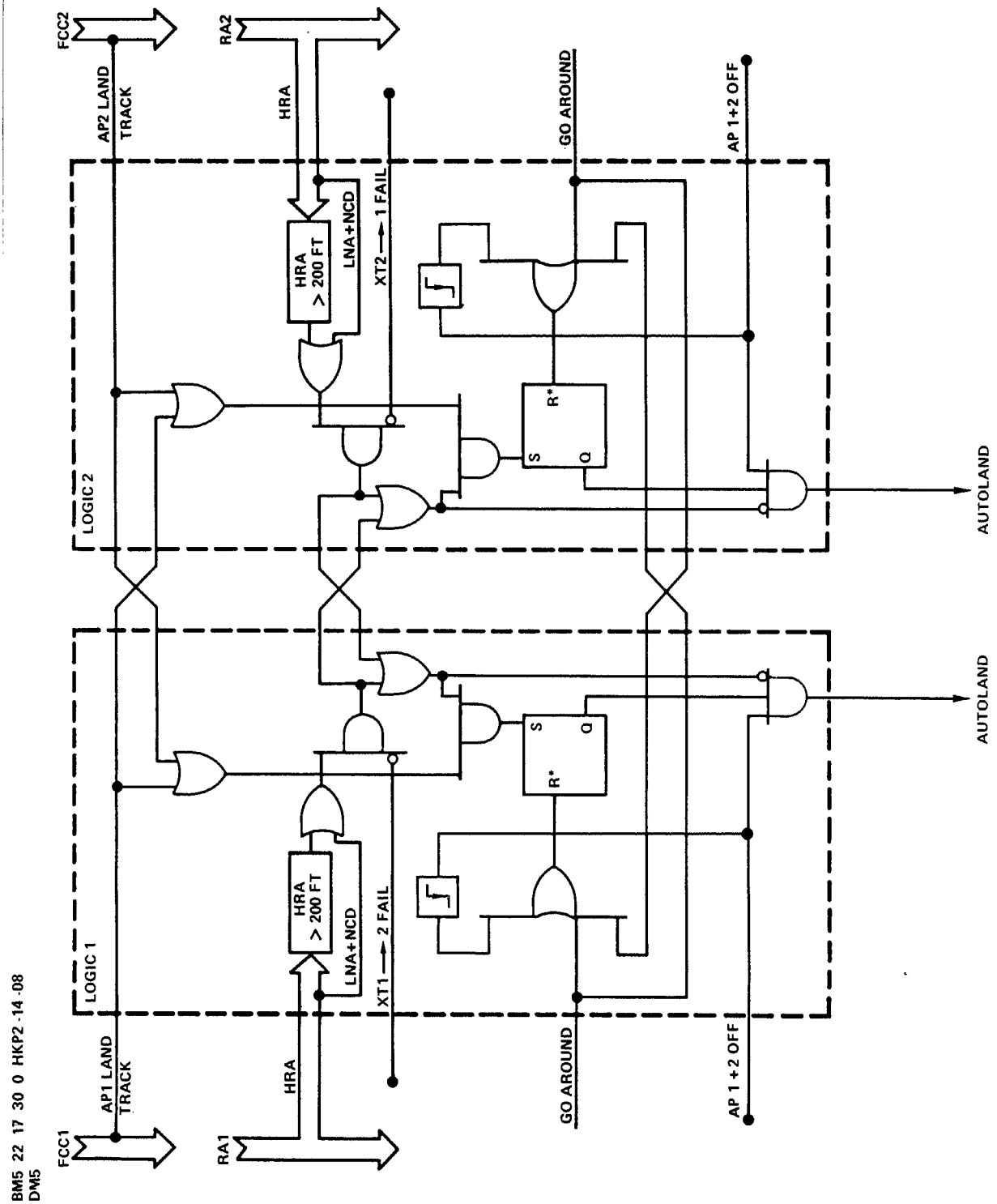
(Ref. Fig. 016)
(Ref. Fig. 017)
(Ref. Fig. 018)
(Ref. Fig. 019)
(Ref. Fig. 020)
(Ref. Fig. 021)



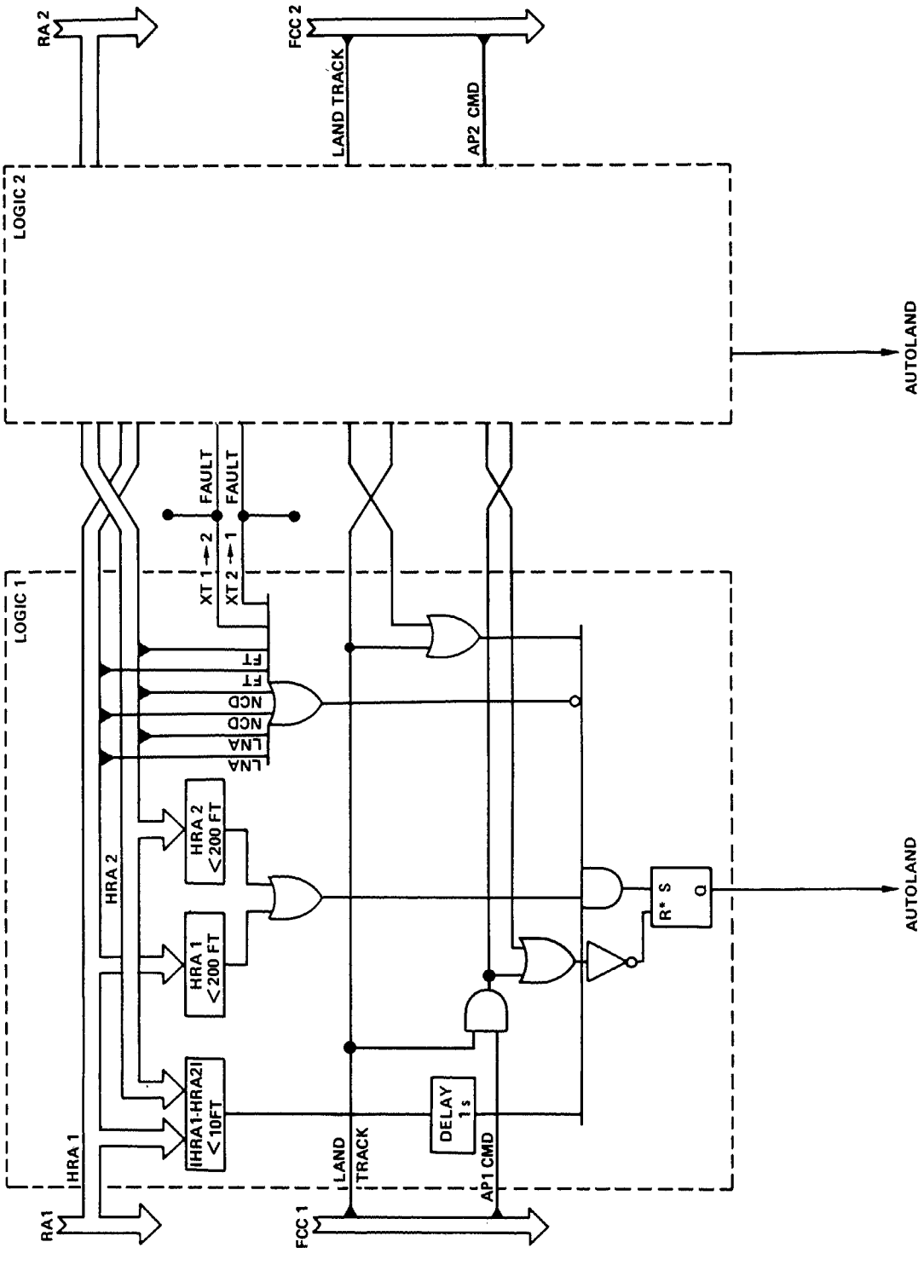
AUTOLAND Warning Logic
Figure 016



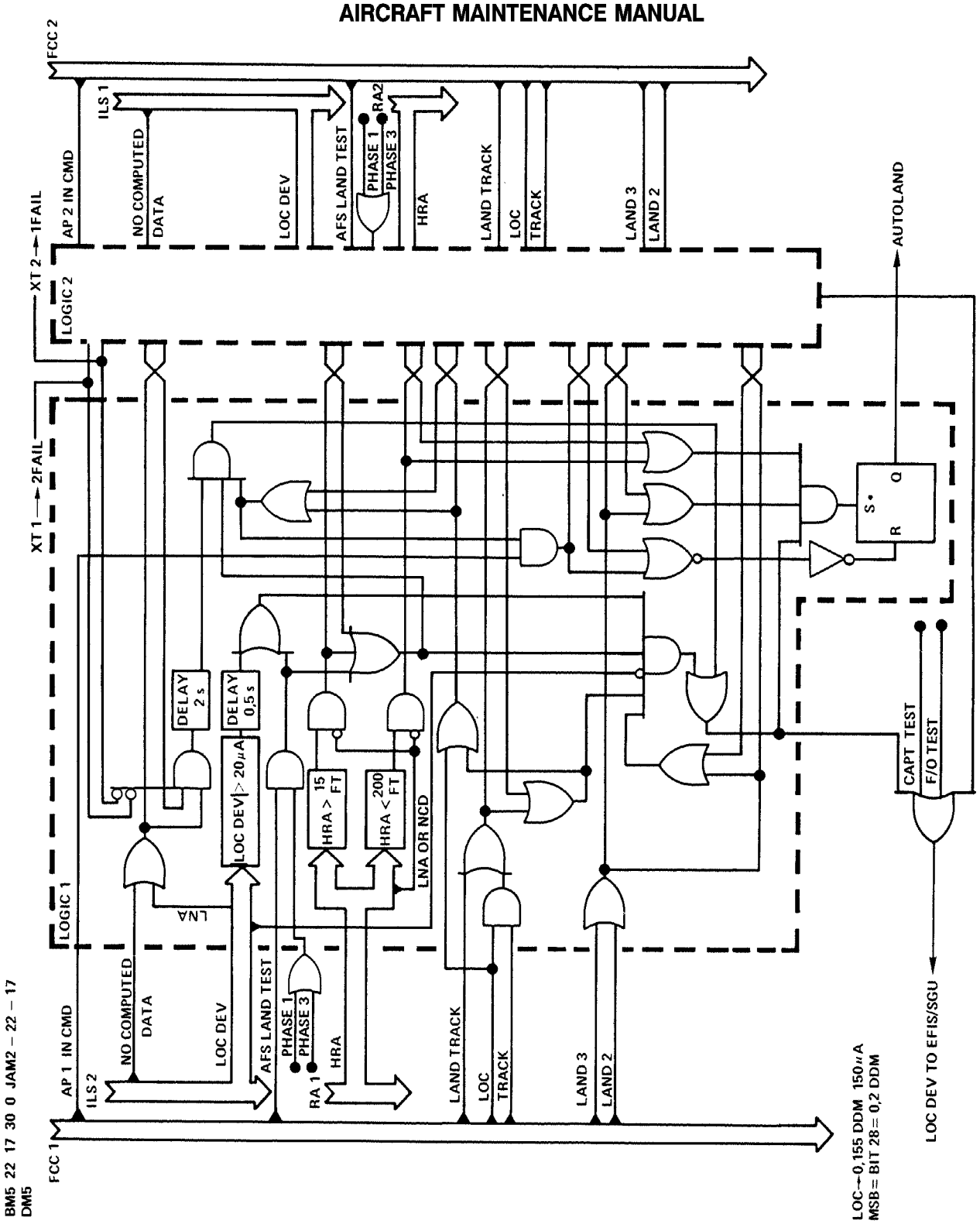
LONG FLARE Warning
Figure 017



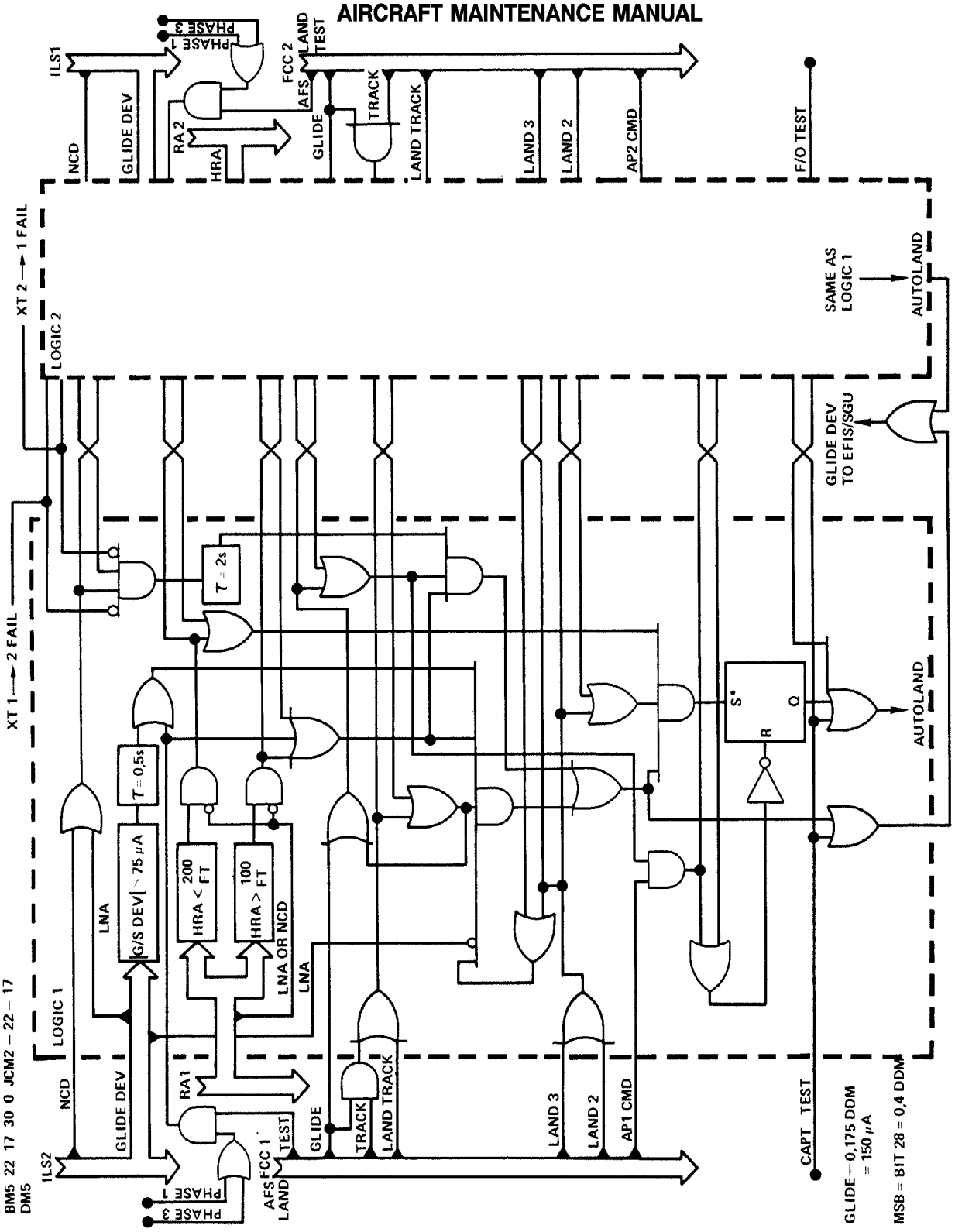
AP OFF Below 200 ft
Figure 018



RA Discrepancy
Figure 019



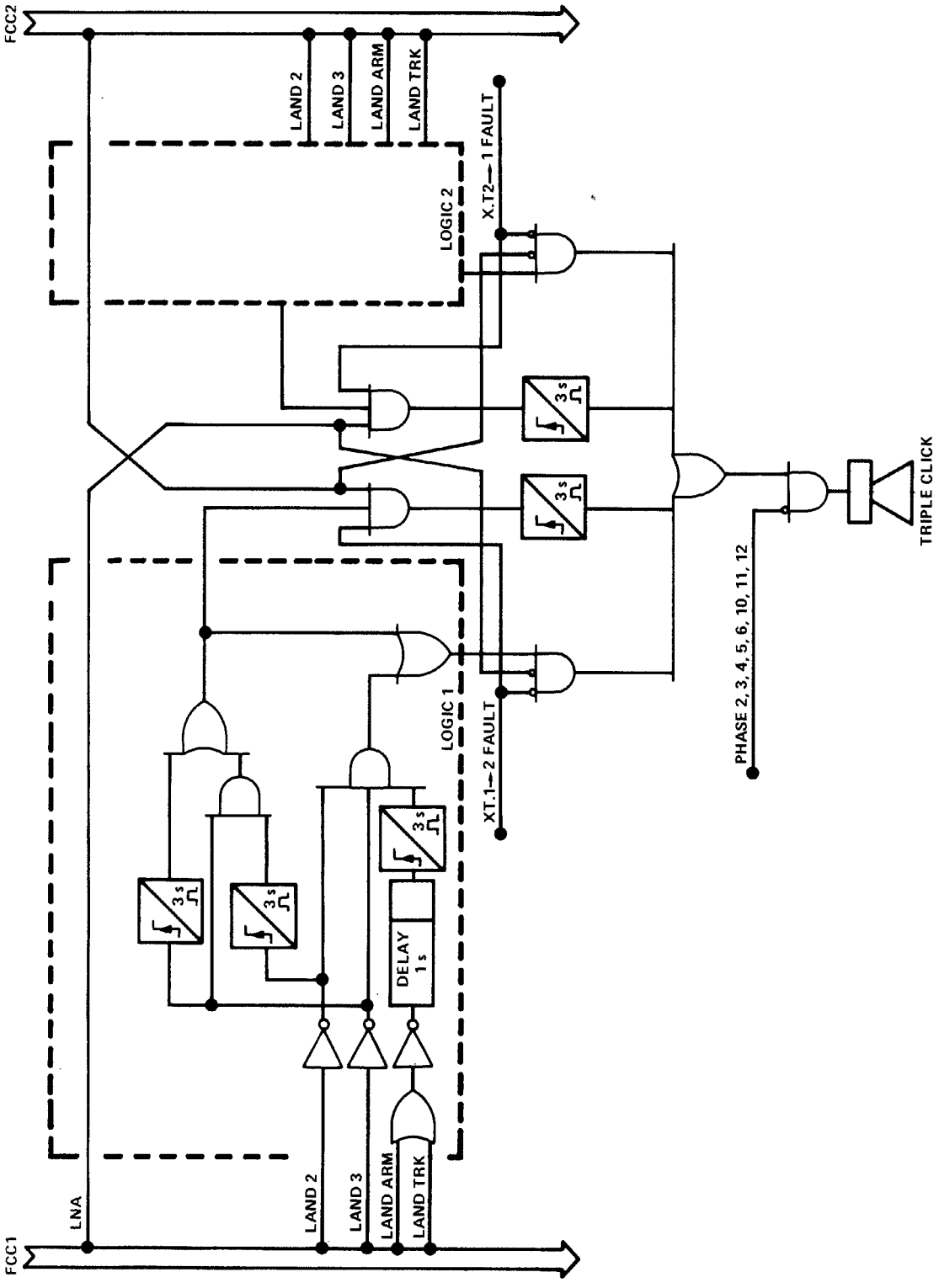
LOC Excessive Deviation Warning Logic
Figure 020



GLIDE Excessive Deviation Warning Logic
Figure 021

(7)Landing capability downgrading
A triple click aural warning is generated in the event of landing capability downgrading (e.g. LAND3 to LAND2).
The aural warning operating principle is shown in following figure. (Ref. Fig. 022)

(8)Loss indicating of decrab and roll out functions (LAND mode)
Loss of the decrab or roll out function in LAND phase results in disengagement of the AP/FD associated to this failure.



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DM5

Capability Change Warning Logic
Figure 022

C. Command generation

(1)Lateral axis

(a)LOC capture phase

The computation principle of the guidance law is given in chapter 22-17-20.

(b)LOC track phase

This law differs from the track law in LOC mode by the following characteristics :

- existence of a gain, function of the radio height, on the LOC deviation signal
- utilization of an accuracy integrator (accuracy input)
- utilization of the LOC beam derivative.

(c)align phase

The computation principle of the guidance law in the align phase is given in figure (Ref. Fig. 024).

The deviation between the aircraft heading and the runway axis is used to place the aircraft parallel to the runway axis just before touchdown (alignment is initialized at 30 ft.).

(d)roll out phase

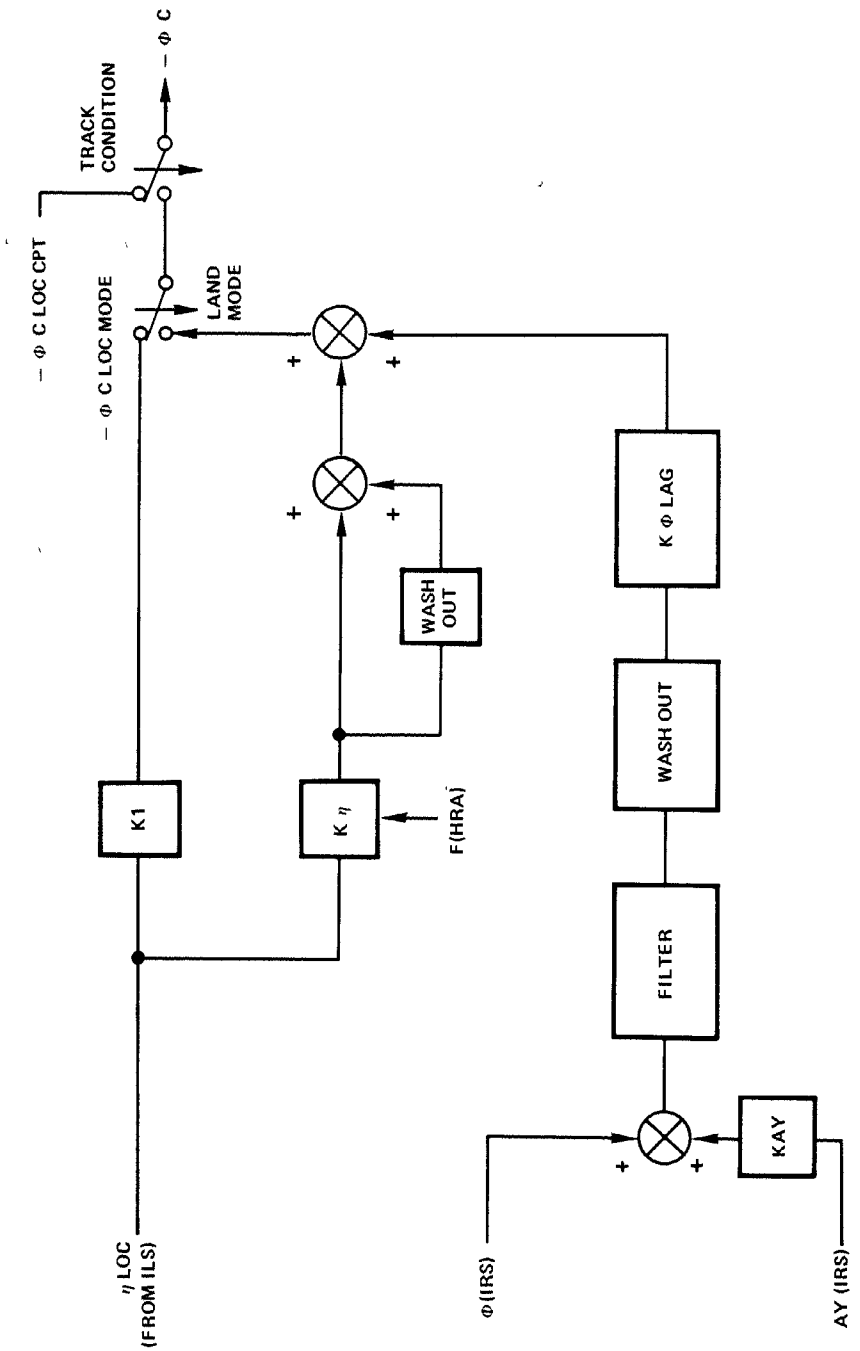
The computation principle of the guidance law used during roll out phase is shown in para 1.C(2).

(e)Not applicable.

(2)Longitudinal axis

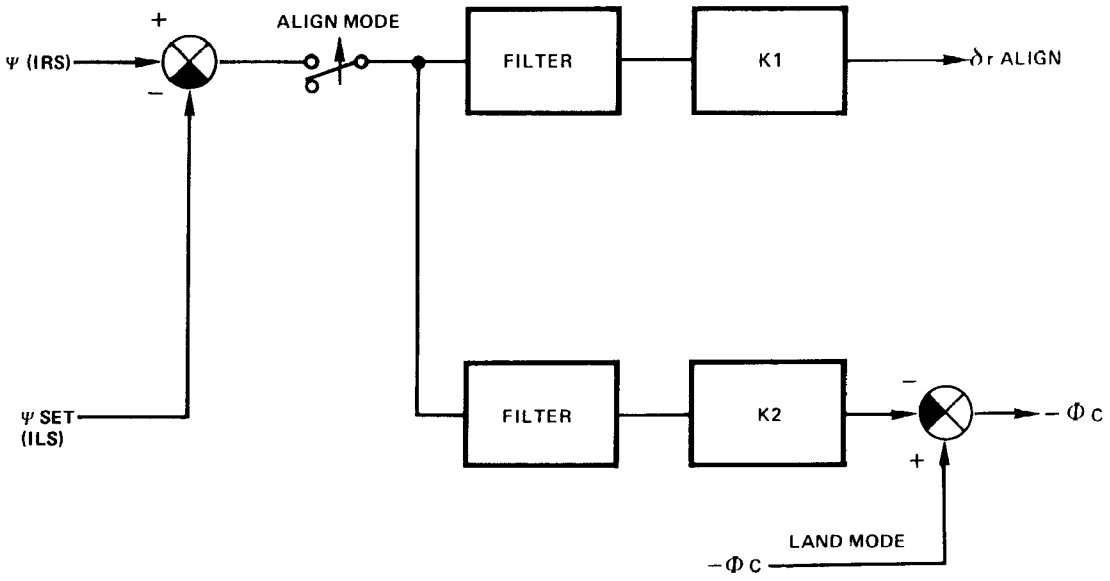
The guidance law principle of GLIDE capture and track phase and flare phase are given in following figures.

(Ref. Fig. 023)
(Ref. Fig. 024)
(Ref. Fig. 025)
(Ref. Fig. 026)



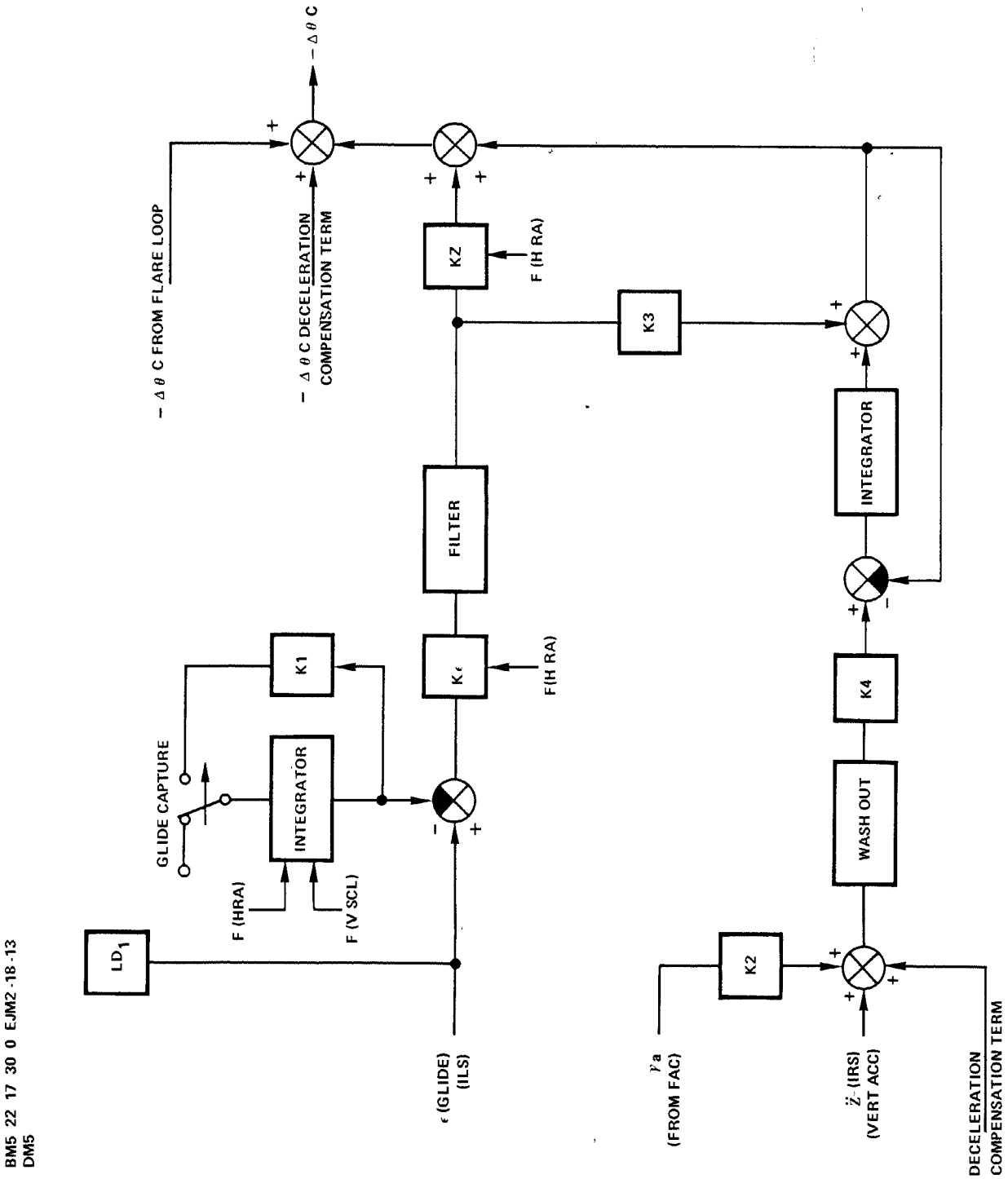
BM5 22 17 30 0 ECP2-18

LAND Mode - LOC TRACK Computation
Figure 023



BM5 22 17 30 0 EEM2 01

LAND Mode - Align Computation
Figure 024



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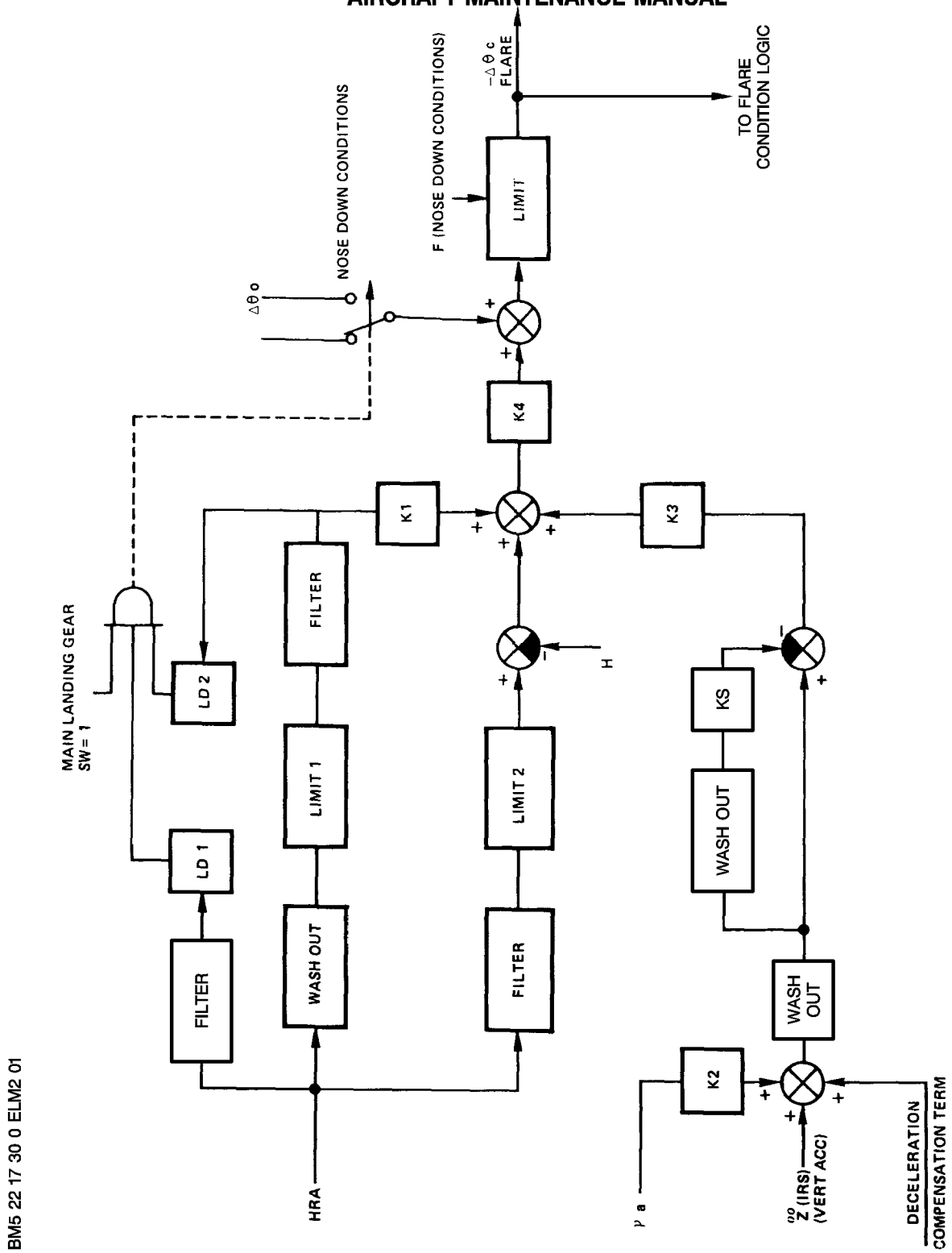
DM5

LAND Mode - Glide Capture and Track Mode Computation Principle
Figure 025



LAND Mode - Flare Computation
Figure 026

D. Display
Selection of the LAND mode is indicated by :
- illumination of the LAND pushbutton switch light on the FCU
- illumination on the PFD of the following various indications :
  . during the arming phase :
    * cyan G/S indication in the lower part of the section corresponding to the longitudinal modes
    * cyan LOC indication in the lower part of the section corresponding to the lateral modes.
      In addition, the active support mode indication remains visible (in green) until the LOC or GLIDE capture is effective.

R **ON A/C 001-100,

  . during the capture phase, the indications are changed into :
    * green LOC* indication : the lateral support mode is disengaged and the corresponding mode indication is cancelled.
    * green G/S* indication : the longitudinal support mode is disengaged and the corresponding mode indication is cancelled.
  The LOC* indication can be displayed at the same time as the cyan G/S indication (G/S mode armed) but the G/S capture phase can start only after the LOC capture phase.

R **ON A/C 101-200,

  . during the capture phase, the indications are changed into :
    * green LOC* indication : the lateral support mode is disengaged and the corresponding mode indication is cancelled.
    * green G/S* indication : the longitudinal support mode is disengaged and the corresponding mode indication is cancelled.

R **ON A/C ALL

  . during the track phase and above 400 ft. the indications are replaced by :
    * LOC (green)
    * G/S (green)
  . during the track phase and below 400 ft. the green LOC and G/S indications are replaced by the green LAND indication across the line dividing the two sections corresponding to the longitudinal and lateral mode annunciations.
  . in flare phase, the green LAND indication is replaced by the green FLARE indication.
  . in ground roll out phase, the FLARE indication is replaced by the green ROLL OUT indication.

In addition, the landing capability is displayed on the PFD, as soon as the LAND mode is selected, by the following indications :
- CAT 1
- CAT 2
- CAT 3

NOTE : Any possible landing capability downgrading is indicated by a

click aural warning.

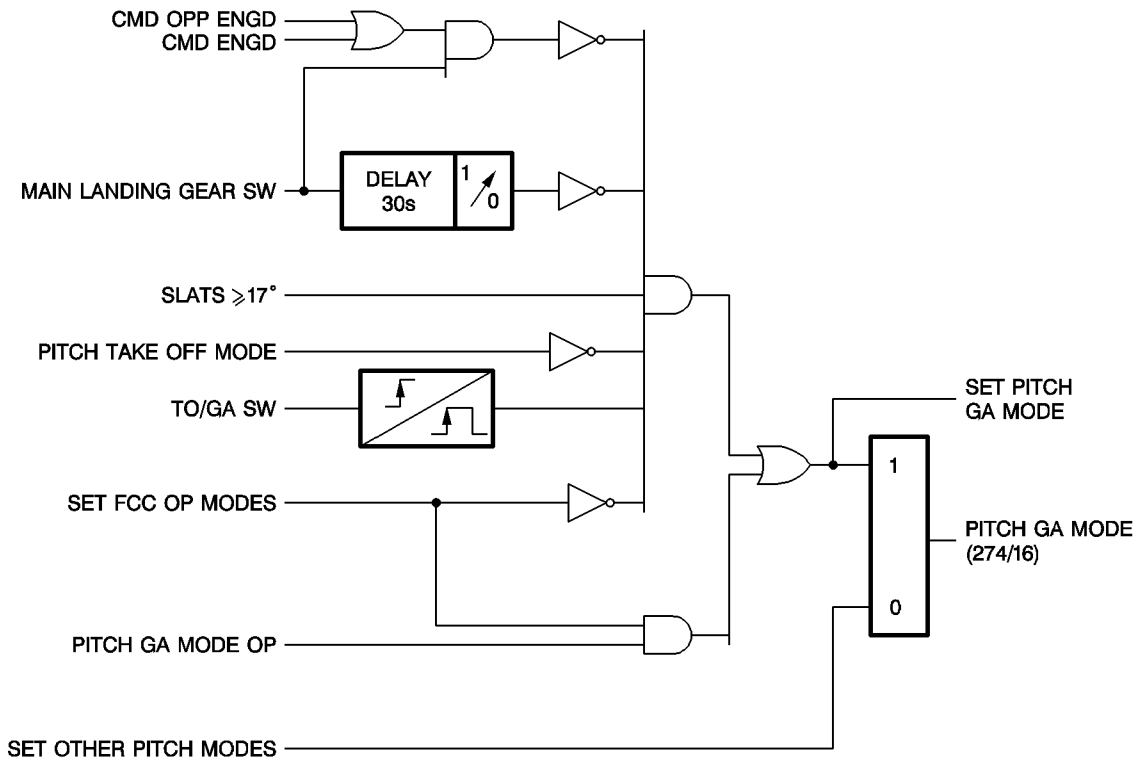
3. GO AROUND Mode (GA)

A. Characteristics
This mode allows to perform a go around when in approach configuration by providing the following functions :
- longitudinal guidance by using the same law as the takeoff law
- lateral guidance (heading hold)
- autothrottle engagement in limit thrust hold mode (GA).

B. Operation (Ref. Fig. 029)
(1)Engagement/disengagement
The GO AROUND mode is engaged through the TO/GA lever(s) hinged on the throttle control levers with the condition that the slats are extended (it is to be noted that the slat extension forces selection of GA on the TRP).
When the GO AROUND mode is engaged, the FD bar display is forced on the PFDs, even if the flight path vectors (FPV) are visible.
Engagement of the GO AROUND mode is possible in dual AP configuration.
The GO AROUND mode can only be disengaged by selecting another mode.
Disengagement is possible one axis after the other. In this case, if both APs are engaged, AP2 disconnects.
The engage logic conditions of the GO AROUND mode are given in following figures.
(Ref. Fig. 027)
(Ref. Fig. 028)
(Ref. Fig. 029)

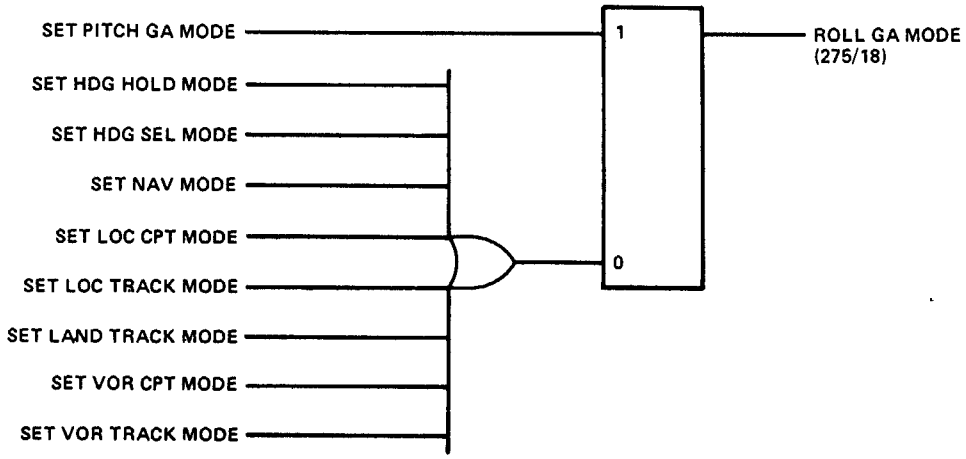
C. Command generation
(1)Pitch axis
The guidance law is the same as the takeoff law and enables to hold :
- V REF + 10 kt with both engines running
- the aircraft speed if it is greater than V REF when engine failure occurs or V REF in the other case.

In addition, the law includes specific devices of maximum pitch attitude protection and pre-command described in the TAKEOFF mode.
(2)Lateral axis
The computation principle of the guidance law is given, for the lateral axis, in figure (Ref. Fig. 030). This law enables to hold the path followed by the aircraft during approach.



BM5 22 17 30 0 FZM2 01

Pitch Go Around Mode Logic
Figure 027



BM5 22 17 30 0 GAP2 14 08
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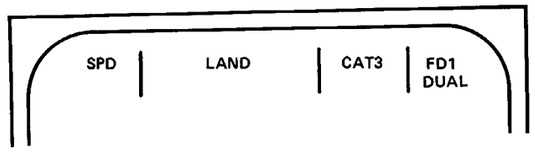
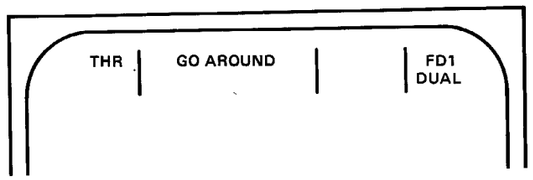
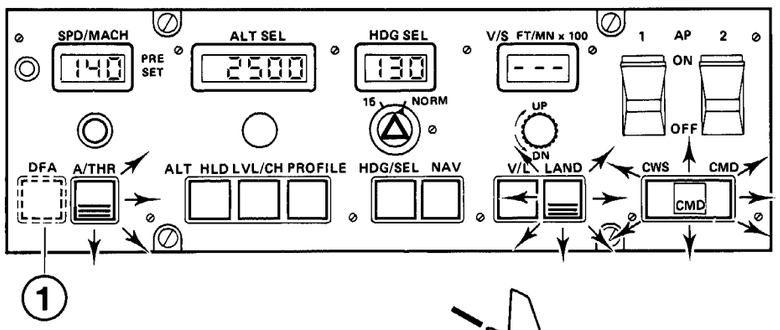
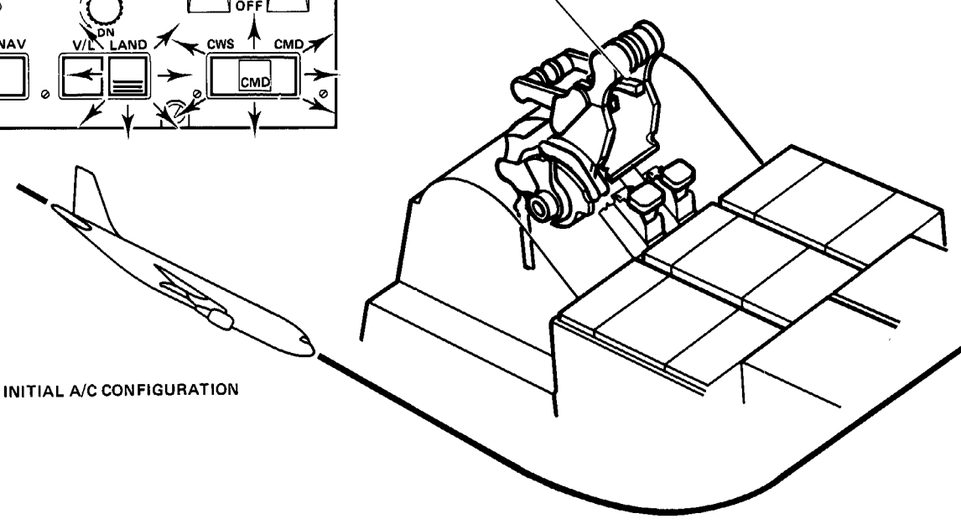
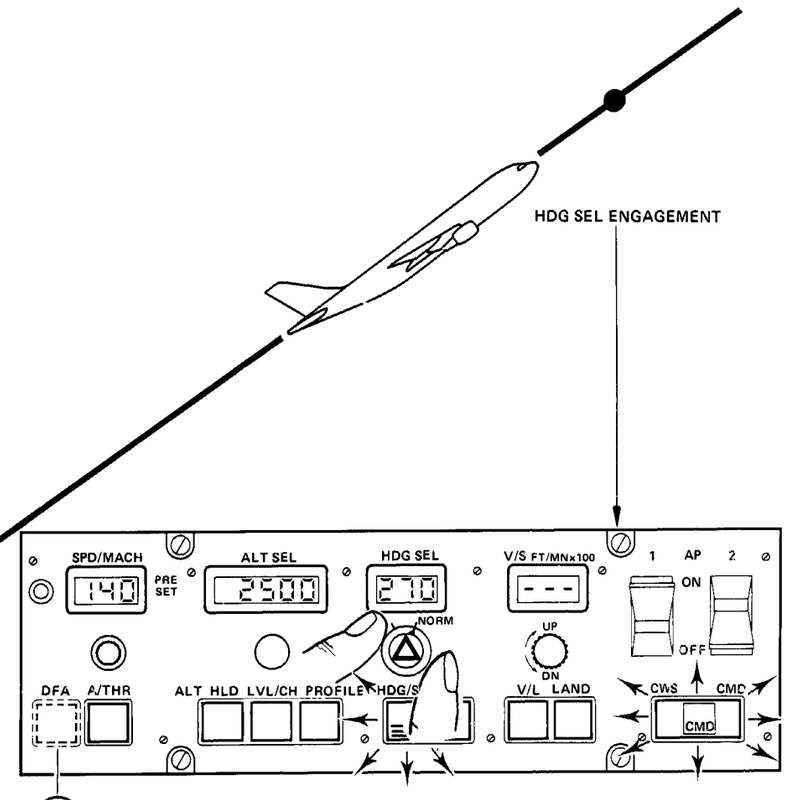
Lateral Go Around Mode Logic
Figure 028

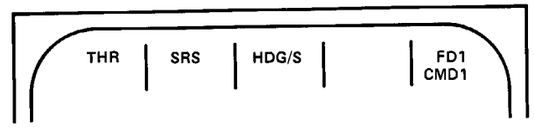







PULL TO/GA LEVERS
GO AROUND MODE IS ENGAGED FOR AP/FD
ATS CHANGES TO THR MODE
ACQUIRES AND HOLDS THR LIMIT (GA)
INITIAL A/C CONFIGURATION





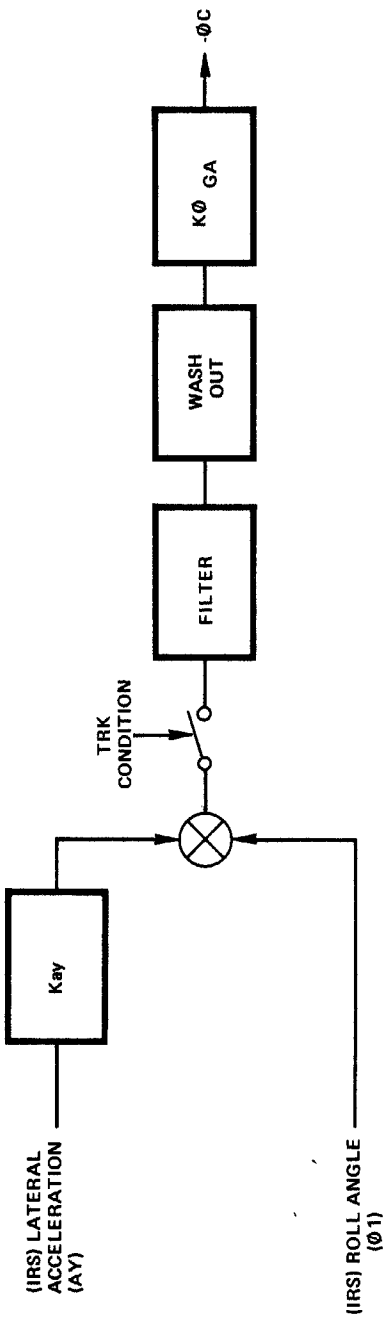
(1) DELAYED FLAPS APPROACH (DFA) (IF FITTED)

BM5 22 17 30 0 FVM2-14-08
DM5

Go Around Mode Operation
Figure 029

D. Display
When the **GO AROUND** mode is engaged, the green **GO AROUND** indication is displayed across the dividing lines of the **PFD** sections corresponding to the annunciation of **AP/FD** longitudinal and lateral mode engagement.
As the **GO AROUND** mode is engaged on both axes :
- engagement of a lateral mode will be taken into account and the **PFD** will show :
  . **SRS** in the longitudinal mode section
  . the indication corresponding to the lateral mode selected in the lateral mode section.
- engagement of a longitudinal mode will be taken into account and the **PFD** will show :
  . the indication corresponding to the longitudinal mode selected in the longitudinal mode section
  . **HDG** in the lateral mode section.

(Ref. Fig. 030)



BM5 22 17 30 0 FXM2.24.19
DM5

Roll Go Around Computation
Figure 030

FLIGHT CONTROL COMPUTER - DESCRIPTION AND OPERATION

1. Introduction

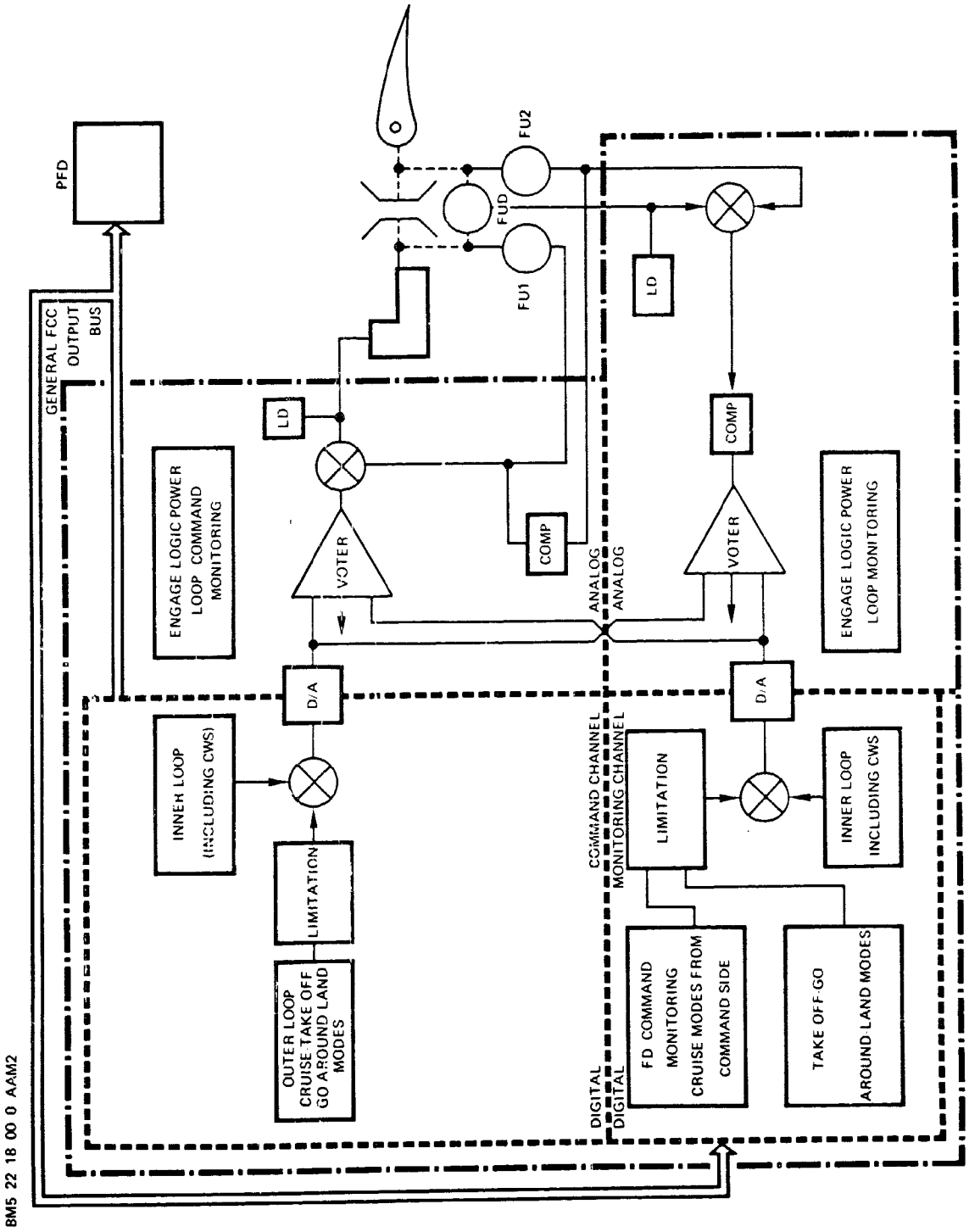
A description of the flight control computer (FCC) (dimensions, weight, power consumption, etc.) is given in 22-10 : GENERAL
The following section deals with the internal description of the FCC with its :
- digital part
- analog part

and its organization concerning safety.
Each FCC includes two computation channels (command and monitoring). Each channel is composed of a digital part and an analog part
(Ref. Fig. 001)
The FCC organization complies with the principle described in 22-00 : AUTO FLIGHT, P. Block 1.

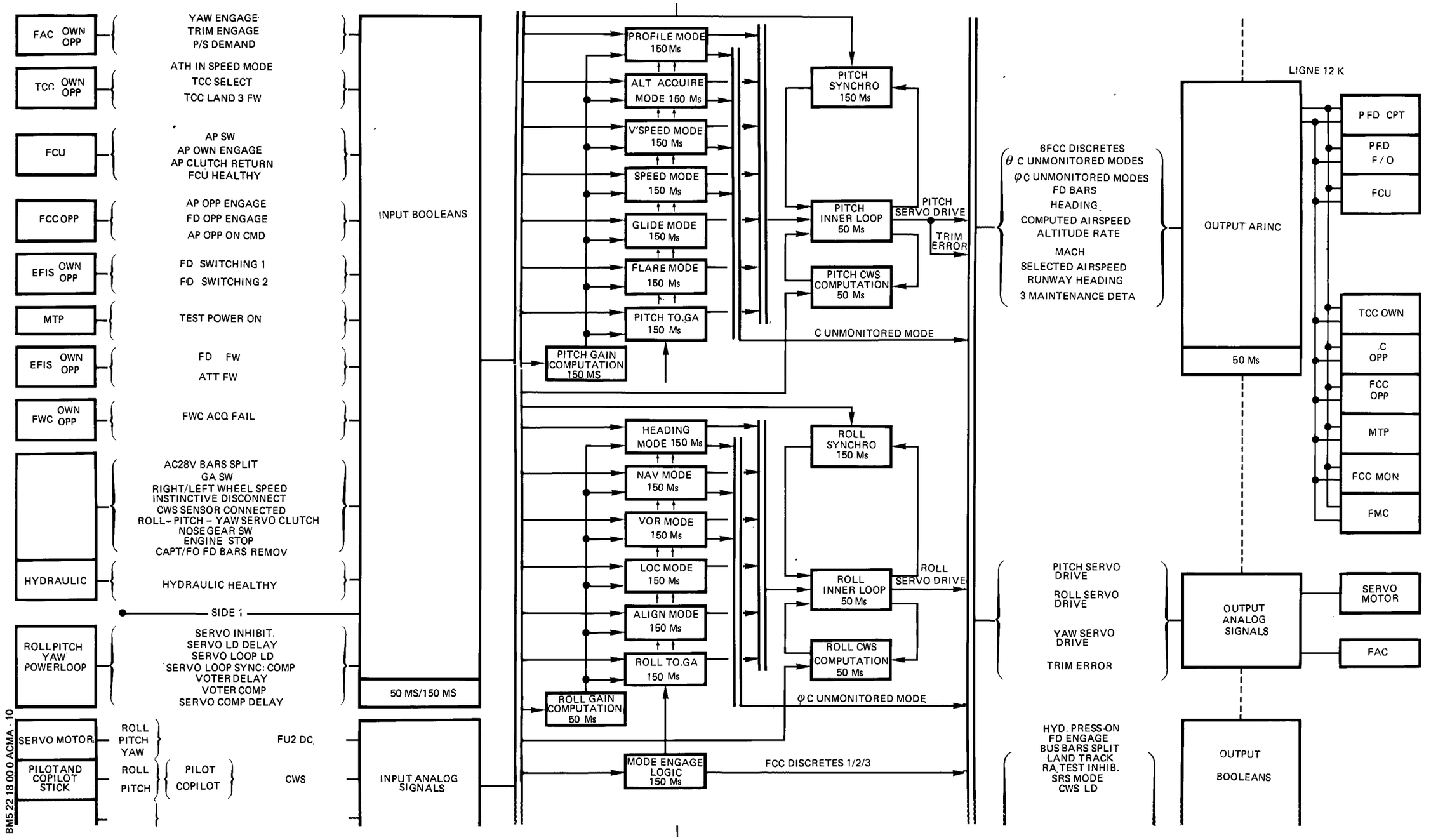
2. Digital Part Architecture

Each computation channel utilizes common boards described in 22-00 : AUTO FLIGHT :
- central processing unit (CPU)
- memory board
- analog/digital and digital/analog conversion
- ARINC management board

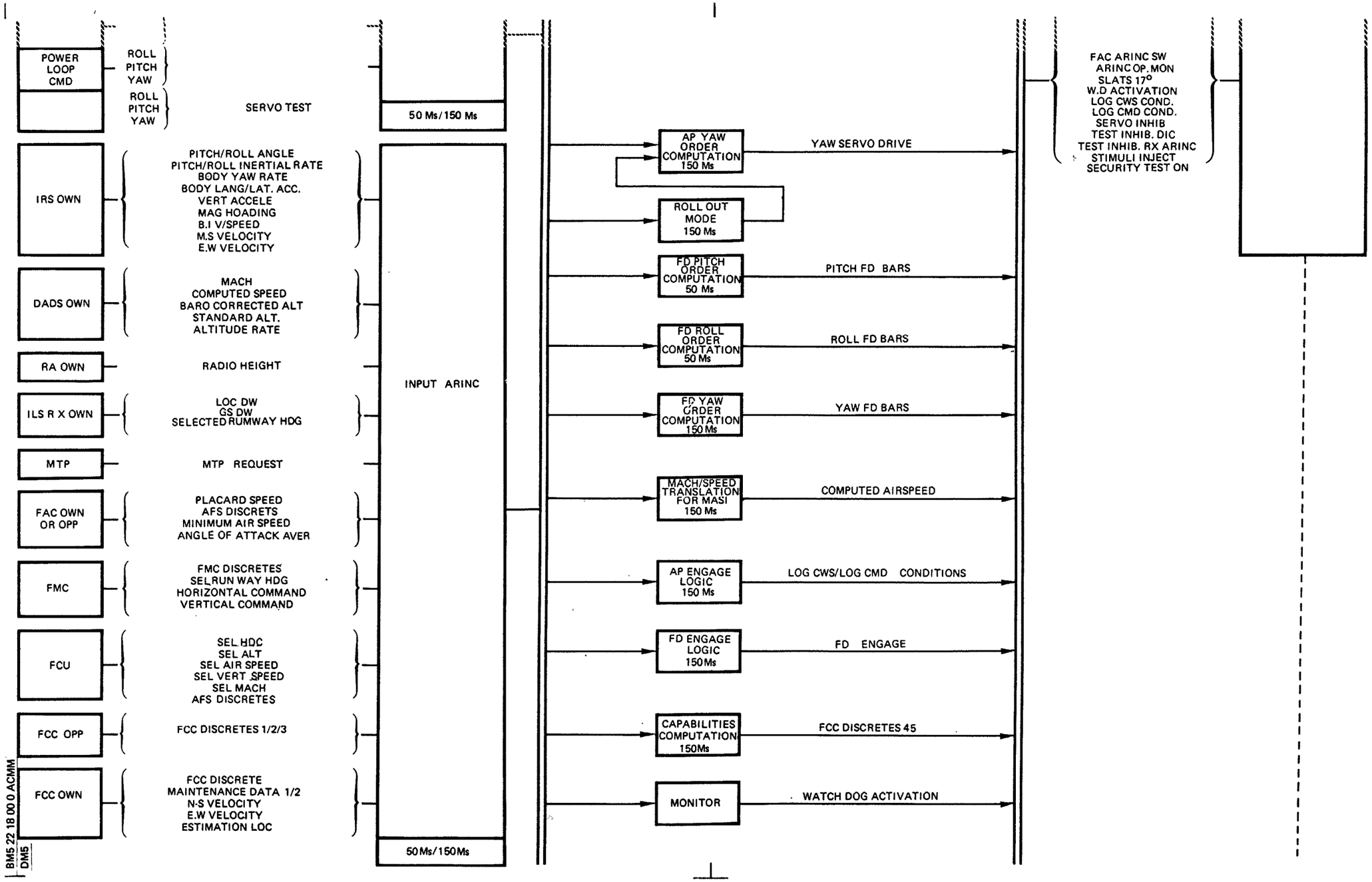
The FCC software architecture (command and monitoring channels) is given in figures (Ref. Fig. 002, 003).
It is to be noted in particular that the software is dissymmetrical. Moreover the following figures show how the functions are duplicated (and therefore monitored).



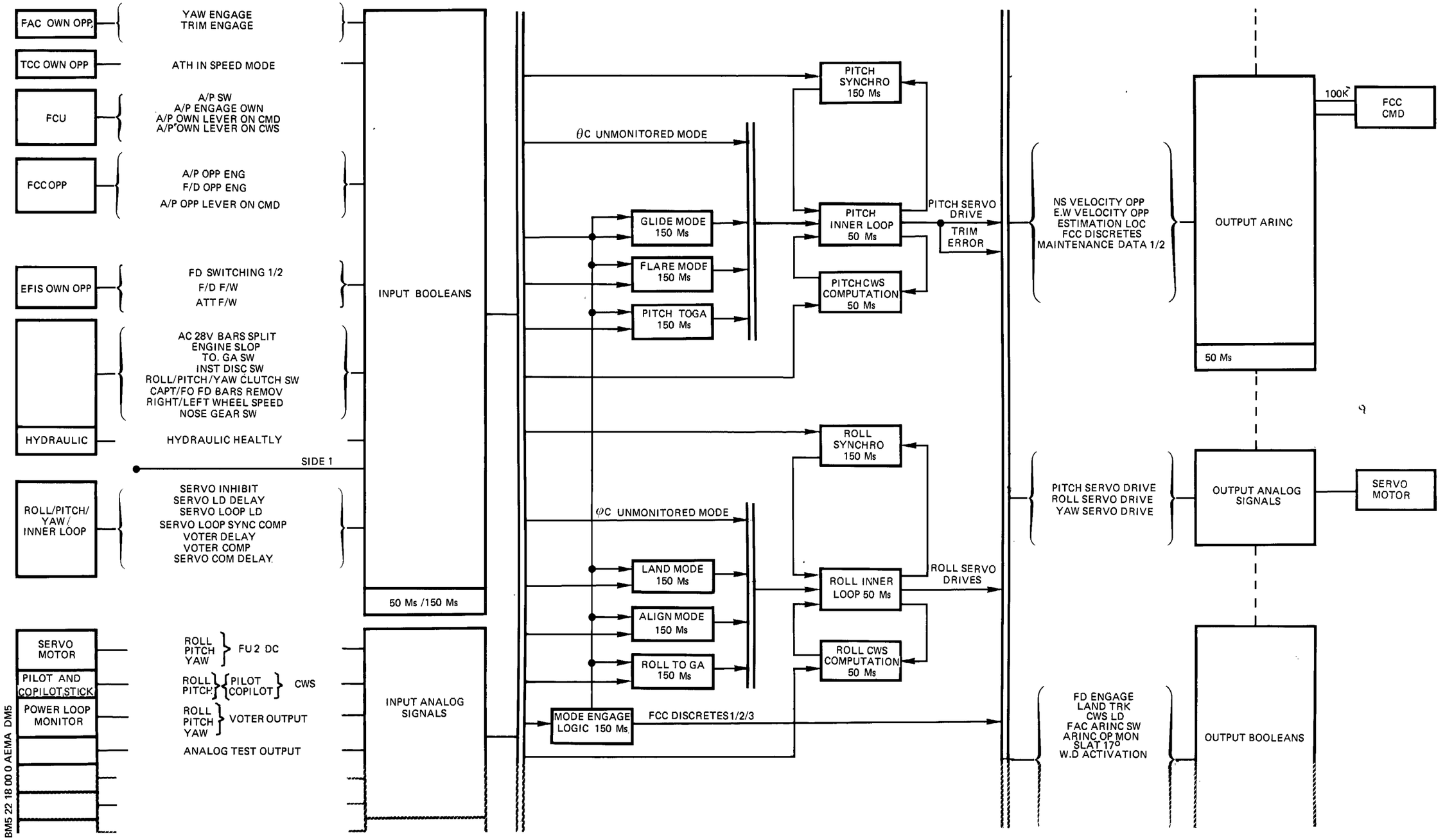
FCC Organization
Figure 001



Software Block Diagram - Command Channel (Sheet 1/2)
Figure 002

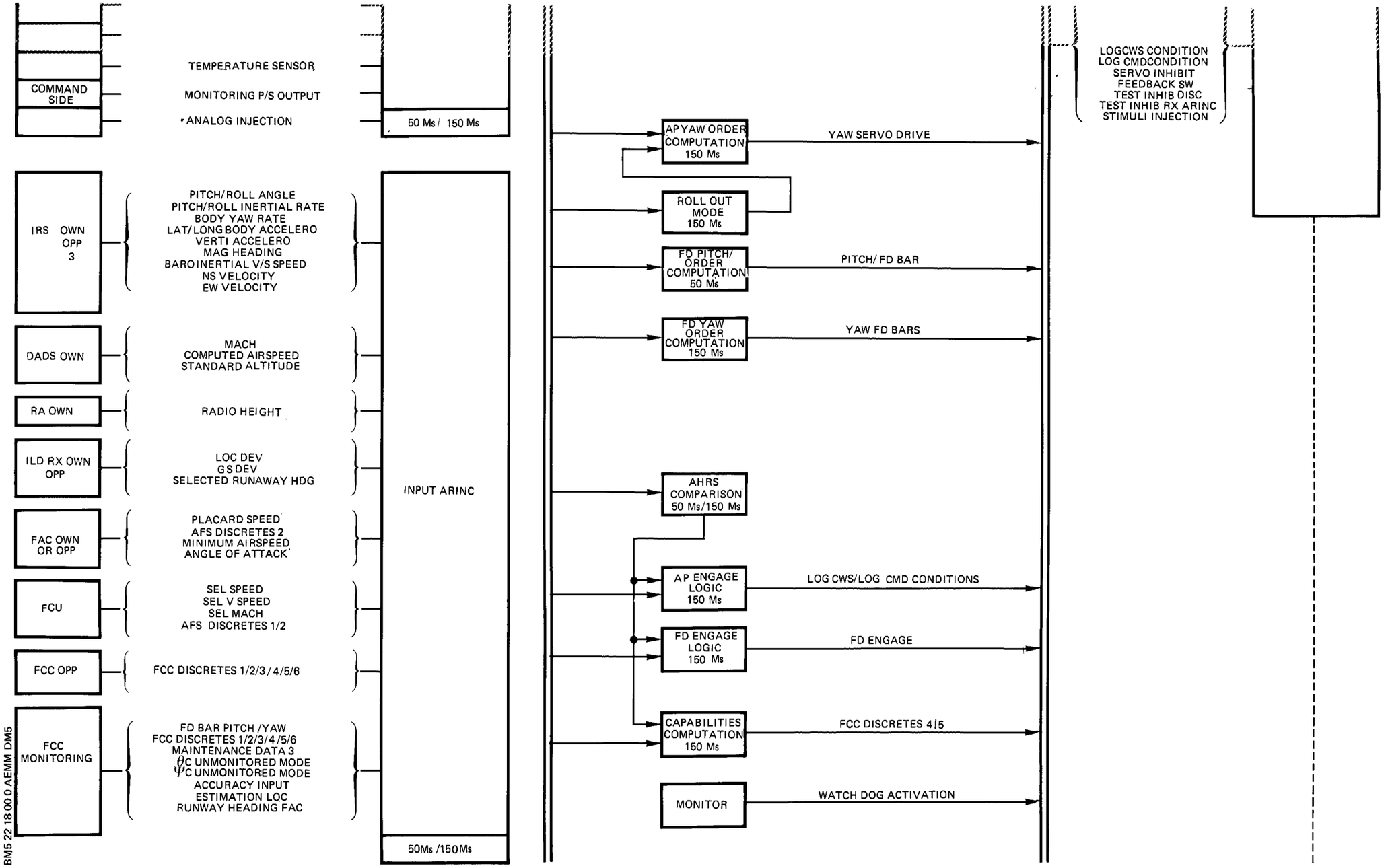


Software Block Diagram - Command Channel (Sheet 2/2)
Figure 002



Software Block Diagram - Monitoring Channel (Sheet 1/2)
Figure 003

R



Software Block Diagram - Monitoring Channel (Sheet 2/2)
Figure 003

3. Analog Part Architecture

The FCC analog part processes the power loop command and monitoring as well as the hard-wired engage logic. After digital/analog conversion, a voter circuit (described in 22-00) enables to select the output AP command between the command computed by the command channel, the command computed by the monitoring channel and a null fixed command.

The voter comparators monitor the computations generated by the command and monitoring channels.

The AP power loop computation and monitoring principle is given in figure (Ref. Fig. 004)

The command voter output is opposed to the control surface position (given by a synchro signal (see below). The error signal thus obtained is amplified through a current amplifier and sent to the AP actuator servovalve of the axis concerned.

A current command is thus performed.

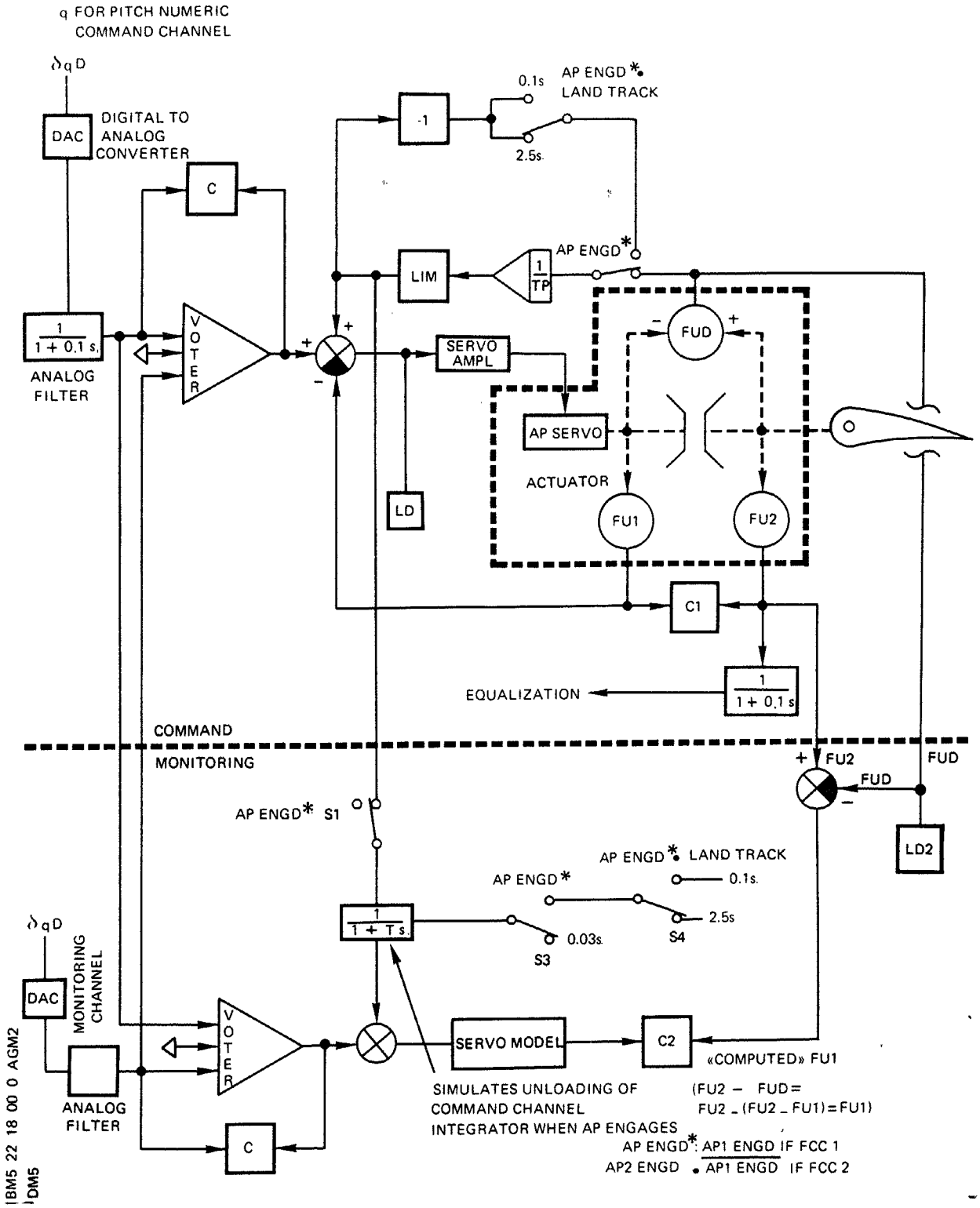
Three position feedback signals are transmitted by each actuator.

These are synchro signals :

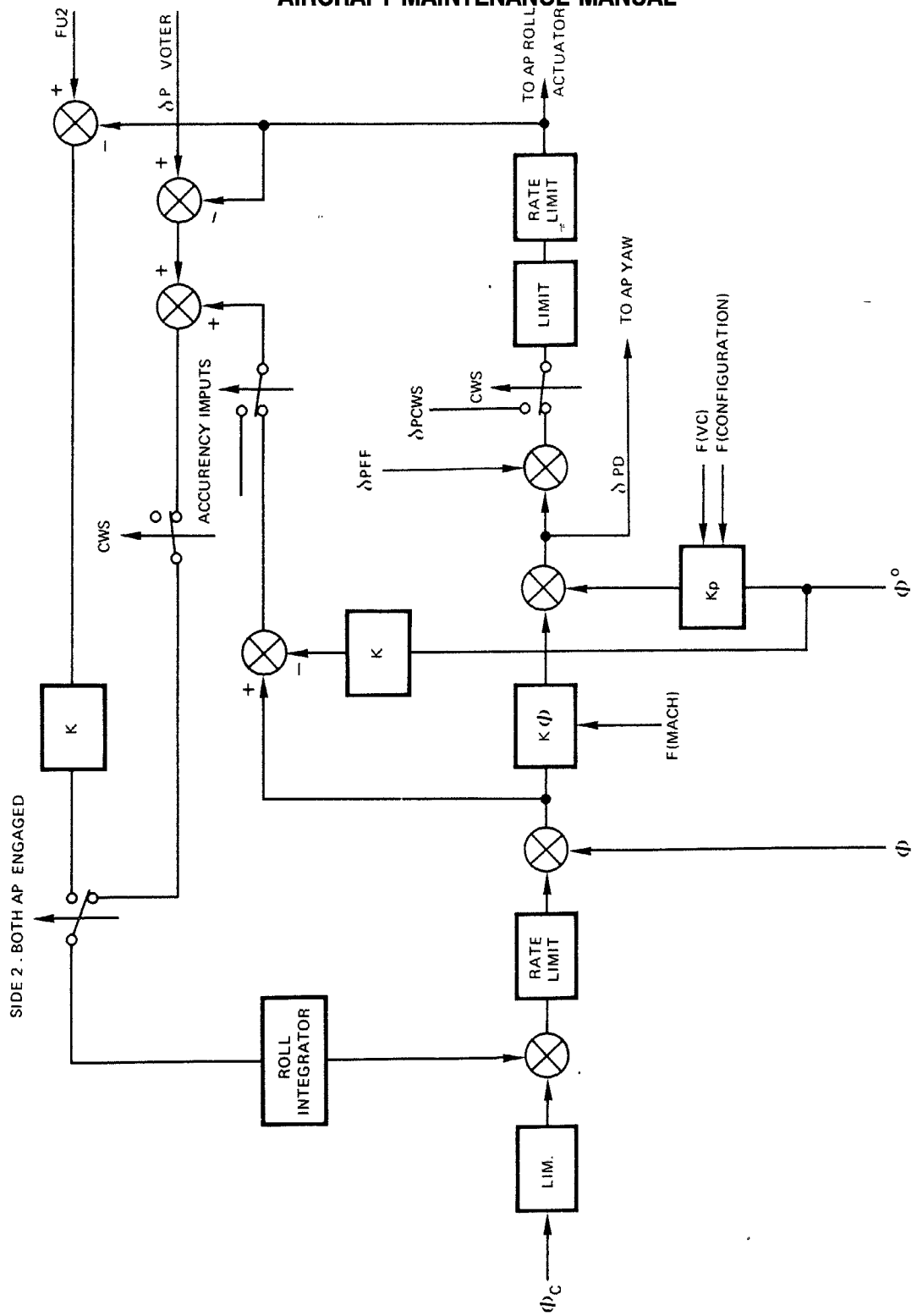
- FU1 (follow up 1) provides the slaving feedback.
- FU2 gives the position of the actuator output shaft, that is the control surface position ; this signal monitors the clutch, when the clutch is engaged : FU1 = FU2.
- FUD (follow up differential) = FU2 - FU1.
  This differential synchro transmits the signal which synchronizes the actuator with the control surface position when the corresponding AP (on the same side as the actuator concerned) is not engaged.

When the AP is disengaged, the AP output is synchronized with the control surface position. To this end, the signal coming from the FUD synchro (representative of the deviation between the AP command and the control surface position) is transmitted through an 0.1 s time constant integrator. The signal coming from the integrator is summed to the AP commands of the command and monitoring channels.

When the AP is engaged, the integrator is looped and discharges slowly under
R normal conditions (T = 2.5 s) and rapidly if in LAND TRACK mode (T = 0.5 s). This avoids jerks when the AP is engaged with the aircraft configuration non-stabilized (for the yaw axis there is only one time constant value of the integrator discharge). In addition, on engagement, switch S1 is open and a low-pass filter simulates the integrator discharge (for the pitch and roll axes the time constant value is contingent on AP engagement in LAND TRACK). For the roll and yaw axes the presence of an integrator in the basic loop (Ref. Fig. 005, 006) requires an additional synchronization achieved by re-injecting to the basic loop integrator input, a term equal to the difference between the voter output command and the computed output command (prior to vote).

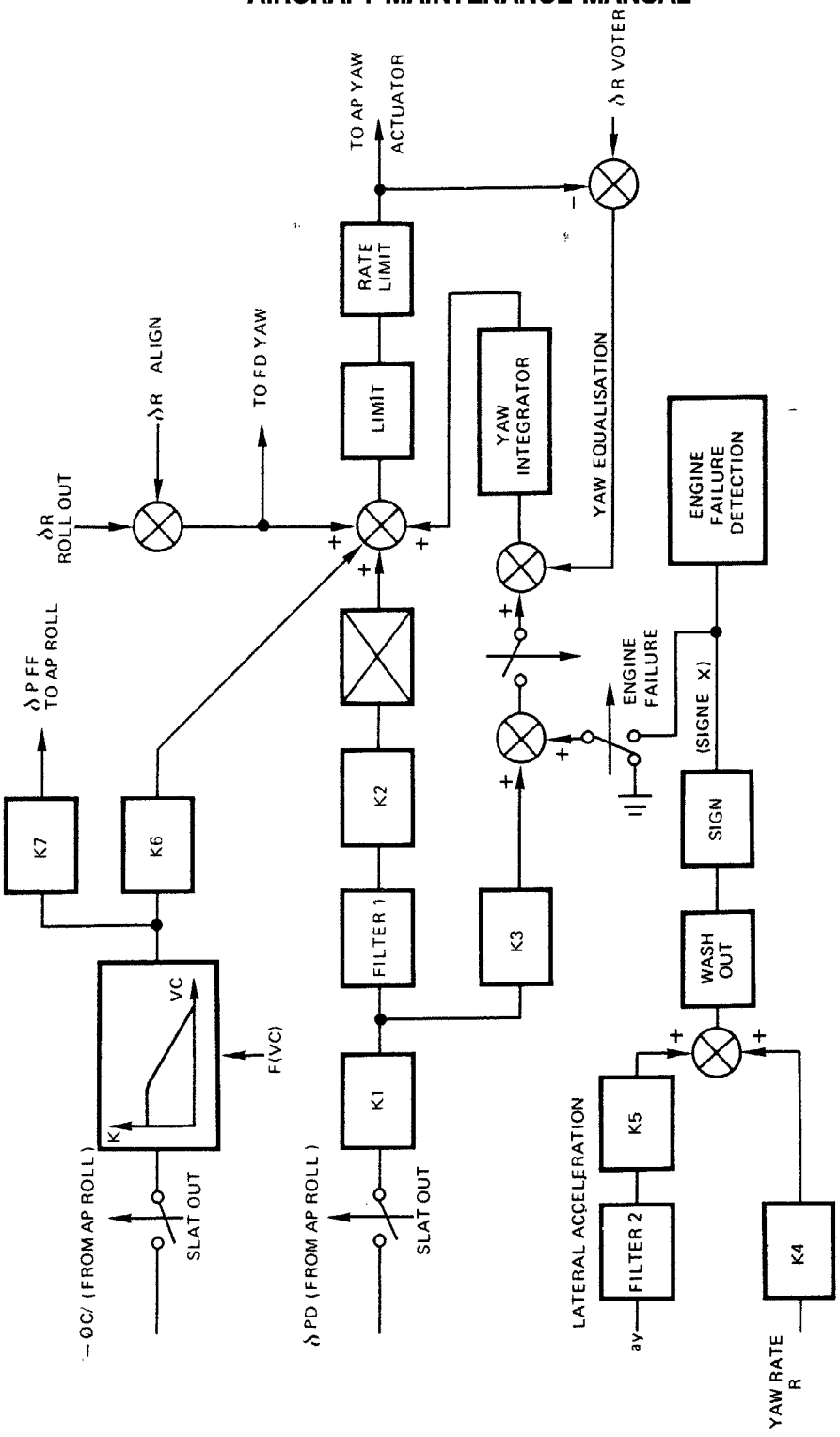


FCC - Power Loop
Figure 004



BM5 22 18 00 0 AJM2
DM5

Roll AP Basic Demand Loop
Figure 005



BM5 22 18 00 0 ALM2
DM5

Yaw AP Basic Demand Loop
Figure 006

Power loop monitoring consists in comparing (through comparator (2) the command voter output via a servo-model circuit and the actuator position computed by the difference FU2 - FUD = FU1. Comparator C2 is inserted in the monitoring channel. Level detector LD1 monitors slaving in the command channel. Comparator C1 inserted in the command channel and level detector LD2 inserted in the monitoring channel monitor synchronization and prohibit AP engagement if the difference FU1 - FU2 is too great (after engagement, the comparator thresholds are increased). The following table gives the threshold and delay values of the power loop comparators.

FCC Comparators - Characteristics

	NAME	THRESHOLD	DELAY	TEST CONNECTOR OUTPUT
	VOTER COMPARATOR			
R	. PITCH	4° control surface	1 s	
R	. ROLL	6° control surface	2 s	
	. YAW	4.4° control surface	3 s	
	POWER LOOP COMPARATOR C2			
R	. PITCH	2° control surface	0.22 s	
	. ROLL	6° control surface	0.10 s	
	. YAW	4.4° control surface	0.20 s	
	SYNCHRONIZATION COMPARATOR C1			
R	. PITCH	1° ; (2° after clutching)	0.5 s	
R	. ROLL	1.6° ; (6° after clutching)	0.5 s	
R	. YAW	4.4°	0.5 s	
R	LEVEL DETECTOR LD43 ON SLAVING ERROR			
	. PITCH	2° control surface	0.22 s	
	. ROLL	6° control surface	0.10 s	
	. YAW	4.4° control surface	0.20 s	
	SYNCHRONIZATION			
R	LEVEL DETECTOR LD42			
	ON FUD			
R	. PITCH	0.675° ; (2° after clutching)	0.5 s	
R	. ROLL	1.8° (6° after clutching)	0.5 s	
R	. YAW	4.4°	0.5 s	

Automatic Change Over Principle

In cruise, one AP only can be engaged at a time (priority being given to the last one engaged). When one AP is disengaged, it is synchronized on the control surface positions following the principle described above (synchronization at the level of the power loop).

In approach, as soon as the LAND Mode is selected, both APs can be engaged. AP1 and AP2 actuators are clutched byt AP1 has priority and AP2 synchronizes on AP1 commands. AP2 synchronization is achieved :

- on one hand, at the level of the power loop following the principle described above

R - on the other hand, (for pitch, roll and yaw AP) by synchronizing AP2 control command on AP1 command.

The deflection command computed by AP2 is compared with the control surface position (FU2) controlled by AP1.

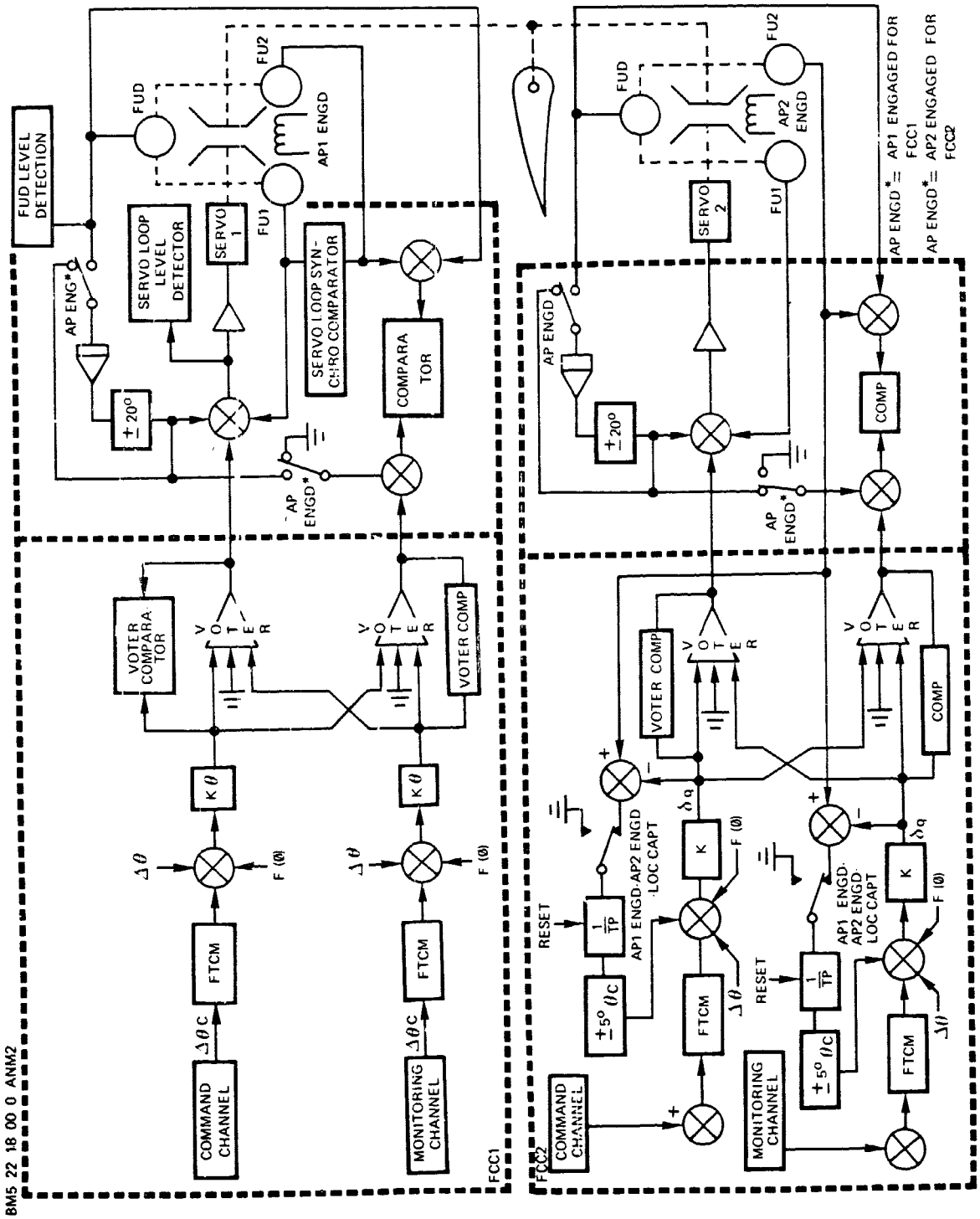
The deviation FU2 - command computed by AP2 is transmitted to an integrator which synchronizes AP2 output on FU2 signal which is representative of the deflection ordered by AP1.

R This additional synchronization of AP2 in dual AP configuration is immediate. When AP1 is disengaged, AP2 becomes active and synchronizations are then cancelled.

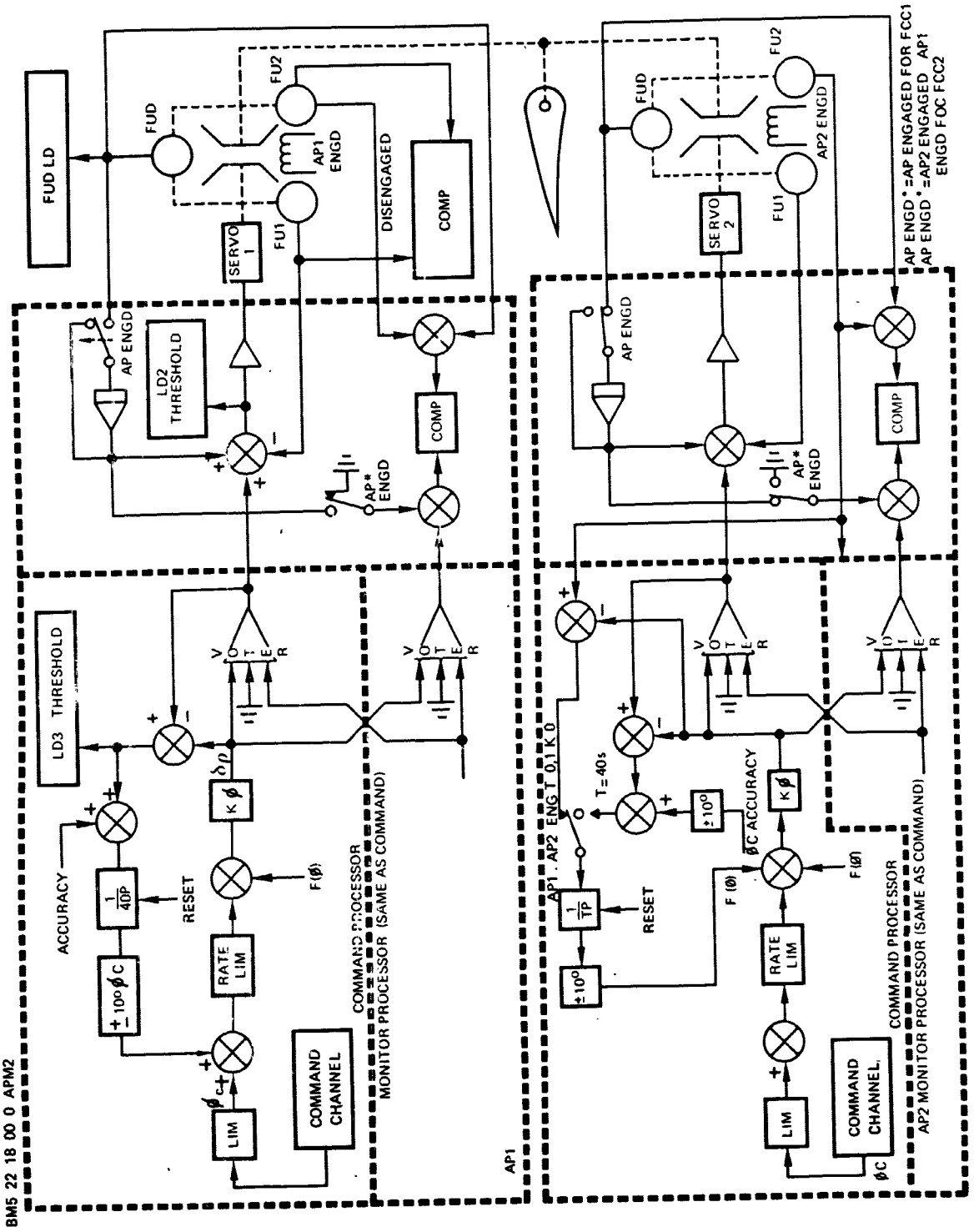
Hard-Wired Logic

The engage logic is duplicated and utilizes hard-wired logic circuits located on the FCC analog boards. The computation principle of this logic is described in 22-16, P. Block 1. It is to be noted in particular that the clutching lines of the AP actuators are interrupted by two independent relay contacts associated with the AP engage lever.

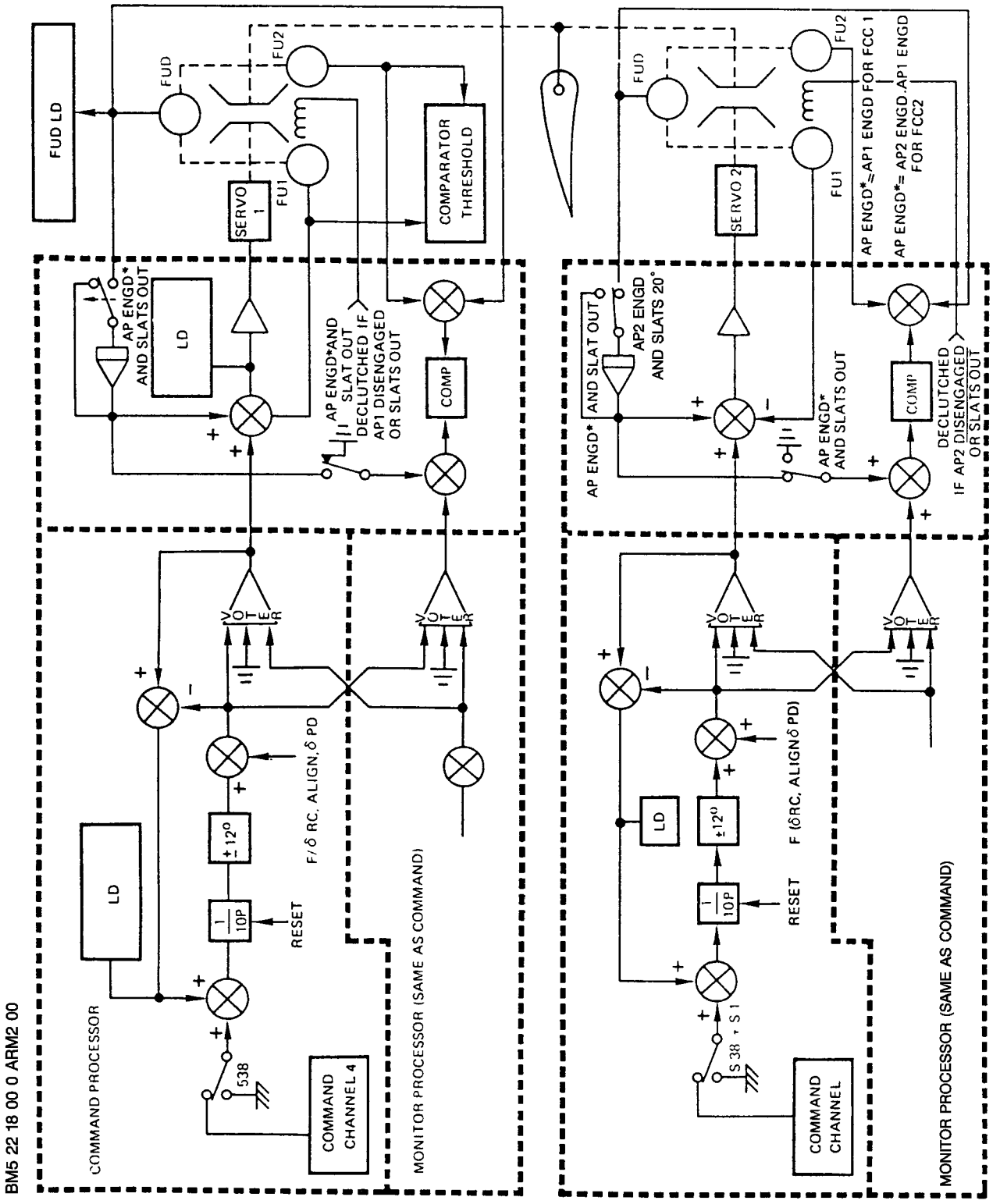
(Ref. Fig. 007)
(Ref. Fig. 008)
(Ref. Fig. 009)



Pitch Channel : Change Over Principle
Figure 007



Roll Channel : Change Over Principle
Figure 008



Yaw Channel : Change Over Principle
Figure 009

4. Architecture of the FCC from the Safety Aspect

A. Reminders Concerning AFS Computer Safety Principles (see 22-00, P. Block 1).
The AFS computer safety is dependent on :
- duplication of the computation channels
- software dissymmetry
Each computer includes two independent computation channels concerning :
- power supply circuits
- central processing units
- memories
- input/output circuits
Each computation channel has its own clock. The two clocks are not synchronized.
Both computation channels are programmed in a dissymmetrical way :
- high level language for the command channel (PLM 86)
- assembler language for the monitoring channel
Both computation channel outputs are converted into analog values : an analog vote is made between the command, monitoring computation results and a third fixed value equal to zero. The analog comparators on the voters provide a means of detecting any hardware failure and any possible software error.
Power loop monitoring is described in paragraph 3 above.

B. FCC Architecture (Ref. Fig. 001)
The FCC architecture is built in compliance with the principles described in paragraph above. Although the cruise modes are not duplicated, any failure effect is limited by limiters computed in the command and monitoring channels.
The monitoring principles of the FCC functions are :
- duplicated and monitored power supplies
- engage logic which utilizes hard-wired circuits in each command and monitoring analog channel
- power loop which utilizes an analog part (see paragraph 3). Each comparator utilized in the monitoring channel is duplicated by an equivalent comparator in the command channel :
  . Comparator C2 between the position feedback FU2 - FUD = FU1 and the command computed by the monitoring channel (in monitoring channel)
  . Level detector LD1 on slaving error (in command channel)
  . Comparator LD2 on the synchro signal FUD (in monitoring channel)
  . Comparator C1 between the synchro signals FU1 and FU2 (in command channel).

Moreover, the clutch is monitored by two microswitches

- attitude loops
  Computation of the attitude loops is made in each computation channel. In addition, the data transmitted by the three inertial reference systems are monitored (Ref. paragraph C).
- outer loop
  A unique computation is made by the command channel for the cruise modes but the command authority is limited by limiters computed in the command and monitoring channels.
  . In LAND mode (when LOC TRACK and GLIDE TRACK are selected), the outer loop computations are duplicated in all the phases
  . Computations are also duplicated for the SRS mode (TAKEOFF and GO AROUND). The protections specific to the SRS mode are described in 22-17 (protection against speed, excessive attitude, minimum climbing rate).
  . Computations of the CWS function are duplicated. Moreover, the signals from the Captain and First Officer pitch dynamometric rods are compared and limitations in load factor and in bank angle are provided.
- input circuit monitoring
  The input circuits associated with the monitored functions (takeoff, attitude loops etc.) are duplicated.
  (Ref. Fig. 002, 003)
- ARINC input monitoring
  Monitoring is achieved by performing the status matrix test (failures and no computed data NCD) as well as parity and data refresh tests. Moreover, the input circuits associated with the monitored functions are duplicated.
- FD command
  The pitch and yaw FD commands transmitted via the ARINC output, in the command channel, are fed back into the FCC input in the monitoring channel where they are compared with the commands generated by the monitoring FCC. In the event of a discrepancy detection, the ARINC transmission is interrupted (this principle is described in 22-00).
- aircraft configuration data monitoring
  Aircraft configuration data (slats, flaps, landing gears) are transmitted to the FCC through the FACs.
  The FCC compares the command and monitoring data of the flap and slat data and the agreement between the following data : nose landing gear, LH and RH main landing gear data.
  (In addition, the FAC monitors the landing gear shock absorber relay power supply).

C. Monitoring of Data Transmitted by Inertial Reference Systems (IRS)

Each FCC is connected to the three inertial reference systems :

- the command channel receives data from associated IRS and performs computations with these data
- the monitoring channel receives data from the three IRS and makes a comparison on the following data :

R . pitch rate
. pitch attitude
. roll attitude
. heading
. vertical acceleration
R . lateral speed generated in the FCC from the following data : course,
R ground speed from each IRS and runway heading (lateral speed is used in
R the LOC TRACK law) (Ref. 22-17)

and performs a vote between the commands transmitted by the three inertial reference for data concerning :

. pitch attitude
. roll attitude
. vertical acceleration
R . pitch rate

The monitoring channel performs the computations with the voted data and the heading transmitted by the associated system.

The comparison which is performed in the monitoring channel enables to isolate the faulty system :

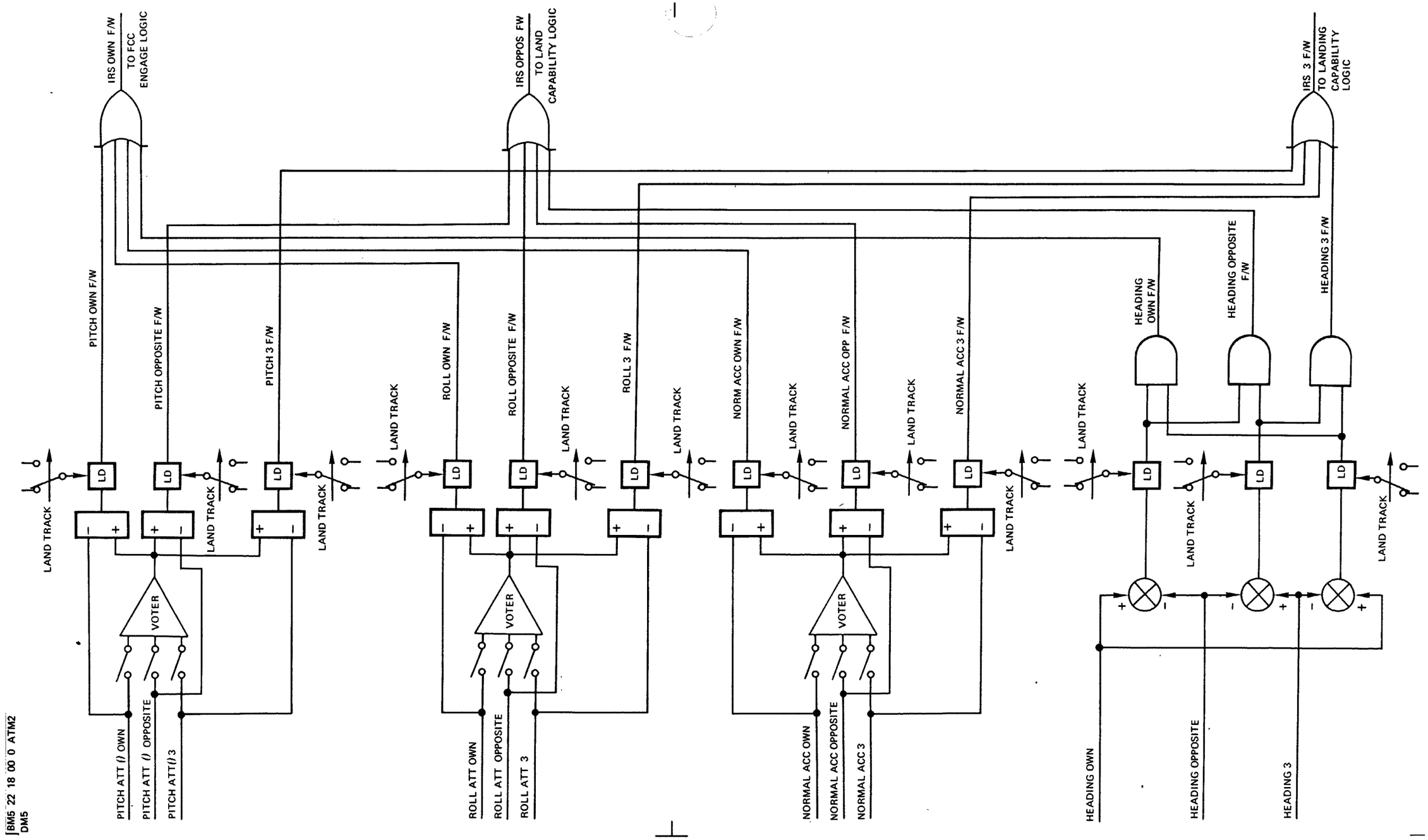
- failure of the associated system causes loss of the FCC
- failure of the opposite system causes a reduction in landing capability from CAT 3 to CAT 2 and loss of the opposite FCC (if failure is detected by the latter).
- failure of IRS 3 will have no effect on engagements but will cause a reduction in landing capabilities from CAT 3 to CAT 2.

The comparison principle is shown in following figure.

Moreover a system is detected faulty if a failure (status matrix, parity, non refresh) is detected by the ARINC input monitoring. The signals used by the FCC but not concerned by the description above (pitch rate, roll rate, etc.) are monitored in the same way as all the other ARINC inputs.

For the approach the FCC sends to the TCC a validity data signal concerning the inertial reference systems.

(Ref. Fig. 010)



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IRS Data Comparison Principle
Figure 010

The IRS validation logic to the TCC is given in figure (Ref. Fig. 011)

R IRS Data Comparison Thresholds

PARAMETER NAME	THRESHOLD	
	CRUISE	LAND TRACK
R Pitch Attitude	2°	2°
Roll Attitude	5°	1.42
Vertical acceleration	0.1 g	0.1 g
R Heading	4°	4°
R Pitch Rate	2°/s	2°/s

D. Miscellaneous Safeguards

(1)Speed monitoring

The AFS operating principle always provides a speed hold function either through the AP/FD or the autothrottle by means of the ATS mode automatic selection associated with the AP/FD mode selection.

In addition :

- in the event of a disagreement between speed and vertical speed, speed hold has priority
- a speed or Mach hold protection exists against underspeed or overspeed conditions. Moreover, approaching limit speed is indicated on the PFD and by specific warnings.

(2)Protections against FCU failures

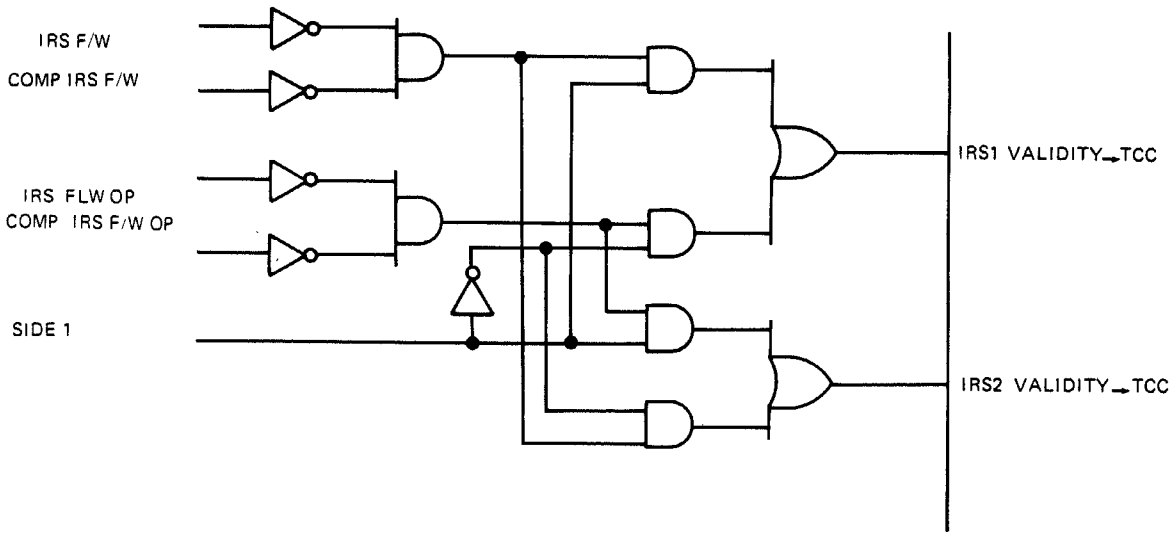
Since the FCU is built around a single processor (the second processor is not supplied with power when the first one is operating : it is used only to increase the system availability) some failures are not detected in order to avoid problems in LAND mode. The FCU is neutralized in LAND TRACK (actions on the mode selection pushbutton switches and failure data are no longer taken into account).

(3)Protections against radio altimeter failures

Although the radio altimeters are self-monitored to provide against an undetected failure on the altitude data, a comparison between both radio altimeter altitudes is made in the FWCs. This comparison will activate the AUTOLAND warning if a deviation greater than 10 ft. is detected below 200 ft. in LAND TRACK phase.

E. Integrity of Software

The safeguards associated with the software are described in 22-00, P. Block 1.



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IRS Validation for TCC
Figure 011

F. External Safeguards
The automatic flight system is provided with the following external protections :
- torque limiter levers are installed on the actuators to reduce AP authority and enable system override by operation of the flight controls by the crew.
- AP disconnect pushbutton switch installed on each control wheel
- specific warnings associated with AP and automatic landing operations described in 22-15.

G. Precautions Concerning Installation
- As far as possible the installation of the FCC wiring complies with the segregation principle of the command and monitoring channels, side 1 and side 2.
- The wirings associated with the system integrity are identified as such (colored sleeves at the end of each wire)
- The clutching lines pass through two relay stages associated with the engage lever.

H. Maintenance and Safety Tests (see 22-00, P. Block 1)
(1)The safety tests built in each computer are performed on the ground at each power rise at least once a day.
These tests provide a means of reducing the time of risk associated with the safety device operation :
- comparator, level detector (threshold and delay test)
- voter
- engage logic
- watchdog
- software integrity

As far as the FCU is concerned, the safety test is performed for FCU2 operation (its power supply will then be cut off) and then for FCU1 test.
In the event of a discrepancy detected during these safety tests, the tests are interrupted as soon as the failure is detected. If the results are negative the engagement of the system concerned is prohibited and a failure message is transmitted to the MTP. The safety tests can be initiated through manual operation (LAND test function of the MTP) and can be developped so as to increase the LAND mode availability (see 22-40).

(2)A test program is located in each computer for maintenance purposes. This program is processed in "background task": The various cases related to the fault isolation procedure initiation are described in 22-00, paragraph 3 and in 22-40.

R

1. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Blanking Caps
Referenced Procedures	
- 22-27-00, P. Block 501	Electric Pitch Trim
- 22-40-00, P. Block 501	System Monitor
- 22-10-00, P. Block 501	Autopilot

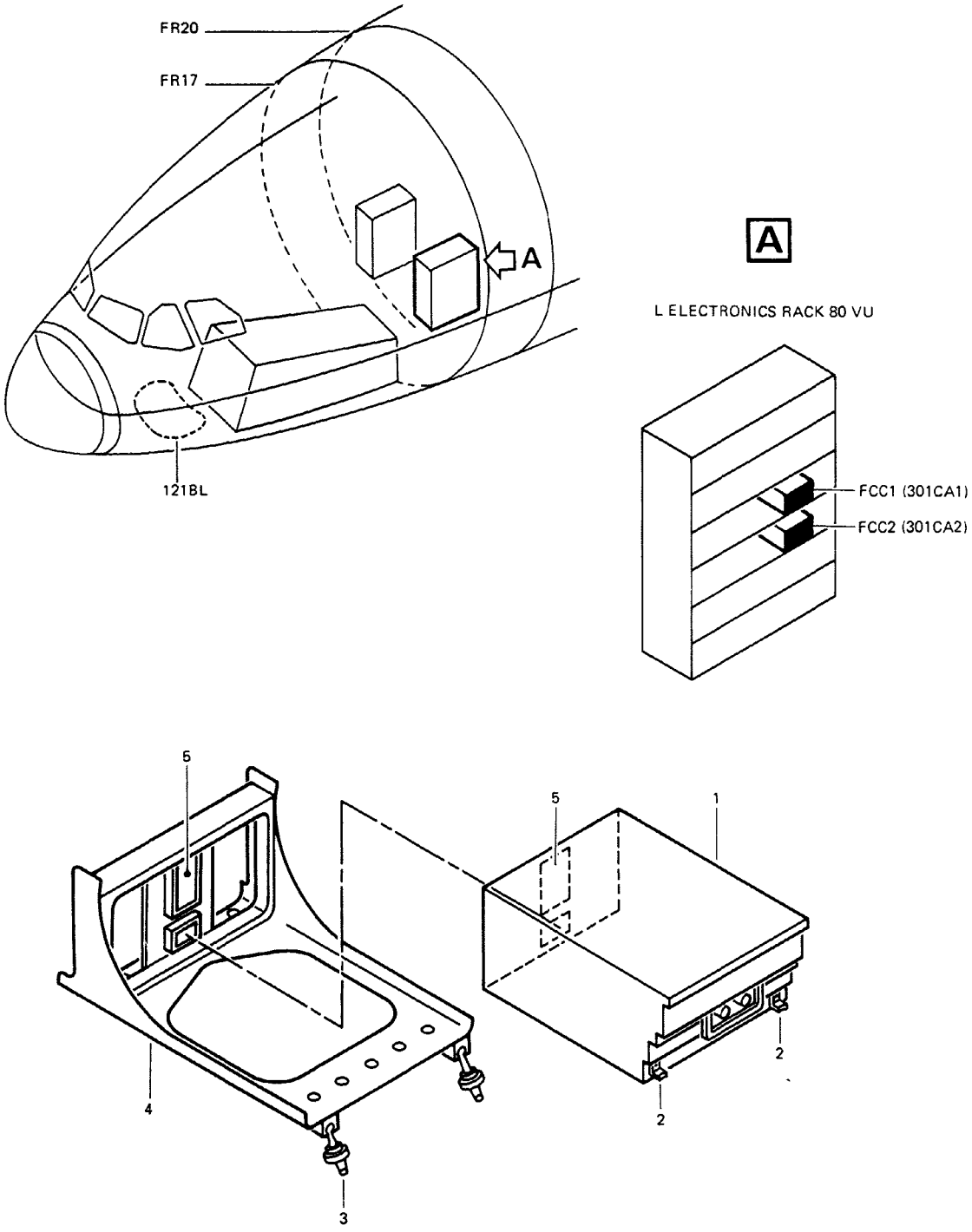
2. Procedure

A. Job Set-Up
(1)Open, safety and tag the following circuit breakers
(a)For FCC1

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/MTP	301CU	102/H 1
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
**ON A/C 101-200,			
21VU	AFS/FAC2/28VDC	306CC2	103/G 12
**ON A/C 001-100,			
21VU	AFS/TCC/SYS2/28VDC	306CJ2	102/H 12
21VU	AFS/FAC2/28VDC	306CC2	103/G 12
**ON A/C ALL			
133VU	L/G PROX DET & RELAYS/SYS1/RELAYS & RETRACT CTL	2GB	332/U51

(b)For FCC2

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/MTP	301CU	102/H 1
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC2/28VDC	306CC2	103/G12
21VU	AFS/TCC/SYS1/28VDC	306CJ1	102/H 8
133VU	L/G PROX DET & RELAYS/SYS2/FLT/GND	1GB	331/V52



Flight Control Computer
Figure 401

(2)Open access door 121BL and gain access to FCC to be removed.

B. Removal
(1)Unscrew knurled nuts (3) at front of mount (4) and tilt them downwards.
(2)Withdraw flight control computer (1) from mount (4).
(3)Cap electrical connectors (5).
(4)Remove flight control computer.

C. Installation
(1)Clean and inspect flight control computer interface and/or adjacent area.
(2)Remove blanking caps from electrical connectors (5).
(3)Check for correct condition of electrical connectors (5).
(4)Carefully slide flight control computer (1) on mount (4).
(5)Engage knurled nuts (3) in studs (2) and tighten.
(6)Remove safety clips and tags and close circuit breakers :

**ON A/C 101-200,

(a)For FCC 1 : 306CA1, 301CU, 306CC1, 306CC2 and 2GB.

**ON A/C 001-100,

(a)For FCC 1 : 306CA1, 301CU, 306CC1, 306CC2, 2GB and 306CJ2.

**ON A/C ALL

(b)For FCC 2 : 306CA2, 301CU, 306CC1, 306CC2, 1GB and 306CJ1.

D. Test
(1)Carry out AFS test (Ref. 22-40-00, P. Block 501, paragraph 1.A.).
(2)Carry out LAND test (Ref. 22-40-00, P. Block 501, paragraph 1.B.), if aircraft is operated in actual AUTO LAND CAT 3 conditions.
R (3)Carry out detailed visual inspection of the autotrim function
R (Ref. 22-27-00, P. Block 501, paragraph 1. G.).
(4)Carry out autopilot operational test (Ref. 22-10-00, P. Block 501, paragraph 1.D. Check of Pitch Trim Autotrim circuits).

E. Close-Up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Close access door 121BL.

SPEED - ATTITUDE CORRECTION - DESCRIPTION AND OPERATION

1. Introduction

The flight augmentation computer (FAC) includes all the circuits required to achieve the following functions :

A. Yaw Damping

The yaw damping function enables Dutch roll damping, turn coordination in manual flight and engine failure automatic compensation.

B. Pitch Trim

The pitch trim function provides a means of stabilizing permanent loads on the pitch axis by modifying the trimmable horizontal stabilizer angle-of-attack.

C. Flight Envelope Protection

The flight envelope protection function enables to generate V min and V max signals to the FCCs, TCCs and instruments.

D. Maneuvering Speed Computation

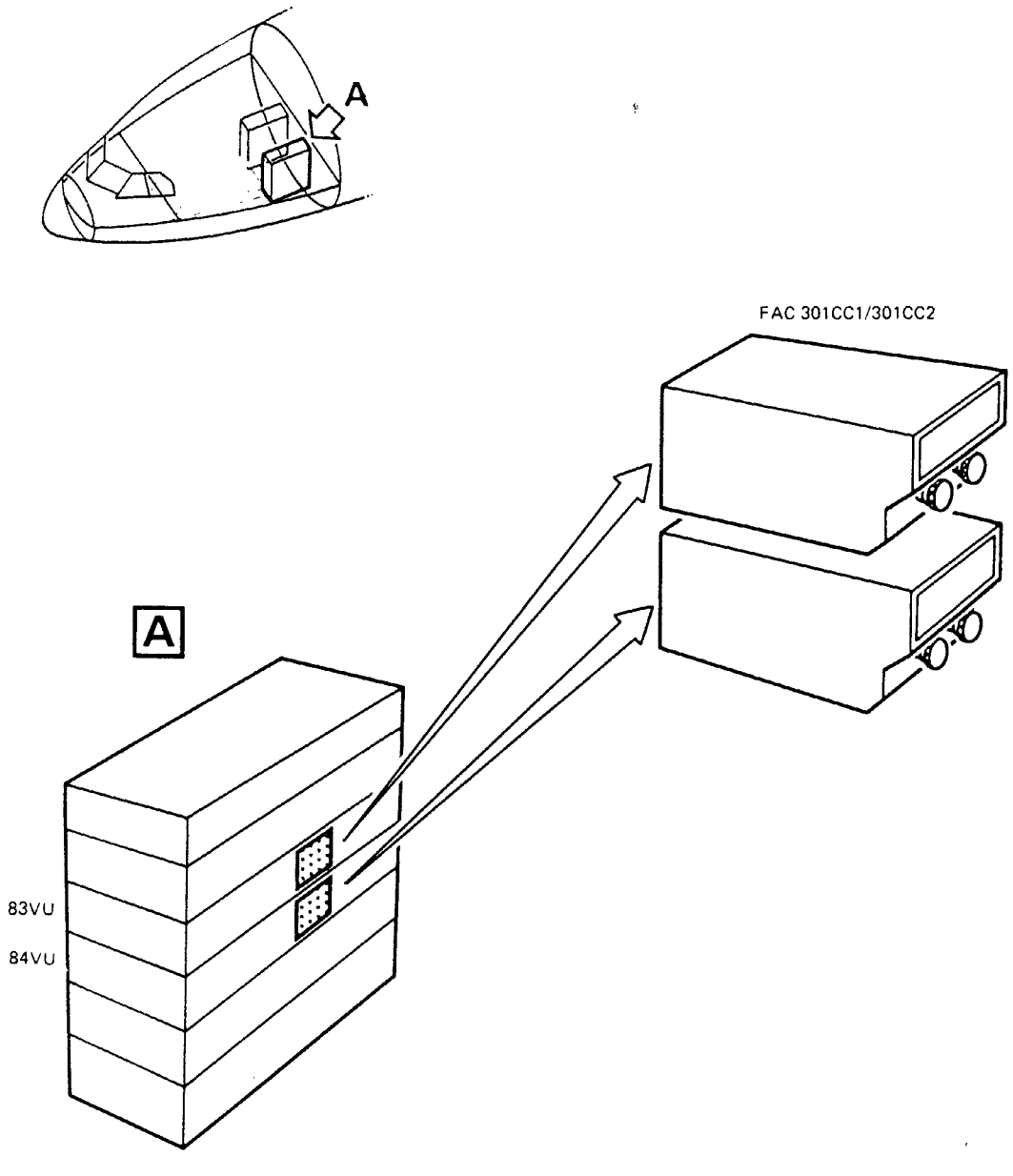
The maneuvering speed computation function enables to display speed values on the speed scale (EFIS).

E. Configuration Data Generation

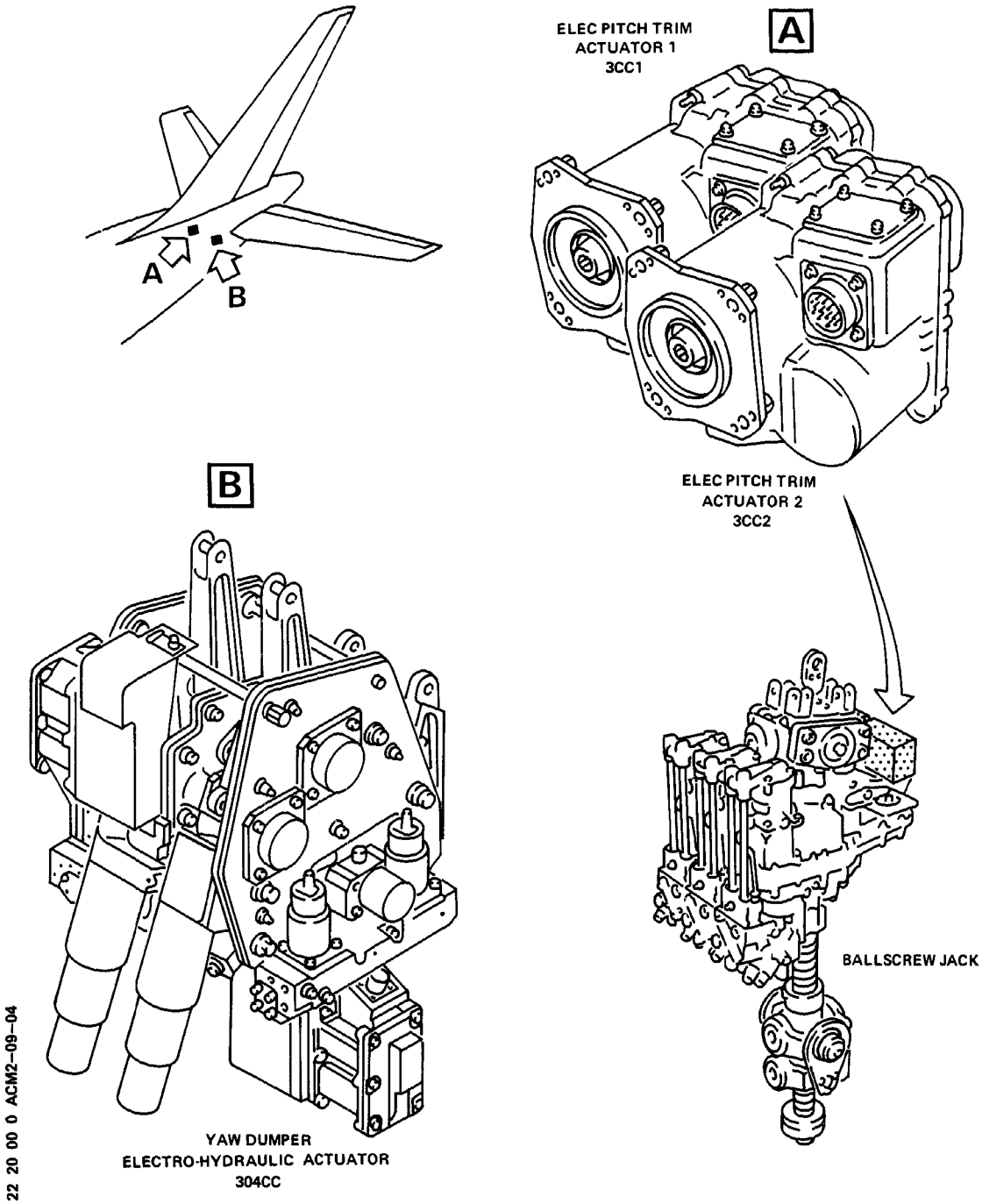
The FAC generates landing gear and flap/slat position consolidated data which are delivered to other auto flight sub-systems.

2. Component Location
(Ref. Fig. 001)
(Ref. Fig. 002)
(Ref. Fig. 003)

FIN	FUNCTIONAL DESIGNATION	PANEL	ZONE	ACCESS DOOR	ATA REF.
301CC1	FAC-1	83VU	121	121BL	22-26-34
301CC2	FAC-2	84VU	121	121BL	22-26-34
304CC	ACTUATOR-YAW DAMPER		311	312AR	22-23-22
318CC	ENGAGE UNIT-FAC/ATS	20VU	211		22-25-12
3CC1	ACTUATOR-ELEC PITCH TRIM, 1		311	312AR	22-23-29
3CC2	ACTUATOR-ELEC PITCH TRIM, 2		312	312AR	22-23-29
7CC	SWITCH-PITCH TRIM CTL, CAPT	12VU	211		22-27-12
8CC	SWITCH-PITCH TRIM CTL, F/O	14VU	212		22-27-12
27CC	STANDBY ALPHA PROBE		122	121BL	22-27-39

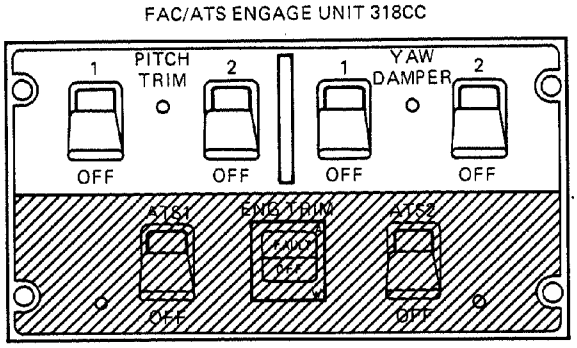


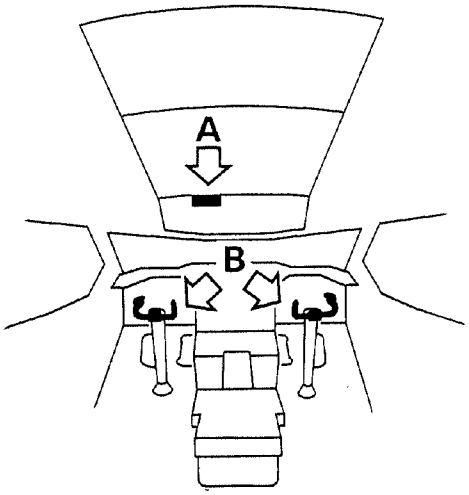
FAC - Component Location
Figure 001



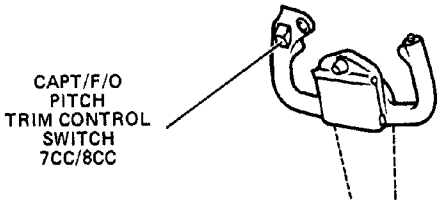
Pitch Trim and Yaw Damper Actuators - Component Location
Figure 002

A





B



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Component Location in Flight Compartment
Figure 003

3. Sub-System Functions

A. Yaw Damper

The yaw damper system provides the following functions :

- dutch roll damping,
- turn coordination in low speed manual flight to reduce the sideslip induced by the turn,
- automatic compensation to assist the autopilot in compensating the effects of the sideslip induced during the transient when an engine failure occurs. A command is generated to the rudder to counteract sideslip only during the transient induced by the engine failure.

B. Pitch Trim

The pitch trim system provides the following functions :

(1)Electric trim

This basic function provides pitch axis stabilization and enables loads applied to the control column in manual flight to be overridden by means of the pitch trim control switches located on the control wheels.

(2)Automatic trim or autotrim

This function which is activated at AP engagement in CMD or CWS without any action on the pitch trim control switch, permanently stabilizes the pitch axis and overcomes out-of-trim conditions :

- to enable AP operation around a zero control surface position
- to prevent jerks at AP disconnection.

(3)Pitch stability correction

The purpose of this function is to restore static stability in that part of the flight envelope affected by a parameter :

- Mach at high Mach number = Mach trim
- Vc at high speeds = Vc trim
- angle-of-attack throughout the flight envelope = incidence trim.

**ON A/C ALL

R

(4)THETA TRIM function

The purpose of the THETA TRIM function is to help the crew to control the pitch attitude of the aircraft during strong thrust variations in go around.

**ON A/C ALL

(5)ANTI STALL function

The purpose of this function is to assist the stall recovery at low speeds through a predetermined nose-down command.

C. Flight Envelope Protection

The FAC generates the following signals :

- Envelope protection signal :
  . stall warning speed (Vss)

. maximum selectable speed (Vmax) contingent on configuration
. minimum selectable speed (Vmin) contingent on configuration
. alpha-floor signal for protection against high angle-of-attack. This signal is used by the autothrottle for thrust correction.
R . a windshear signal for protection against sudden wind change. This signal is used by the EFIS and ECAM in order to generate the warnings.

D. Maneuvering Speed Computation
The FAC generates the following signals :
- Maneuvering speed signal
  . minimum flap and slat retraction speeds (V3, V4)
  . optimum maneuvering speed (Vman) contingent on configuration
  . airspeed tendency within 10s (Vc TREND).

All the speed information signals are displayed on the speed scale of the primary flight displays (PFD).
V min and V max information is used as minimum and maximum limits in certain modes related to the FCCs, TCCs and FMCs.

E. Configuration Data Processing
Configuration data are processed and monitored by the FAC and then transmitted to the other AFS computers via 12K data bus line.
These "reliable" signals are :
- angle-of-attack signals generated from three information streams (ADC own and opposite, standby alpha probe)
- landing gear signals generated from landing gear relays and proximity detectors
- flap and slat configuration signals generated from the SFCCs.

1. Electrical Power Supply

A. General

(1) 115 V/400 Hz power supply

FAC1 is supplied with 115 V/400 Hz through circuit breaker 309CC1.
FAC2 is supplied with 115 V/400 Hz through circuit breaker 309CC2.
This power supply is used for the electric pitch trim actuator as a reference winding on one hand (phase A and B) and to generate the control command on the control winding on the other hand (phase C).

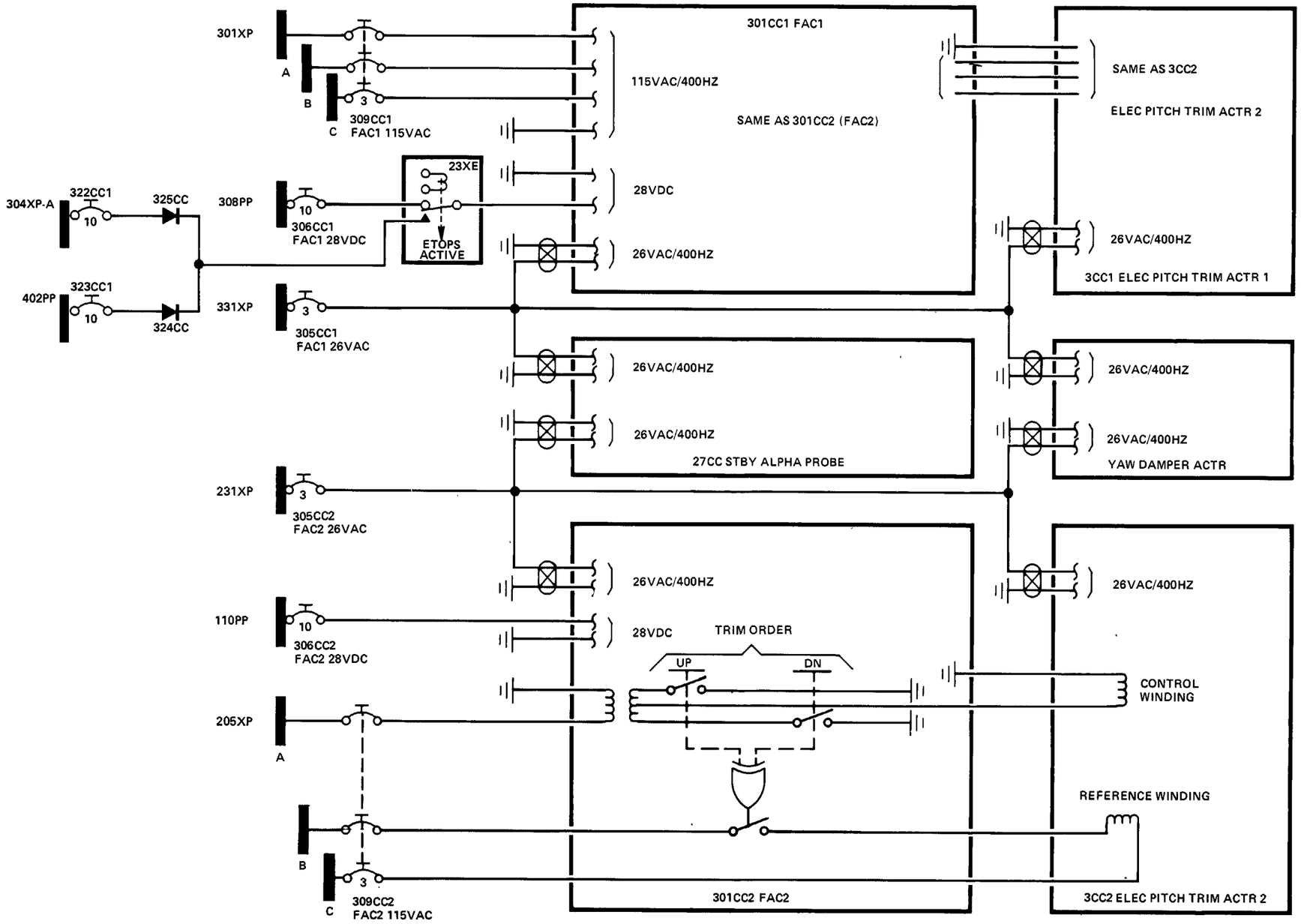
(2) 28VDC power supply

This power supply constitues the computer main supply. The system is supplied by two independent sources :
FAC1 is supplied from essential busbar through 10A calibrated circuit breaker 306CC1.
If the normal power supply is lost, FAC1 can be 28VDC supplied from the standby generator (ETOPS configuration) by the AC essential busbar via 10A calibrated circuit breaker 322CC1.
FAC2 is supplied from normal busbar through 10A calibrated circuit breaker 306CC2.
The 28VDC power supply enables to generate all the voltages associated with internal computation (+5VDC, -5VDC, +15VDC and -15VDC) as well as the logic processing voltages (discretes, relays etc.).
All these voltages derived from the 28 V power are monitored by the FAC in the command and monitoring channels.
In normal operation, normal and essential busbars are coupled. In approach, 28V power supplies are independent (uncoupling of normal and main busbars) when both APs are engaged in CMD and with LAND mode selected.
Monitoring of 28V power supply, description and monitoring of internal supply are described in 22-00.

(3) 26 V/400 Hz power supply

FAC1, electric pitch trim actuator 1 and part of the yaw damper hydraulic actuator associated with side 1 are supplied with 26 V/400 Hz through circuit breaker 305CC1.
FAC2, electric pitch trim actuator 2 and part of the yaw damper electrohydraulic actuator associated with side 2 are supplied with 26 V/400 Hz through circuit breaker 305CC2.
The 26 V/400 Hz power supply energizes the various synchros used as detectors (follow up, standby alpha probe) and is also used as a demodulation reference.
As the pitch trim system works on very low variations, its slaving requires a compensation of all the possible voltage fluctuations at the demodulation level.
The 26 V/400 Hz power supply is monitored in the FAC by a level detector set at ± 20% of value which provides, in the event of a cutoff condition, position feedback synchro data inhibition, power comparator inhibition and system loss if the 26 V cutoff exceeds 200 ms.
The standby alpha probe data are used by both FACs.
Each FAC supplies the 26 V/400 Hz power on different synchros in order to

prevent demodulation problems with different 26 V references.
(Ref. Fig. 001)



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FAC1 and FAC2 Electrical Power Supplies - Block Diagram
Figure 001

B. Internal Power Supply Automatic Cutoff

To improve the computers MTBF, their internal power supply is automatically cutoff when the aircraft is on ground. The automatic cutoff and restoration principle of the internal power supply is described in 22-00.

C. Short Cutoff Protection

(1)28 V power supply short cutoff

The FACS must remain engaged during power supply cutoffs which do not exceed 200 ms.

Precautions are taken at the level of :

- the engage lever
- the computation value which is memorized prior to cutoff.

At power supply restoration, disconnections may happen as the trim power loop comparator states may change because of the filters on the position feedback demodulators.

(2)Other precautions taken for 28 V short cutoffs on the trim

Independently of disconnection problems which may happen following 28 V short cutoffs due to the power loop, jerks may appear on the THS control. In fact the 115 V/400 Hz power remains present, the system remains engaged and at power rise the trim command can cause jerks due to the deviation between the controlled position memorized before cutoff and the true position of the actuator.

This is avoided by :

- declutching the control during 28 V short cutoff
- remaining in synchro mode at power rise
- waiting for power supply stabilization (150 ms delay) to authorize trim clutching.

Clutch opening is initiated as soon as the 28 V short cutoff is detected at the level of the hard-wired logic.

Timing at clutch closing is performed by generating an overriding signal for 300 ms.

(3)26 V/400 Hz short cutoff processing

The 26 V/400 Hz power supply is used for position feedback signal energization and demodulation.

This power supply can be subject to short cutoffs during power switching. The FAC system must remain engaged during short cutoffs of less than 200 ms.

The 26 V cutoff safeguard, which only concerns the TRIM part as the YAW DAMPER monitoring characteristics have a very large setting, is performed as follows :

- a 26 V/400 Hz level detector utilizing the hardware part enables detection of the 26 V signal drop.
  The threshold is set to 22 V approximately.
- when the detector is activated :
  . the FU demodulator values are inhibited.
  The memorized values are the latest values prior to detection
  . a hardware monostable is triggered and if cutoff exceeds 200 ms, direct action on the hard-wired logic causes the engage lever to return to OFF position.
  . a specific software device enables inhibition of servoloop comparators as long as the 26 V detector is activated.

Comparator C4 is not concerned as FU1 and FU2 evolution must be identical.
The 26 V detector logic status is delayed when taking into account inhibition during change-over from wrong to right state as the system must overcome the deviation between the position value memorized before cutoff and the true position after cutoff through demodulator filters. The delay which is introduced is the same as the 28 V cutoff processing delay, that is 1.2 -1 s.

2. Hydraulic Power Supply

A. General

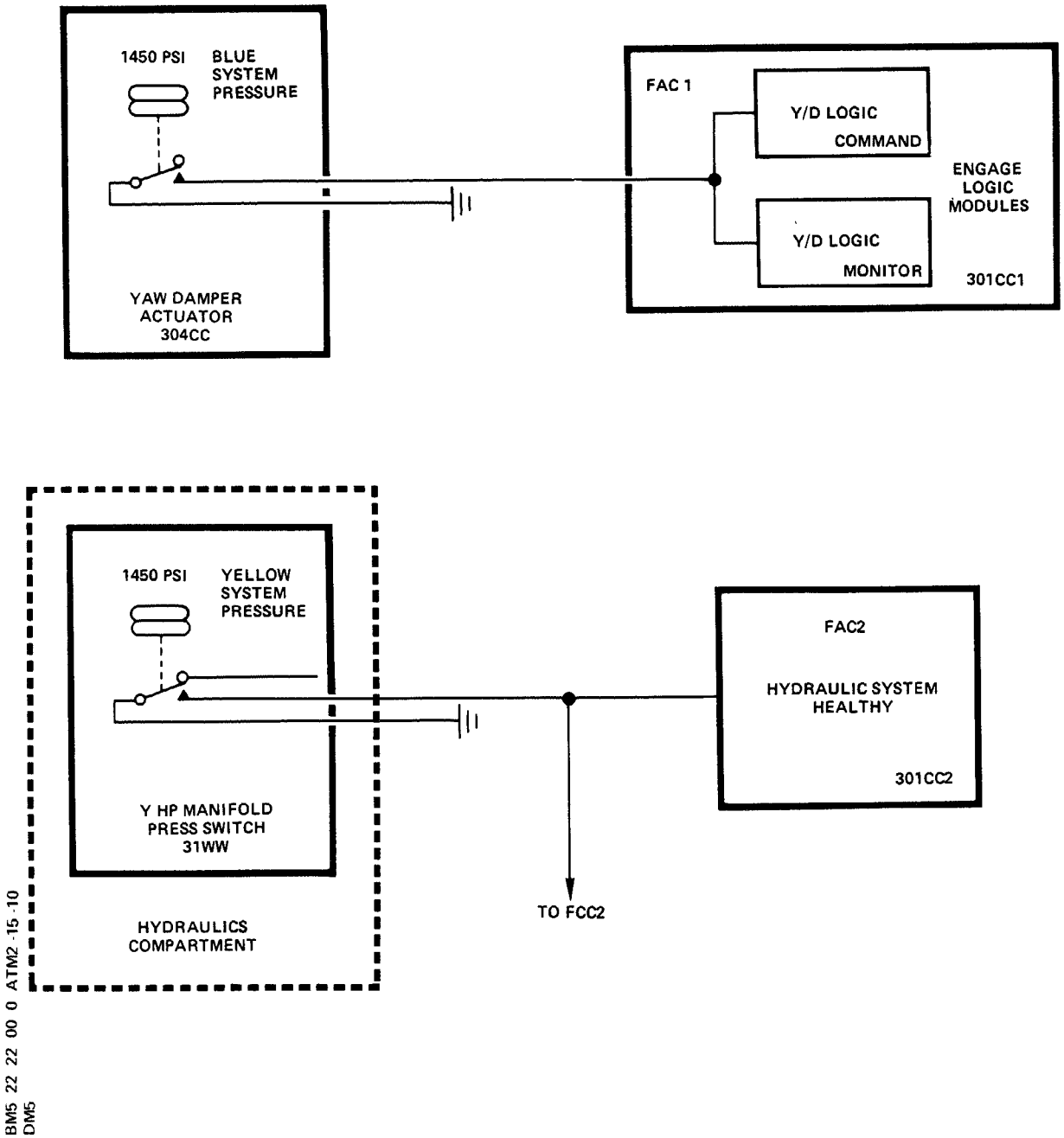
The aircraft hydraulic power supply is used :
- directly by the electro-hydraulic yaw damper actuator
- indirectly in the trim power channel at the level of the THS actuator.

B. Yaw Damper Actuator Hydraulic Power Supply

(Ref. Fig. 002)

The yaw damper actuator is powered :
- on side 1, by the Blue hydraulic system in order to be protected against any possible turbine breaking which could damage either the Green or the Yellow hydraulic system
- on side 2 by the Yellow hydraulic system.

Pressure switches set to 100 bars (1450 psi) are installed to check hydraulic system pressurization.
- on side 1 pressure switch integrated in the actuator is used to check that the driving side is depressurized and is therefore no longer connected to the rudder in order to prevent hidden failures in which the driving side would be driven by the driven side.
- on side 2 pressure switch 31WW located in the hydraulics compartment, zone 147/148, is used.



Hydraulic Pressure Logic Signals
Figure 002

1. Pitch Trim

Each pitch trim sub-system drives an electric actuator.
Both actuators are located at the top of the ball screwjack and operate the two control valves of the hydraulic motors of the THS actuator.
Each actuator consists of :

- a motor which drives the ball screwjack through a 151/1 reduction gear and a torque limiter. 2.65° synchro equals 1° THS.
  The torque limiter is so designed that it is capable of transmitting the load necessary to override the system in case of valve jamming.
- two synchros FU1 and FU2 which repeat input and output position of the torque limiter clutching unit through a 112/1 reduction gear.

The synchros are driven through a system of backlash self-adjusting gears so that hysteresis is eliminated.

- a tachometer generator to regulate the motor speed and provide slaving loop damping.

NOTE : A friction clutch enables internal jamming of the actuator to be overriden.

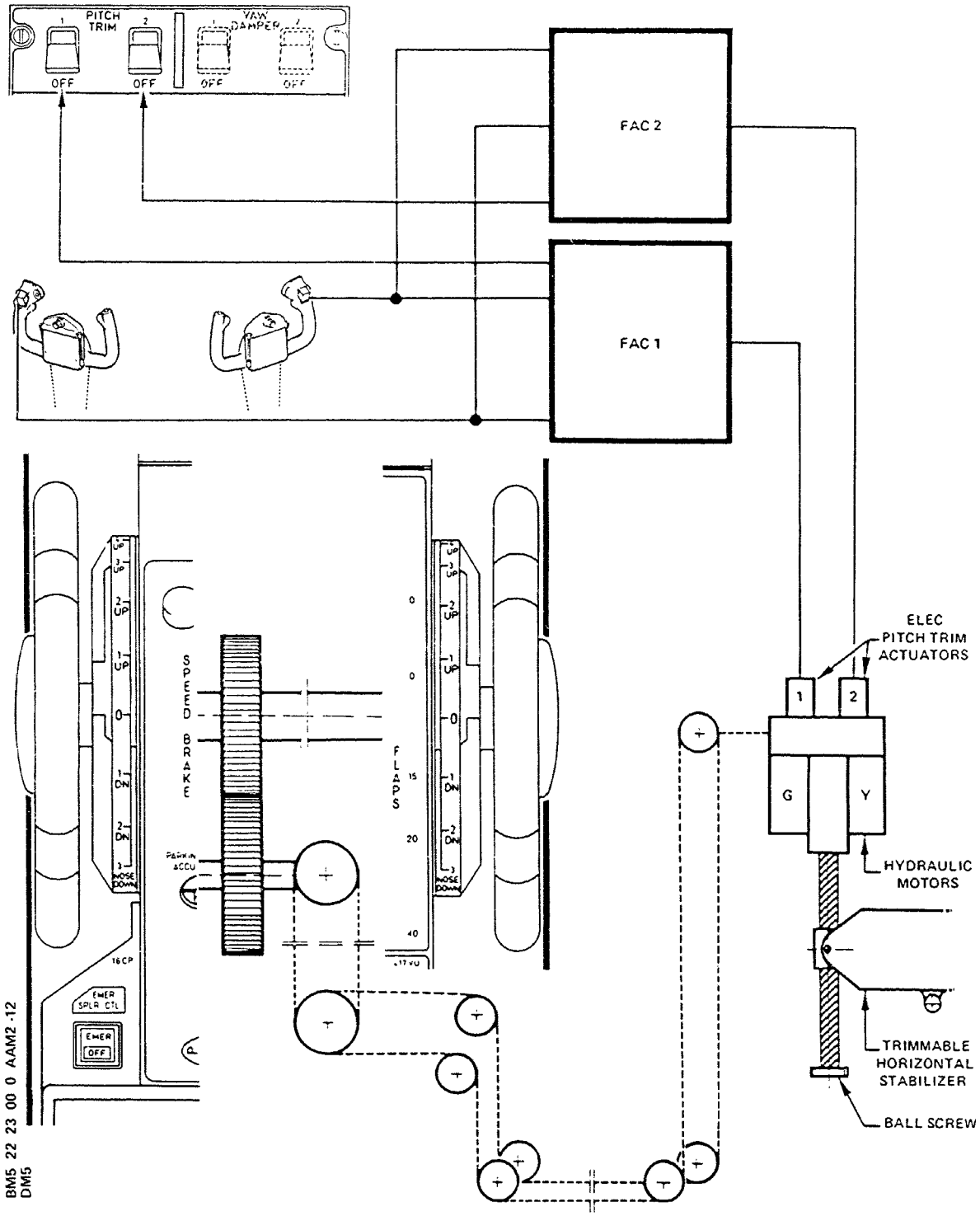
The actuator works in a dissymmetrical way :

- 5.85 turns of the motor correspond to the nose-up range, 1.65 turns corresponding to the +2° position of the stabilizer
- 8.17 turns for the nose-down range, 12.37 turns corresponding to -15° position of stabilizer.

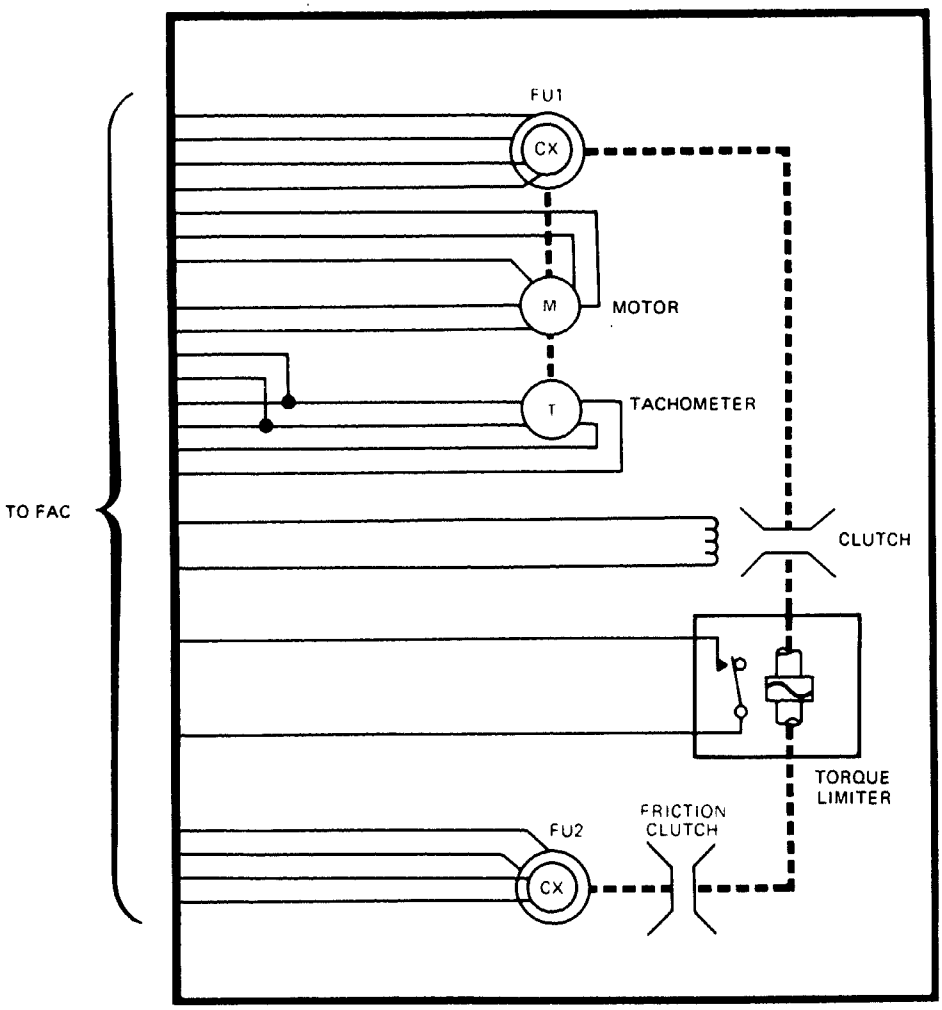
The stabilizer authority is of +3, -14.
(Ref. Fig. 001)
(Ref. Fig. 002)

A. Electric Pitch Trim Actuator Mechanical Declutch

This device serves to disconnect the mechanical trim control from the electrical trim control in the event of jamming.
In Detail A, the actuator clutch is not engaged, the pitch trim motor outputs are transmitted to yoke (3) which is free to rotate about shaft (1).
In Detail B, the actuator clutch is engaged. The movements of yoke (3) are transmitted to shaft (1) by means of roller follower (5), cam (6) through declutch slot (11) and actuating pin (10).
In 4, shaft (1) is held in position (under the load applied to the trim control wheel) which prevents the cam from moving. Under such conditions, the cam is displaced by the yoke and torque limit switch (7) trips thus causing system disengagement.
In 5, the system has not disengaged and the roller follower operated by the yoke moves into a recess in the cam which is freed and displaced upwards under spring action. This upward movement of the cam releases shaft (1).
Removal of the actuator is required to restore the actuator mechanical clutch to normal operating condition.
(Ref. Fig. 003)



Trimmable Horizontal Stabilizer Control
Figure 001



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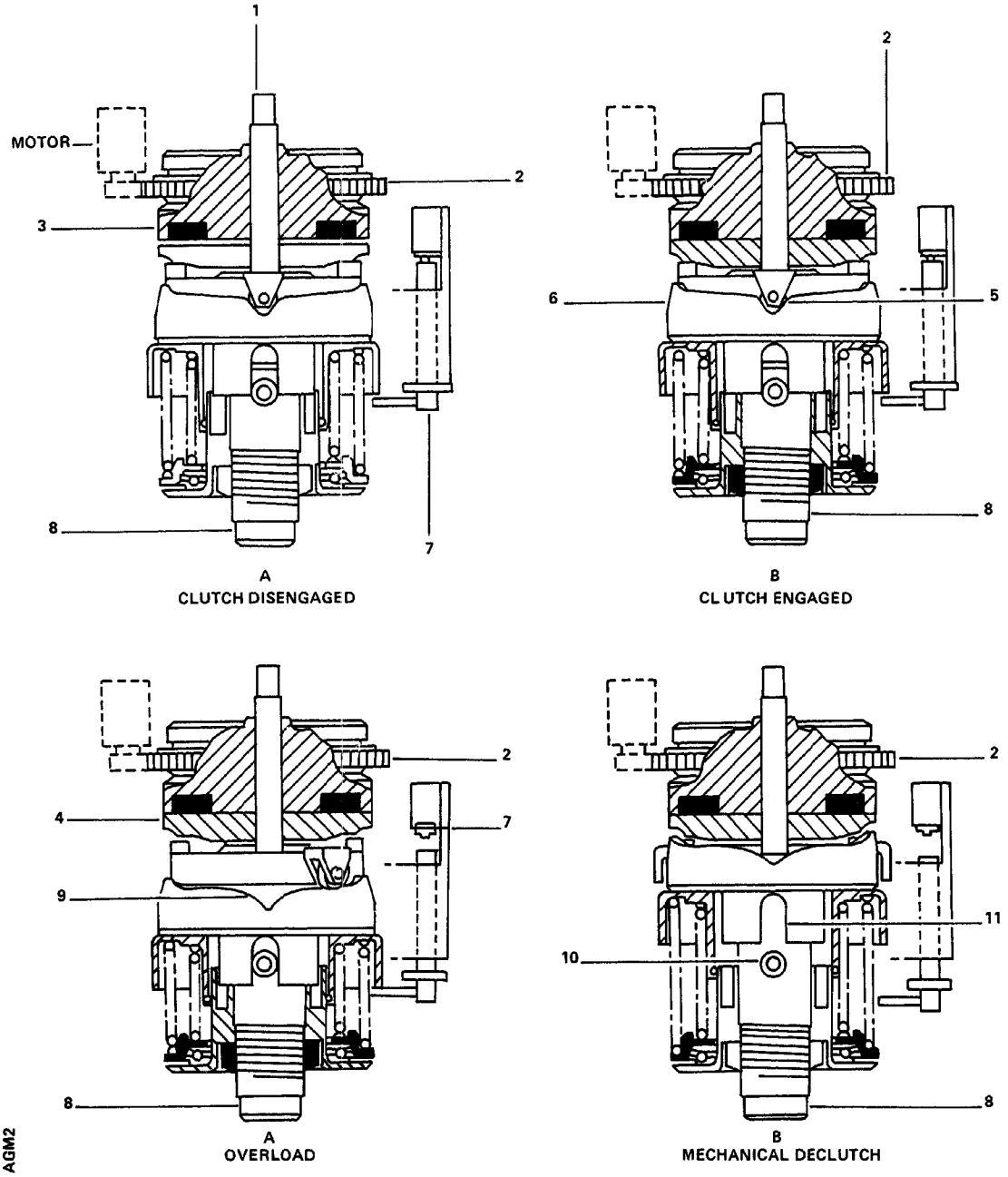
Electric Pitch Trim Actuator
Figure 002



Pitch Trim Actuator Mechanical Declutch
Figure 003

2. Yaw Damper

The yaw damper system applies mechanical inputs to the rudder control linkage via an electro-hydraulic actuator.

A. Actuator Architecture

This actuator is composed of two channels :
- the driving channel associated with yaw damper 1 (FAC1)
- the driven channel associated with yaw damper 2 (FAC2)

Each channel consists of :
- a release cylinder (1)
- a locking spring (2)
- a solenoid valve (3)
- an electro-hydraulic servovalve (4)
- a locking cam (5)

R - synchro resolvers (6).

On driving channel is :

R - a driving cylinder (7)

R - a pressure switch (8).

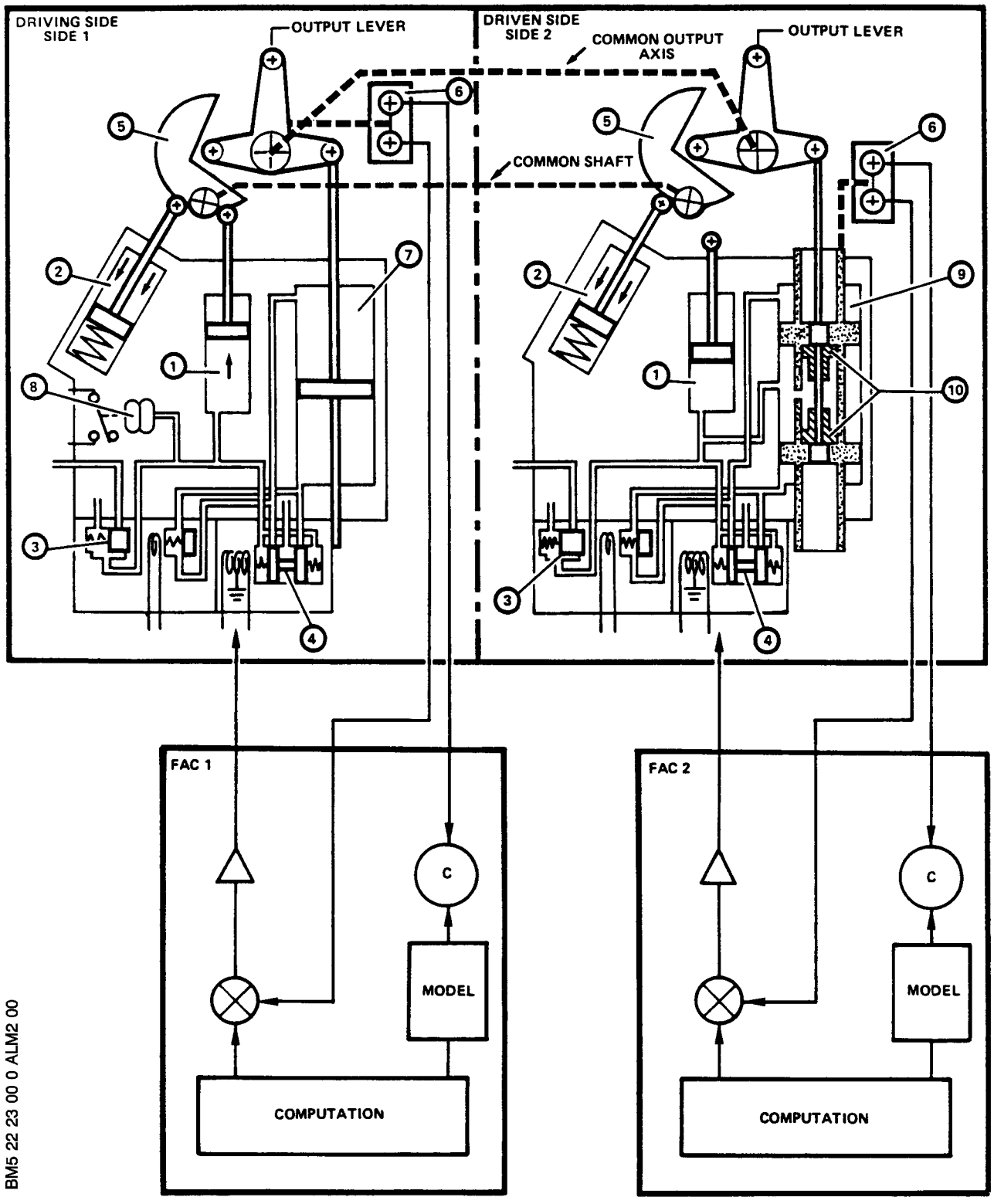
On driven channel is :

R - a driven cylinder (9)

R - a transparency device piston (10).

NOTE : The driven cylinder is fitted with a transparency system.

(Ref. Fig. 004)



Interconnection Between FACs and Yaw Damper Actuator
Figure 004

B. Actuator Principle of Operation

Both actuating cylinders receive an input signal from the associated computer (Ref. Fig. 004) and simultaneously actuate a common output axis.
The common output axis operates two output levers connected to the rudder control linkage.
The driving cylinder differs from the driven cylinder :
- the driving cylinder directly actuates the common output axis.
- the driven cylinder actuates the common output axis through a hydraulic transparency system.

The purpose of the hydraulic transparency system is to compensate for the difference in the relative positions between the two cylinders.
This hydraulic transparency system is provided by a piston concentric to the driven cylinder.
This system is designed to correspond to 9° of control surface deflection.
In the event of a detected failure of the driving side, the latter is eliminated because of its monitoring characteristics and before the driven side disagrees which could result in the loss of both sides.
Each release cylinder actuates the two locking cams.
When one yaw damper system is engaged the solenoid valve allows pressurization of the corresponding channel.
Upon pressurization :
- the locking cams are actuated by the release cylinder,
- the common output axis which is thus freed can be actuated by the driving cylinder.

The driving cylinder is operated through the electro-hydraulic servovalve which applies hydraulic pressure to the respective chambers according to control signals transmitted by the FAC.
The synchro resolvers provide position feedback in the actuator slaving. It is to be noted that the synchro resolvers associated with the driven channel indicate the position of the cylinder upstream of the hydraulic transparency.

NOTE : The control signals transmitted by the yaw damper actuator are not felt by the pilot.
They deflect the rudder without displacing the rudder pedals.

(Ref. Fig. 005).

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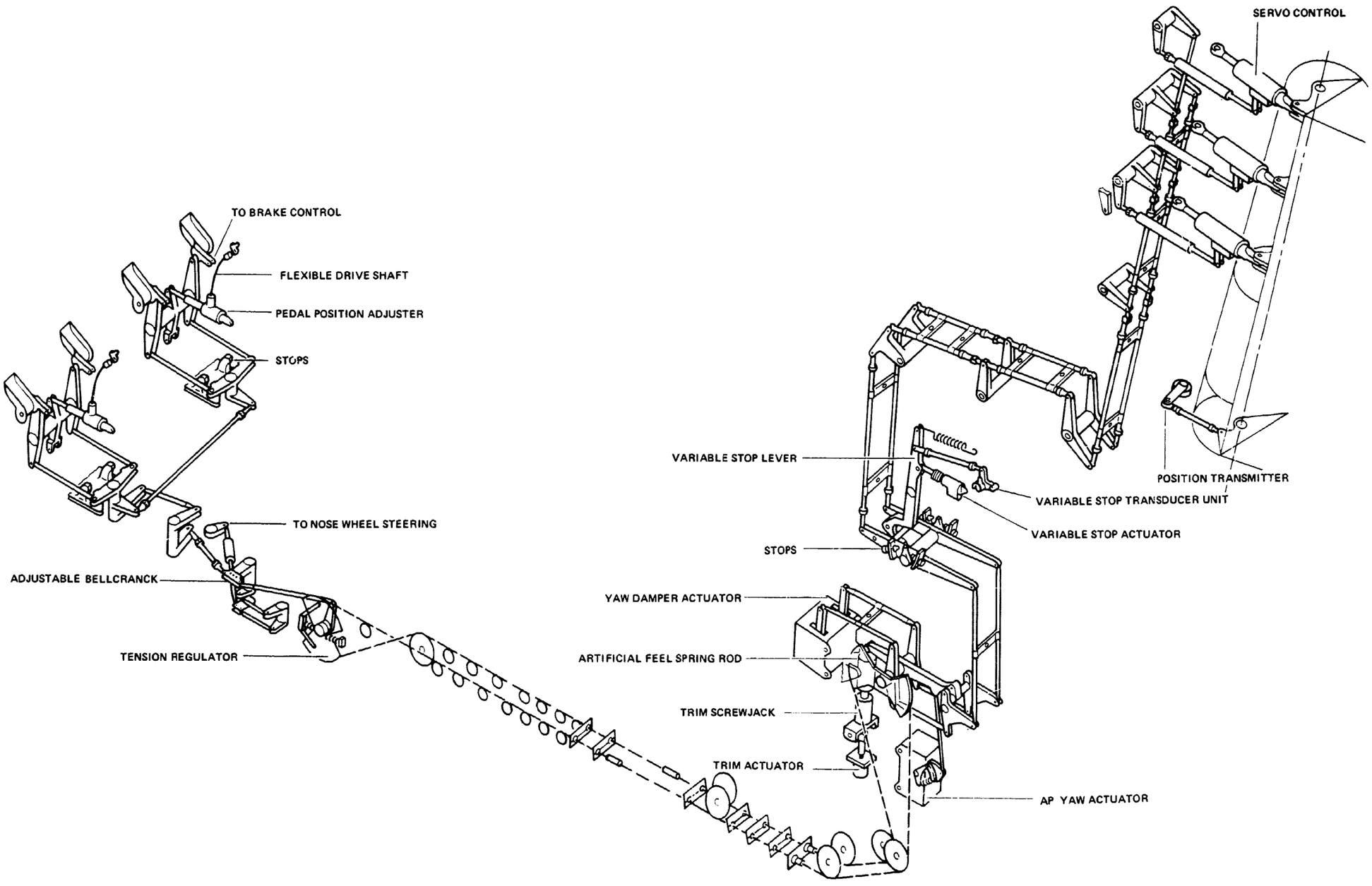


BM5 22 23 00 0 ARM2 11
DM5

Flight Control System - Layout
Figure 005

C. Actuator Performance
- the effective stroke corresponds to ±12° of the control surface deflection,
- the transparency system on the driven side allows a deviation of 40% of the total stroke, that is approximately 9° of the control surface deflection,
- the maximum speed is 42°/s, the nominal speed is 39°/s,
- the pressurization time is less than 80 ms.

Should a difference between the two systems produce a deviation exceeding 40% of the total stroke, the systems work in opposition to one another, resulting in disengagement of one or both systems under power loop monitoring action.

D. Yaw Damping Power Loop Characteristics

(1)Yaw damping output command gradient
- . 1V corresponds to 1° of rudder deflection at aircraft low speeds
- . negative if turn to the left
- . Scale factor : 0.4 mA/V or 5.6°/s/V at the level of the servovalve command with max. current = 25 mA.

(2)Position feedback gradient
- . 26V/400 Hz energization
- - FAC input gradient : 533 mV corresponds to 1° of rudder deflection at aircraft low speeds
- . direction = if phase in opposition with energization reference, turn to left.
- - Gradient after demodulation
- . 1V corresponds to 1° of rudder deflection at aircraft low speed
- . positive if turn to the left.

(3)Yaw damper actuator
- - actuator output deflection of ±16° for rudder with aircraft low speeds
- - limitation in actuator speed
  - . in current : 4mA ≅ 10V
  - . in rate : 39°/s at nominal pressure.

(4)Clutch
+28V or ground (+28V = clutched).

3. Flap and Slat Data

A. General

Flaps and slats are controlled by the SFCCs (slat/flap control computers) and the FAC receives digital data generated by these computers.
Data include :
- the command signal ordered to the control surfaces = flap and slat control lever position
- the executed command signal taken at the output of the slat electric motors and flap motors shaft.
  Therefore this signal represents the actual obtained position even in the event of control surface jamming.
  These data are encoded in the form of synchro-type signals.

B. Description

The flap and slat control system provides the FAC with flap and slat control lever position data and control surface actual position data given by the position of the output shaft of the actuator driving flaps and slats respectively.
After processing, monitoring and utilization the FAC re-transmits flap and slat control lever position data through its data bus line on discrete word 2, label 274, to the FCC, FMC, TCC, opposite FAC and EFIS.

(1)Interface

Each FAC is connected to one SFCC (slat/flap control computer) through one data bus only in the monitoring channel and in the event of own SFCC loss, an internal reconfiguration is possible. Each FAC receives a data bus line of the ARINC 429 type comprising two words made up as follows :

(a)Label 127

Bit Nos 1 to 8 : label
Bit Nos 11 to 15 : five discretes giving the status of the flap and slat control lever microswitches
Bit Nos 17 to 28 : digital data giving the synchro rotation of the slat motor output

(b)Label 137

Bit Nos 1 to 8 : label
Bit Nos 11 to 15 : five discretes giving the status of the flap and slat control lever microswitches (set of micro-switches different from label 127 microswitches)
Bit Nos 17 to 28 : digital data giving the synchro rotation of the flap motor output

Characteristics of these words are in compliance with ARINC Specification No. 429. Flap or slat synchro output is encoded as follows :
- from 0 to 179° : positive encoding
- from 180 to 359° : negative encoding

Therefore encoding is performed on ±180° with :
- a maximum slat synchro rotation of 337.188°
- a maximum flap synchro rotation of 323.673°

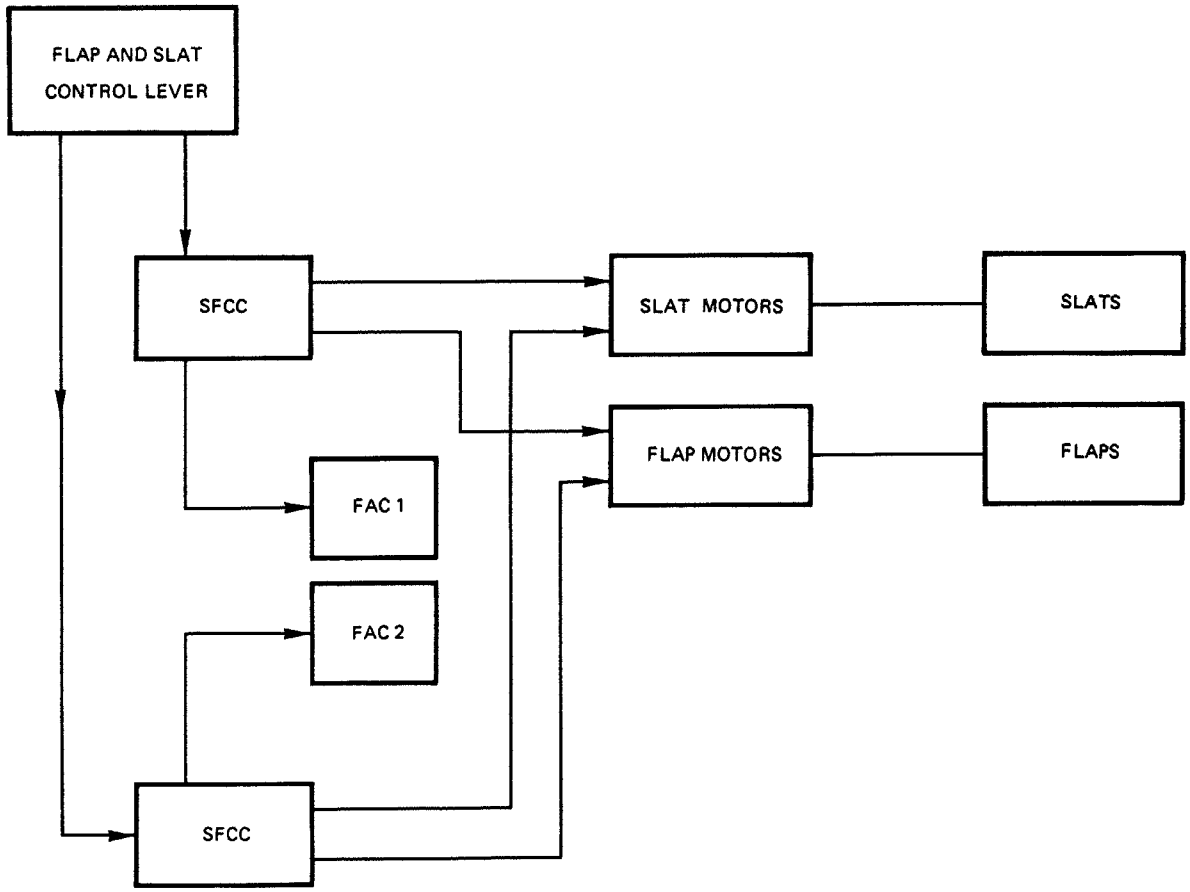
The FAC can then be provided with :
- the flap and slat control lever position data (re-transmission for AFS and internal computation)
- the actual position data of the control surfaces (computation

of maneuvering speeds).

FLAP AND SLAT CONTROL LEVER/CONTROL SURFACE DEFLECTION/SYNCHRO ROTATION/ DISCRETES 11 TO 15 - CORRESPONDING CHARACTERISTICS

BIT AT 1	FLAP/SLAT CTL LVR POS	SLAT POS (°)	FLAP POS (°)
11	1	0	0
12	2	17	8
13	3	17	8
14	4	21	16
15	5	25.4	31.5

(Ref. Fig. 006)



8M5 22 23 00 0 ATM2
DM5

FAC - SFCC Interconnection Schematic
Figure 006

C. Flap and Slat Monitoring

Two types of monitoring are performed : one is made in the SFCC and is transmitted through the status matrix of words 127 and 137 and the other is specific to the FAC.

(1)Monitoring performed in the SFCC

The status matrix content of the words transmitted by the SFCC monitors :

- slaving fault in the flap and slat control at the level of the SFCC through transmission of a flag
- slaving error affecting the power loop (motor, operation) through transmission of an NCD.
- by setting all bit Nos. 11 to 15 to zero state : a test of the flap and slat discrete content enables to check that the bit corresponding to the slat/flap lever position is at state 1.

If one fault indication appears, the associated SFCC is detected faulty and a reconfiguration can be made. However, there is no specific signal indicating a control surface jamming.

(2)Monitoring performed in the FAC

At the level of word 127 and 137 acquisition, the FAC monitors :

- the status matrix at the level of the condition analysis :
  . No refresh
  . flag
  . no computed data.

It also checks that :

- only on word 127, one and only one bit is set to state 1 on the flap and slat control lever position values
- agreement between the two control lever data signals is received on words 127 and 137 (exclusive OR gate, bit by bit comparison).
- In clean configuration, agreement between the control lever position, and the actual slats/flaps position.

In the event of associated SFCC failure detected this way, after a 10 second delay, a reconfiguration is made possible.

(3)Flap and slat data reconfiguration

Each FAC is associated with one SFCC (data bus interconnection in the monitoring channel).

The FAC transmits airspeed and configuration data for two distinct types of users.

(a)AFS (TCC and FCC)

These are mainly V min, V max and discrete position (discrete word 2, label 274) data. These data are used as minimum and maximum limit values in longitudinal or speed acquisition modes.

(b)EFIS and FMC

These are data required by the EFIS speed scale and performance optimization for the FMC together with discrete word 2 indications.

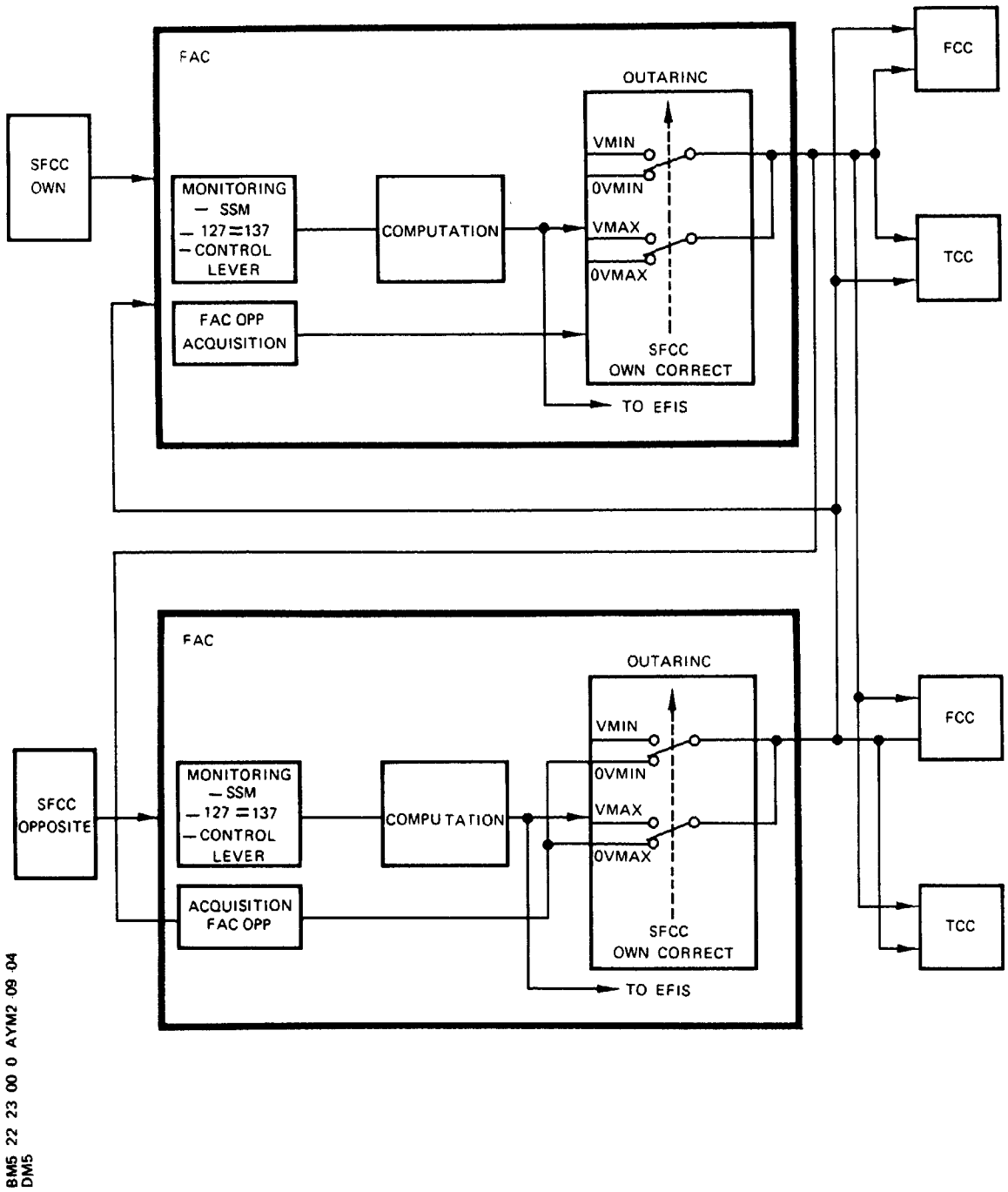
The retained principle consists, in the event of associated SFCC loss, in safeguarding the data transmitted to the AFS only. EFIS and FMC data will be lost.

However manual switching on the other FAC data will be possible in the event of ADC failure.

In the event of associated SFCC detected failure such as :

- status matrix incorrect
- dissymmetry in words 127 and 137

- discrepancy in flap and slat control lever positions.
V min and V max data associated with concerned FAC are no longer transmitted through 12K data bus. They are replaced by the data computed by the opposite FAC thanks to interconnection between FACs.
Discrete word 2 content is transferred to discrete word 2 coming from the opposite FAC, for flap and slat control lever positions only.
(Ref. Fig. 007)



Reconfiguration Principle Following SFCC Failure
Figure 007

(4)Flap and slat data processing
(a)Integrated monitoring functions
1 General
Processing is shown in the following figures (Ref. Fig. 008).
It is made as follows :
- validation monitoring of the data from the SFCC and the opposite FAC is monitored
- transmission monitoring of 12K bus by looping and of 100K bus.

2 Operation (Ref. Fig. 009)
The monitoring channel ensures acquisition of :
- words 127 and 137 : FLAPS and SLATS
- discrete word from the opposite FAC.

Associated SFCC is validated by taking into account :
- the status matrix of both words 127 and 137
- the flap and slat control lever position not locked at 1
- the identity of the control lever positions given by the two words through bit by bit comparison of bit Nos. 11 to 15.
- in clean configuration the agreement between the lever position and the surfaces position.

The result of these conditions enables to validate one bit (bit No. 17 : SFCC OWN to 1 meaning associated SFCC) which can be used by the operational capability downgrading in the FCC.
Discrete word 2 of opposite FAC is validated by taking into account :
- the status matrix of word 274
- the flap and slat control lever position not locked at 1
- check of the word bit No. 17 meaning that the opposite SFCC is correct.

The result conditions a signal which is taken into account by the pitch trim and yaw damper engage logic.
This signal is of course unconditionally forced to the right state when the associated SFCC is detected correct but in the monitoring channel only.
Several cases may occur :
- Both SFCCs are correct : in this case the transmitted word is the word delivered by the associated SFCC after transit through 100K bus and acquisition in the command channel where a new validation logic is accomplished :
  . check of the flap and slat control lever position not locked at 1 (only one bit at 1)
  . status matrix.

Discrete word 2 is then transmitted to the users through 12K data bus including bit No. 17 (SFCCOWN = 1) used to validate Cat. 3 landing operational capability.

(b)Opposite SFCC loss
No modification entails the considered FAC but the opposite FAC is reconfigured on that FAC for AFS data.

(c)Associated SFCC loss (Ref. Fig. 010)
In that case a test is performed to check if the opposite SFCC is at right state. If this is the case, several actions are carried out :
- discrete word 2 is computed with the flap and slat control lever values contained in discrete word 2 of the opposite FAC

- airspeed values delivered through the EFIS data bus are coded with SSM equal to F/W (SPEED LIMIT is displayed)
- SFCCs' status bit is no longer forced to right state and is then ready to detect any possible loss of the second SFCC.
- associated SFCC's status bit is positioned to wrong state (zero) which results in a possible landing capability downgrading.

The word is transmitted via 100K data bus after transit from monitoring to command channel. The following tests are performed in the command channel :

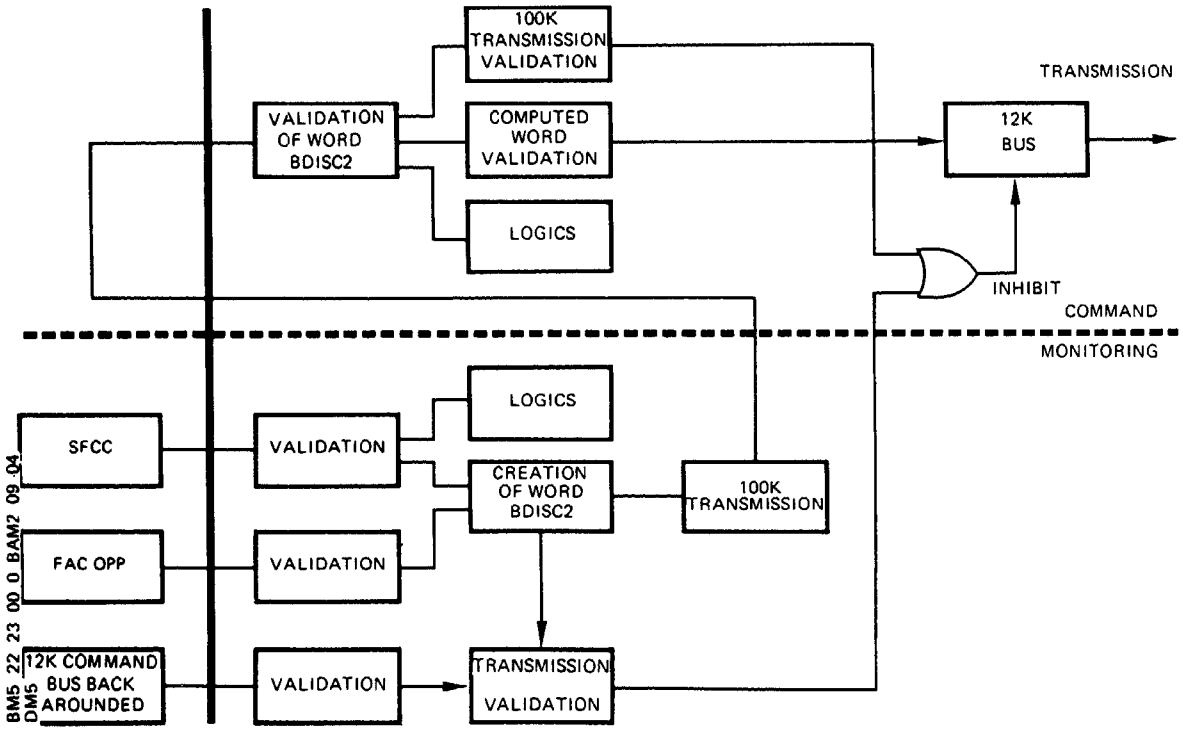
- monitoring discrete word 2 is acquired and checked
- the following airspeed values :
  . V min with associated status matrix
  . V max with associated status matrix are no longer transmitted. They are replaced by the following values delivered by the opposite FAC :

Discrete word 2 is transmitted with the validity bit (bit No. 17) set to zero in order to inform the users. A check is always performed between computed word and transmitted word.

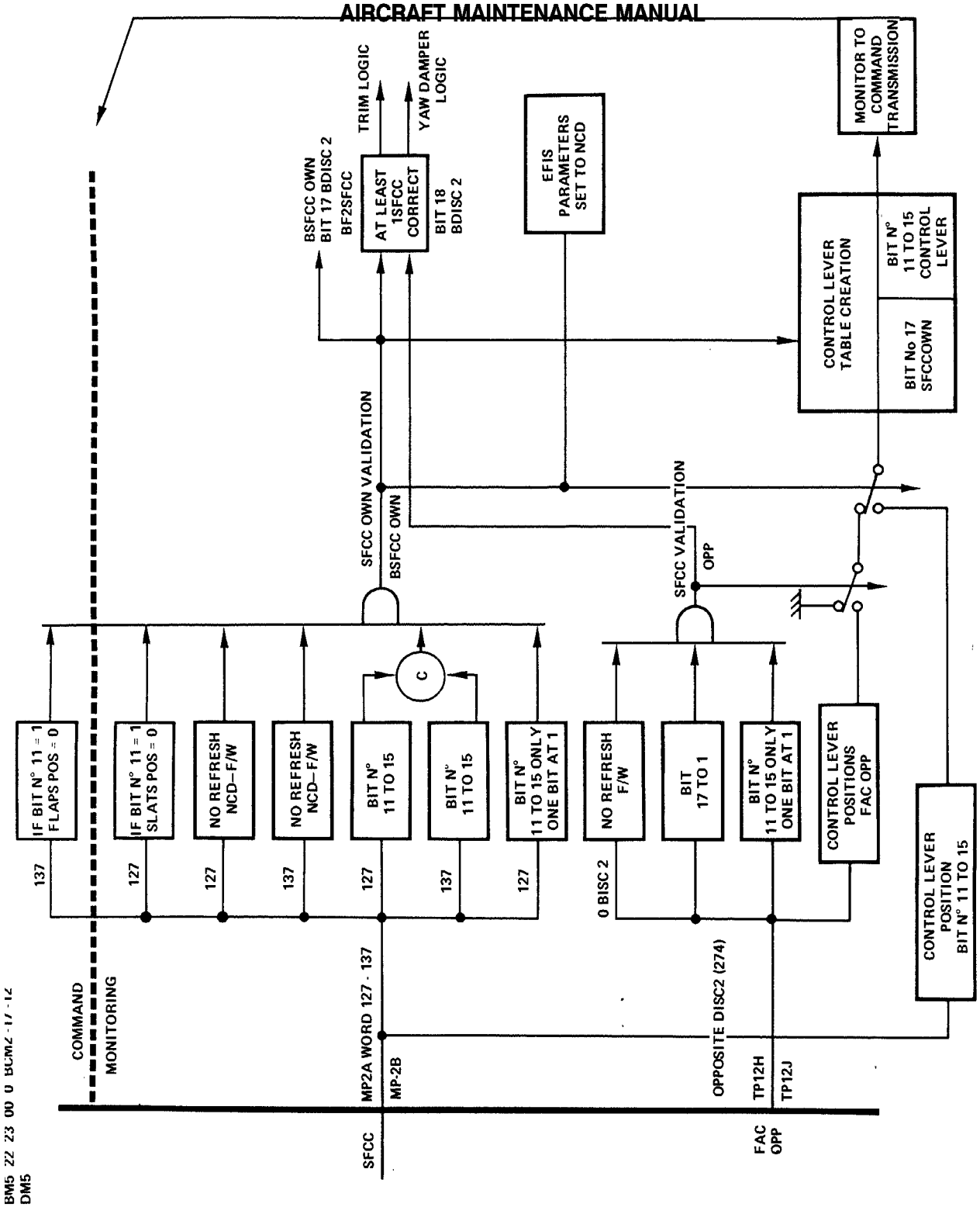
(d)SFCC total loss

In this case :

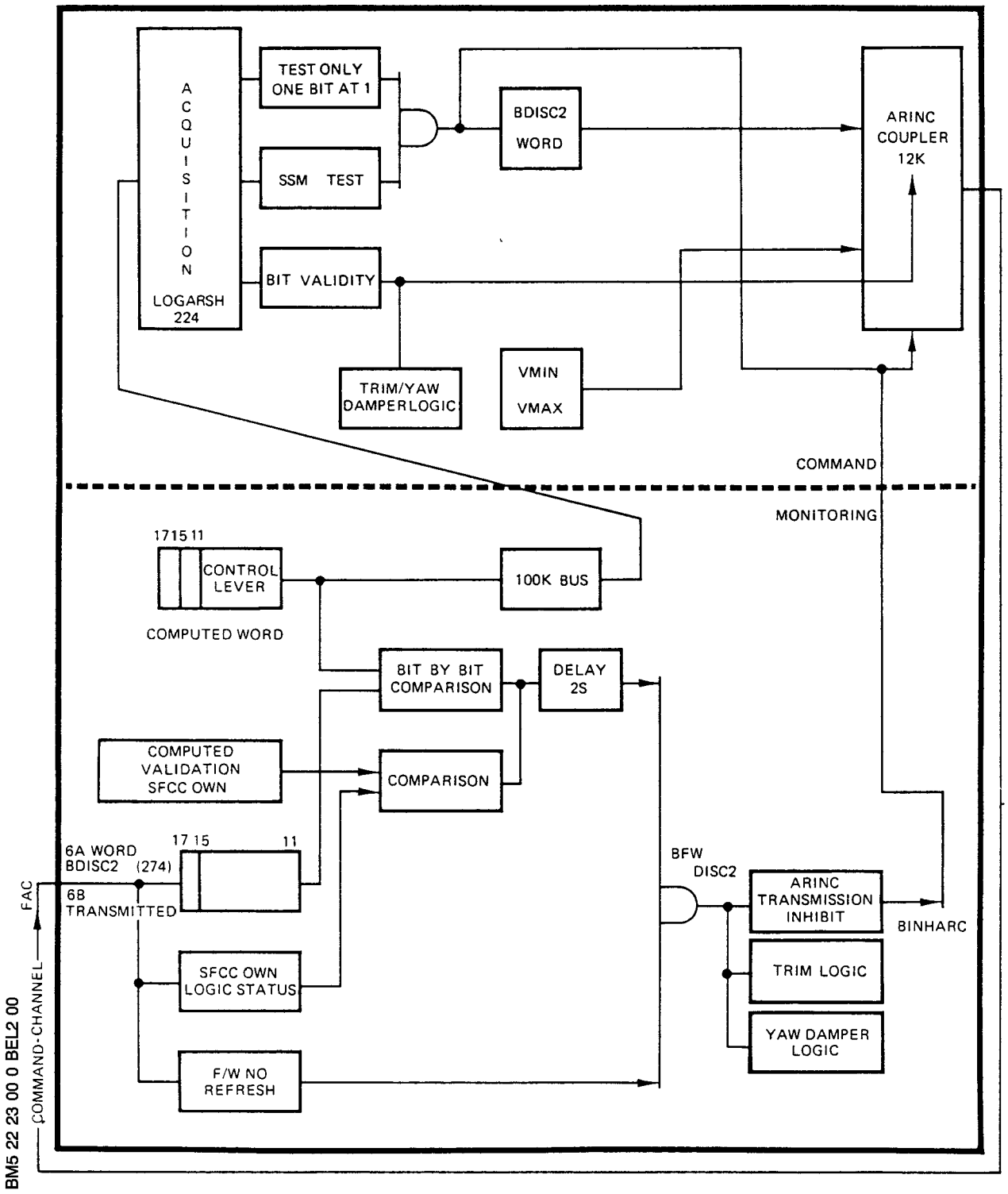
- flap and slat control lever position values are forced to the following position :
  . landing gear extended : S/F CTL LEVER in notch 5.
    Vms = 100 Kt V max = 270
  . landing gear not extended : S/F CTL LEVER in notch 1.
    Vms = 0 V max = 360.
- both SFCCs' valididy bit discrete word 2 is positioned at zero.



Monitoring General Principles
Figure 008



Flap and Slat Control Lever Positions
Figure 009

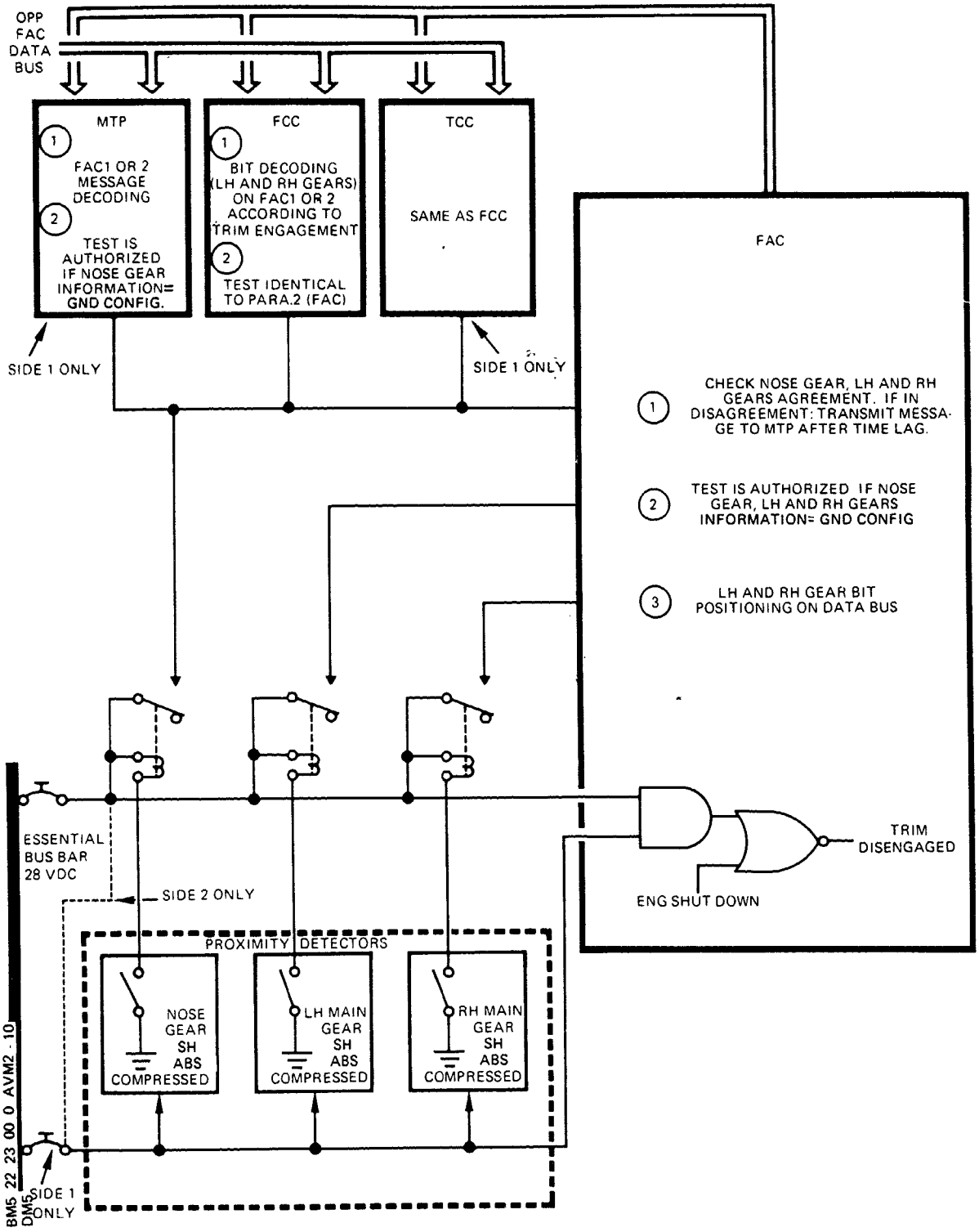


Flap and Slat Control Lever Position Table-to-User
Figure 010

4. Landing Gear Data (Ref. Fig. 011)

A. Principle
The FAC is provided with landing gear discrete data which are monitored and re-transmitted to the other AFS components through the main data bus (Ref. 22-00 Para.2.B.(3)).

B. Interconnection
- Main landing gear (shock struts) compressed :
Each FAC is connected to two relays (R and L gear), which are different for FAC1 or FAC2.
Information from these relays is transmitted by each FAC, in digital form, to the other AFS computers.
- Nose landing gear (shock struts) compressed :
Information is transmitted to the AFS from two relays : one relay to which FCC1, FAC1, TCC2 and MTP are connected in parallel; a second one which supplies FCC2, FAC2 and TCC1.
Since this information signal is used directly by the other computers, it is not re-transmitted. It is used to check for agreement between landing gear data (flight or ground configuration).
- Relay power supply :
There is a total segregation between AFS sub-systems 1 and 2 : relays associated with side 1 are fully independent from relays associated with side 2.
However, at AFS sub-system level the three associated relays are supplied through the same 28VDC circuit breaker.
The FAC therefore receives 28VDC power for validating landing gear data and checking for their presence.
NOTE : On side 1, the proximity detectors are supplied through a circuit breaker which shall never be opened on the ground. In order to facilitate AFS ground tests, relays associated with AFS1 shall be supplied through a different circuit breaker which may be opened on the ground to simulate the flight configuration.
FAC1 will then process both voltages for validating the information.


OPP FAC DATA BUS
MTP
1
FAC1 OR 2 MESSAGE DECODING
2
TEST IS AUTHORIZED IF NOSE GEAR INFORMATION= GND CONFIG.
FCC
1
BIT DECODING (LH AND RH GEARS) ON FAC1 OR 2 ACCORDING TO TRIM ENGAGEMENT
2
TEST IDENTICAL TO PARA.2 (FAC)
TCC
SAME AS FCC
FAC
SIDE 1 ONLY
SIDE 1 ONLY
1
CHECK NOSE GEAR, LH AND RH GEARS AGREEMENT. IF IN DISAGREEMENT: TRANSMIT MESSAGE TO MTP AFTER TIME LAG.
2
TEST IS AUTHORIZED IF NOSE GEAR, LH AND RH GEARS INFORMATION= GND CONFIG
3
LH AND RH GEAR BIT POSITIONING ON DATA BUS
ESSENTIAL BUS BAR 28 VDC
SIDE 2 ONLY
TRIM DISENGAGED
ENG SHUT DOWN
PROXIMITY DETECTORS
NOSE GEAR SH ABS COMPRESSED
LH MAIN GEAR SH ABS COMPRESSED
RH MAIN GEAR SH ABS COMPRESSED
BM5 22 23 00 0 AVM2 - 10
DM5
SIDE 1 ONLY

Interconnection between Shock Absorber Relays and AFS
Figure 011

C. Landing Gear Data Monitoring (Ref. Fig. 012, 013, 014)

Landing gear data which are to be re-transmitted to other users are specially monitored by the FAC.

Such monitoring deals with :

- consistency of transmitted data
- presence of relay 28 V power supply.

(Ref. Fig. 013, 014)

To this end, the following checks are made :

- conformity between left and right main landing gear status after a certain delay (15 s).
  This comparison is made in both the command and monitoring channels. The result is taken into account by the trim disconnect logic so as to enable possible reconfiguration of FCC or TCC.
- conformity between data transmitted by bus on word DISCRETE 2, label 274, bit Nos 19 (left gear) and 20 (right gear) and received data.

This comparison is made in the monitoring channel only. After 2 s and in the event of a disagreement bit No. 21 of word DISCRETE 2 concerning fault data validity is set to wrong state.

The result of the monitoring channel is transmitted to the command channel by the status of word 274 bit No. 21 via 100K line.

- ARINC transmission monitoring through the No refresh check on the status matrix of word DISCRETE 2, label 274.
  When No refresh is present, validity bit No. 21 is positioned at wrong state.
- monitoring of 28 V power supply of relays 141GB, 143GB and 144GB as well as proximity detector power supply. The engine stop signal is used to override this monitoring function in order to enable simulation of a flight configuration on the ground (landing gear circuit breakers 1GB and 2GB open).

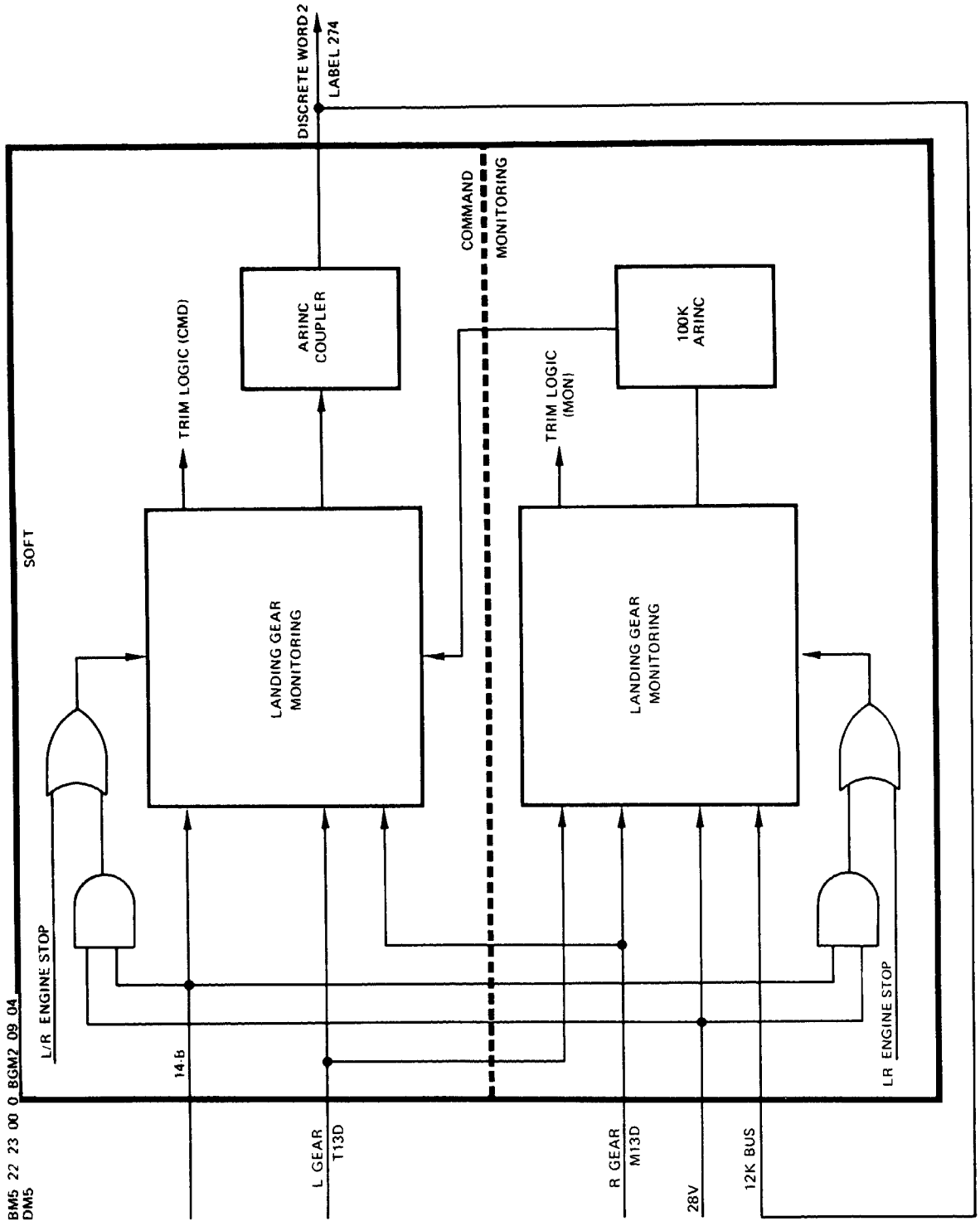
This monitoring function which is accomplished in the command and monitoring channels is used :

- in the trim disconnect logic to enable reconfiguration of users on data bus of remaining FAC
- to generate validity bit No. 21 enabling the users not to take into account transmitted data.

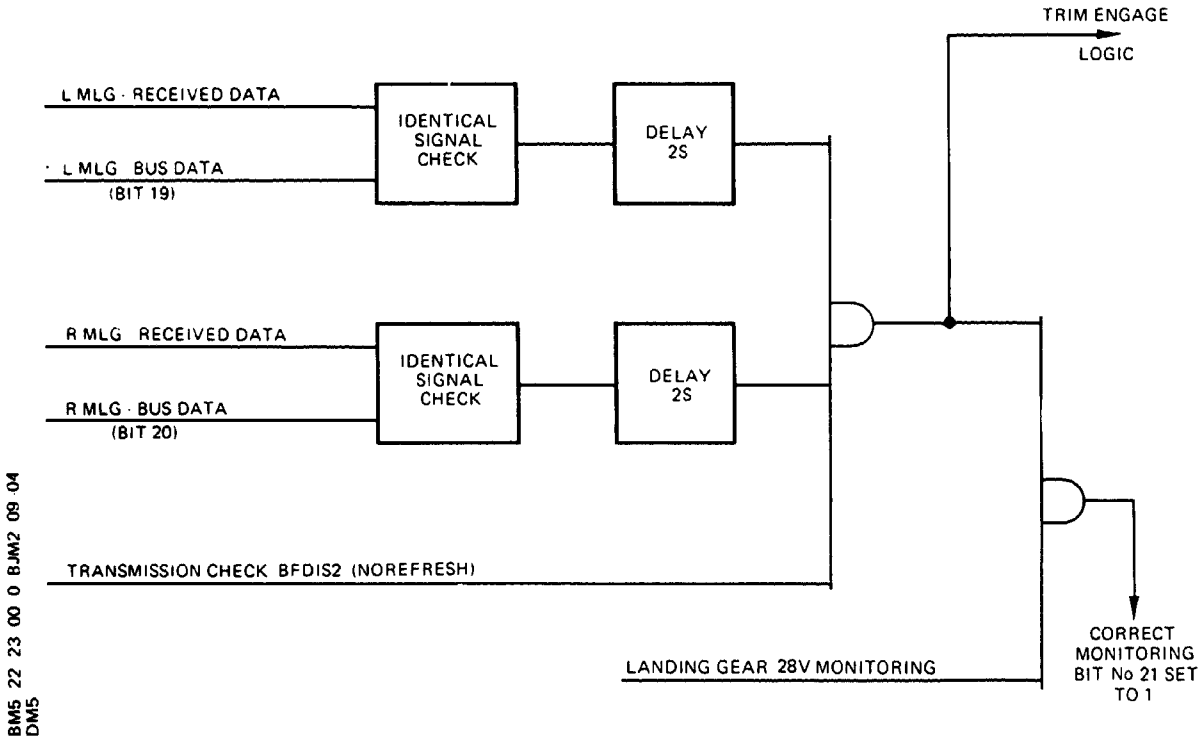
Data processing by the FCC or TCC is delayed in the system disconnect logic in order to allow any possible trim disconnection ensuring the FAC data bus reconfiguration.

A hard-wired logic signal is used for the validation of the safety tests initiated on the ground at general power rise.

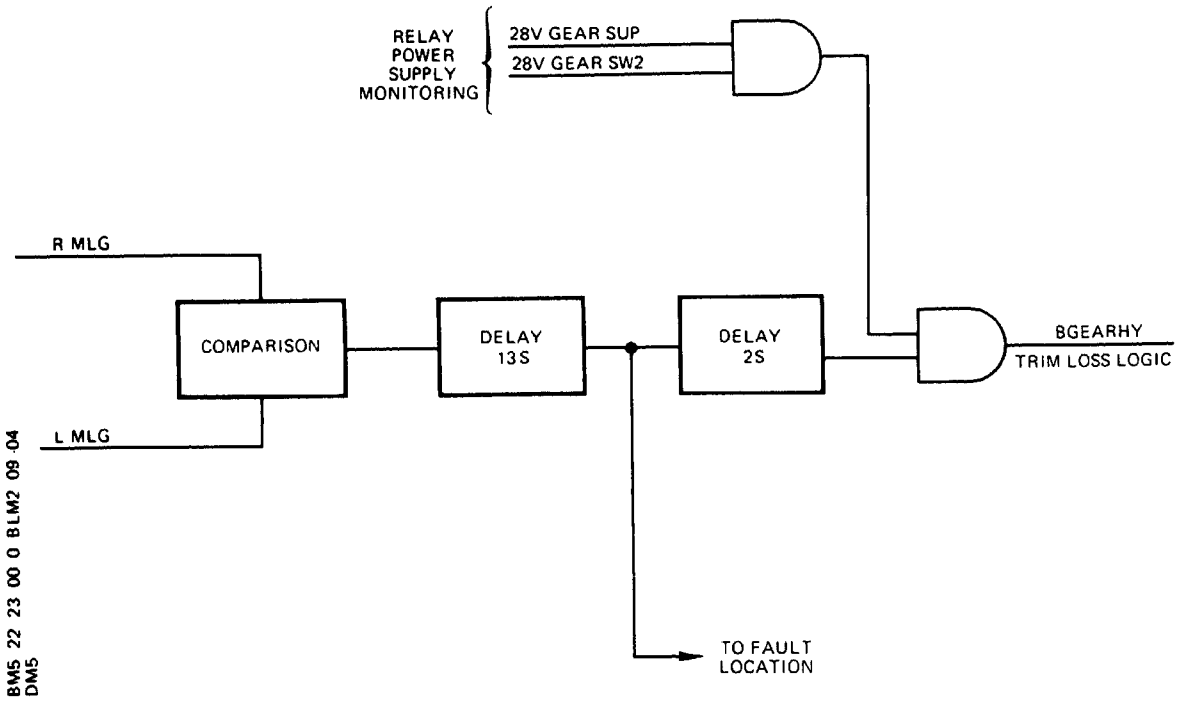
This signal performs a validity check of right, left and nose gear data. If a disagreement exists, the trim system is lost.



Landing Gear Monitoring Principle
Figure 012



Landing Gear Data Transmission Monitoring
Figure 013

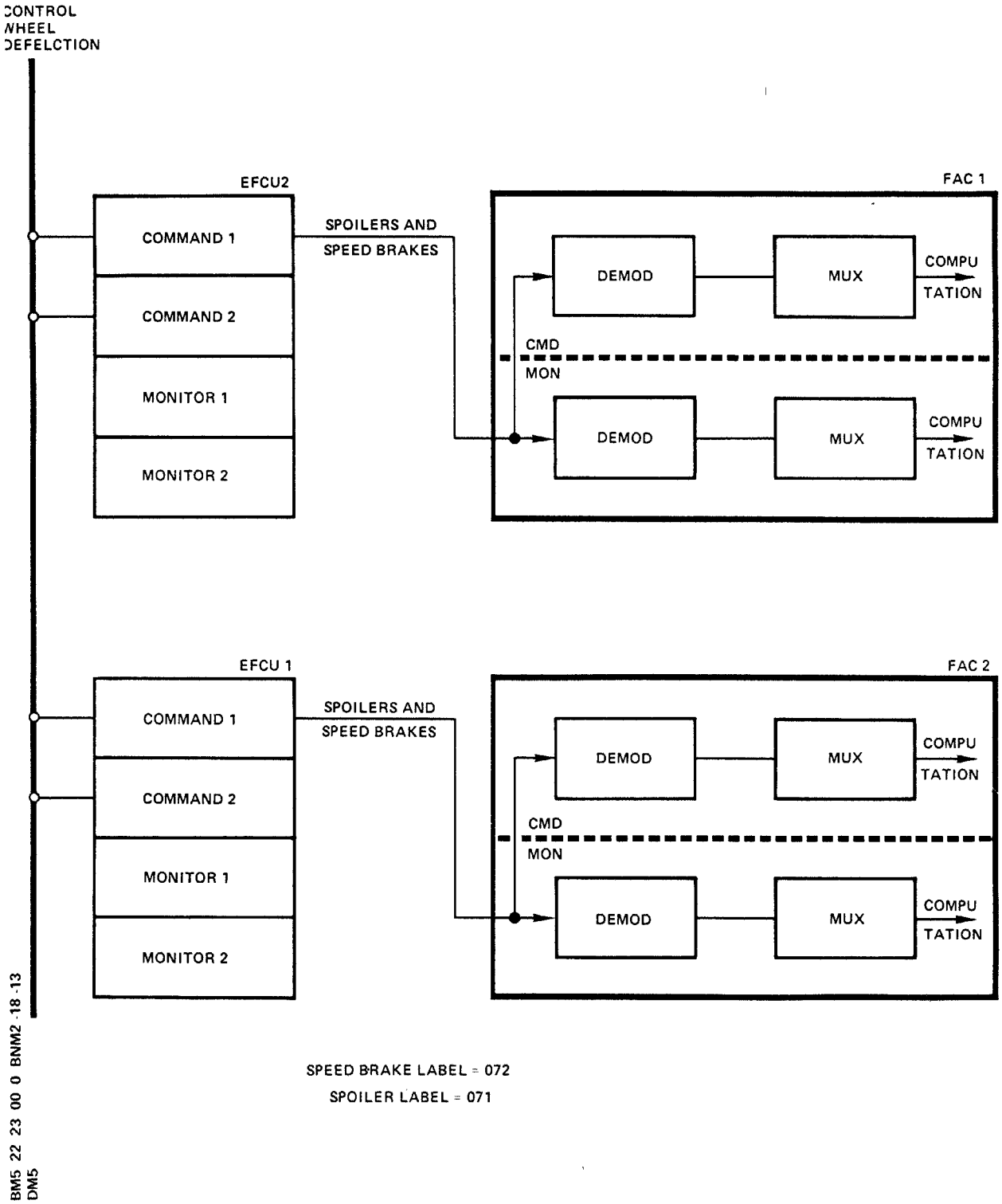


Landing Gear Dissymmetry Monitoring Principle
Figure 014

5. Spoilers/Speed Brakes and Control Wheel Deflection Data
(Ref. Fig. 015)

A. Principle
For its computations, the FAC utilizes two types of information signals generated by the EFCU (electrical flight control unit) :
- control wheel deflection data
- spoiler and speed brake position data.
To this end, each FAC is connected to an EFCU.

(1) Control wheel deflection
Control wheel deflection transducer units are used by the EFCU system which re-transmits the spoiler deflection data in ARINC 429 format under the label 071.
Internal monitoring is performed and transmitted on the status matrix of the considered word.
- label 071
- data field : 189.88° on 11 bits (bit 17 to 28). This corresponds to 128° control wheel.



EFCU/FAC Interconnection Schematic
Figure 015

YAW DAMPER ACTUATOR (304CC) - REMOVAL/INSTALLATION

WARNING : BEFORE PROCEEDING WITH MAINTENANCE WORK ON OR NEAR MECHANICAL FLIGHT CONTROLS OR PRIMARY FLIGHT CONTROL SURFACES, LANDING GEARS, ASSOCIATED DOORS OR ANY MOVING COMPONENT, MAKE CERTAIN THAT GROUND SAFETIES AND/OR WARNING NOTICES ARE IN CORRECT POSITION TO PREVENT INADVERTENT OPERATION OF CONTROLS.

CAUTION : SERVO-ACTUATOR WEIGHS APPROXIMATELY 25 KG (55 LB) ; IN ORDER TO PREVENT INJURIES TO WORKMEN OR DAMAGE TO THE SERVO-ACTUATOR, A SECOND MAN IS REQUIRED.

1. Equipment and Materials

ITEM	DESIGNATION
A.	Access Plaftorm 6 m. (19 ft. 8 in.)
B.	Circuit Breaker Safety Clips
C.	Lockwire - Corrosion Resistant Steel, Dia. 0.7 mm (0.027 in.)
D.	Blanking Caps
E.	Warning Notices and Safety Barriers
F. 98A27307546000	Rigging Pin - Main Bellcrank, Rudder Mechanical Controls
G.	Torque Wrench
H. 98A27307545000	Rigging Pin - Cable Quadrant, Rudder Mechanical Controls
I. 98A27207601000	Rigging Pin - Variable Stop Unit, Rudder Mechanical Controls
Referenced Procedures	
- 22-26-00, P. Block 501	Yaw Damper
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization
- 29-14-00, P. Block 301	Hydraulic Reservoir Pressurizing System
- 29-22-00, P. Block 301	Blue Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Lower Transfer Unit)

2. Procedure

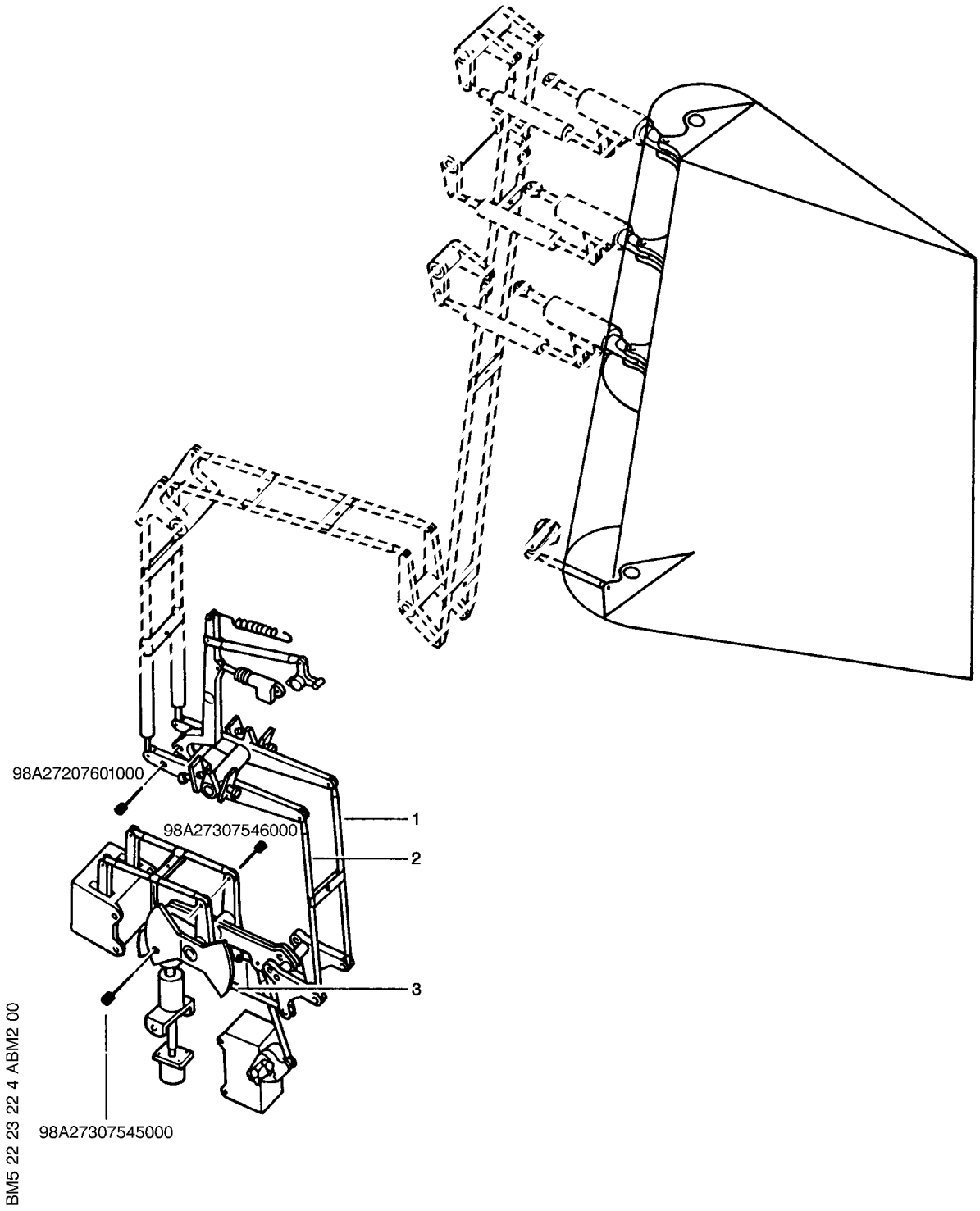
A. Job set-up

(1)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC1/26VAC	305CC1	103/G 7
21VU	AFS/FAC2/26VAC	305CC2	103/G 8
21VU	AFS/FAC2/28VDC	306CC2	103/G12

(2)Place warning notices and/or safety barriers.

(3)Position access platform at access door 312AR and open it.
(Ref. Fig. 401)
(4)Installation of the rigging pins :
CAUTION : DO NOT DISTURB PRE-ADJUSTED LENGTHS OF PUSHRODS (3).
(a)Immobilize cable quadrant using rigging pin 98A27307545000.
(b)Immobilize variable stop lever using rigging pin 98A27207601000.
Rigging pin must be inserted and withdrawn easily. If not, adjust length of rods (1) and (2) as follows :
- Remove clamp assemblies interconnecting the two rods.
- Cut lockwire and loosen nuts. Disengage washers and adjust length of rods.
- Engage washers, fully tighten nuts and safety with lockwire.
- Install clamp assemblies.
(c)Install rigging pin No. 98A27307546000.
(5)Depressurize Yellow and Blue hydraulic systems (Ref. 29-10-00, P. Block 301 and 29-14-00, P. Block 301).



Yaw Damper Servo-Actuator - Rigging Pins
Figure 401

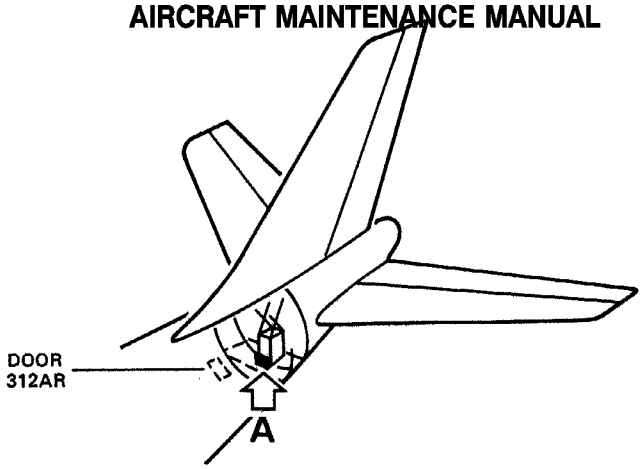
R

3. Removal/Installation of Pressure Switch on Servo-Actuator
(Ref. Fig. 402)

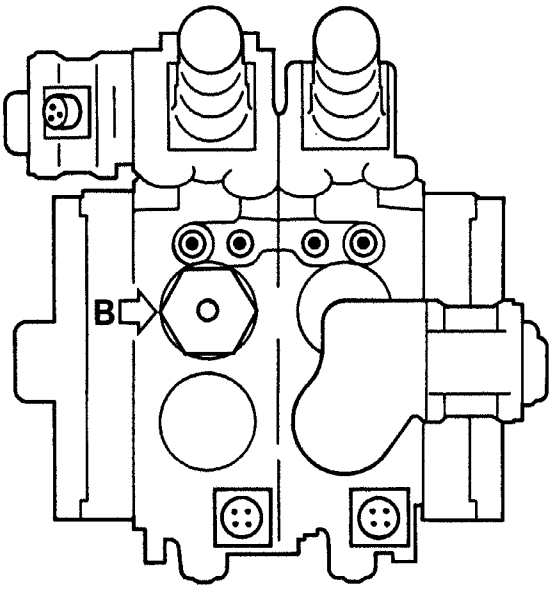
A. Removal of Pressure Switch
(1)Disconnect electrical connector (2) and install blanking cap.
(2)Cut and remove lockwire and unscrew pressure switch (1).
(3)Remove pressure switch (1) and install blanking cap on hydraulic connection.

B. Preparation of Replacement Component
(1)Make certain that component is clean and free from dents and corrosion and that electrical connector is in correct condition.
(2)Make certain that component is conditioned for use on the aircraft (hydraulic fluid).
(3)Clean and inspect component interface and adjacent area.

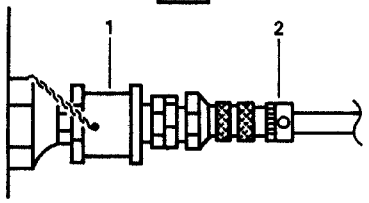
C. Installation of Pressure Switch
(1)Remove blanking caps.
(2)Position pressure switch (1) in recess and TORQUE to 6 m.daN ± 0.3 (525 lbf.in ± 25).
(3)Wirelock with corrosion resistant steel lockwire, dia 0.7 mm (0.027 in.).
(4)Connect electrical connector (2).



A



B



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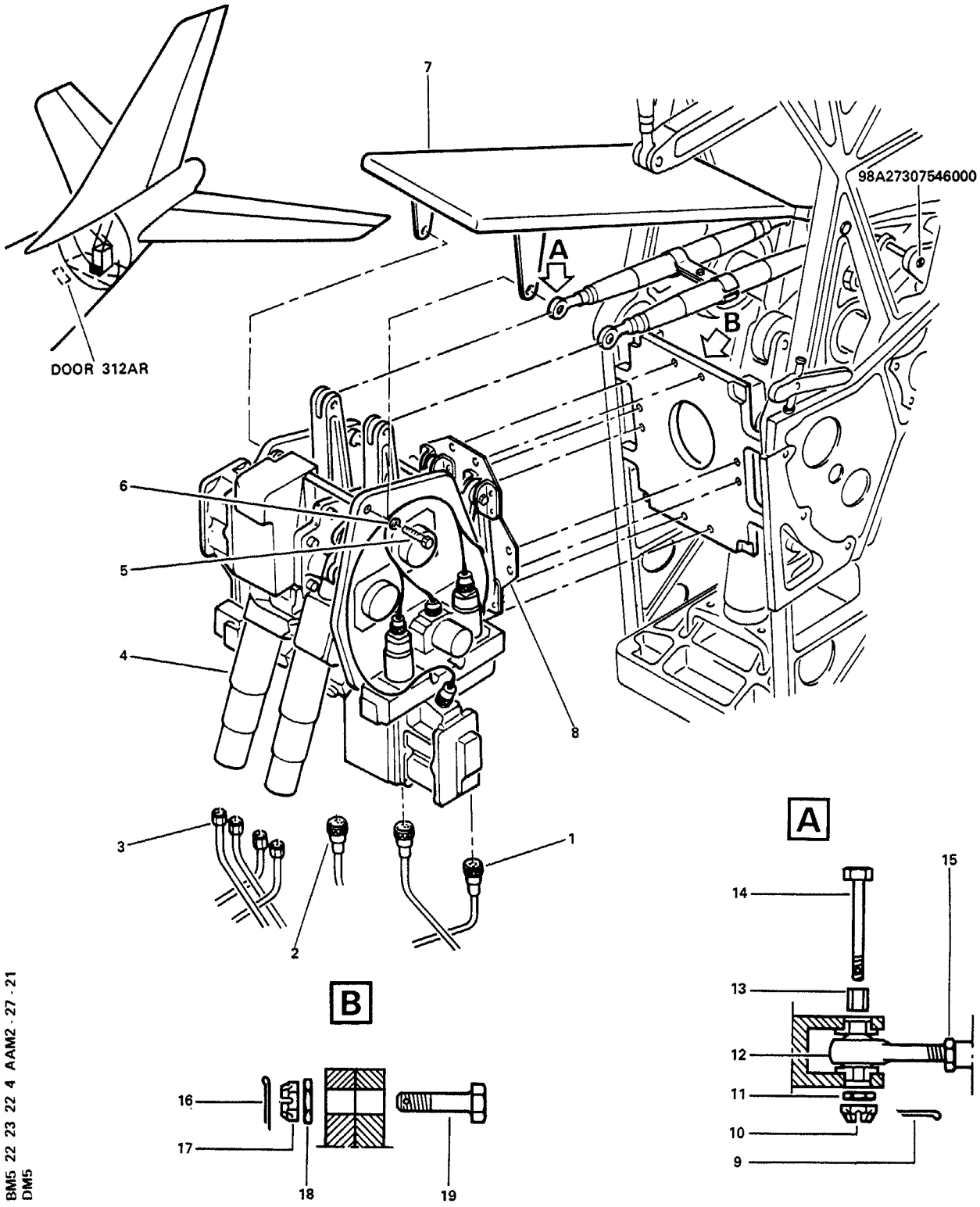
Servo Actuator Pressure Switch
Figure 402

4. Removal/Installation of Yaw Damper Servo-Actuator
(Ref. Fig. 403)

A. Removal of Servo-Actuator
(1)Disconnect electrical connectors (1) and (2) and install blanking caps.
(2)Remove cover plate (7) from actuator (4) as follows :
(a)Cut and remove lockwire.
(b)Remove screws (5) and washers (6) from cover plate (7).
(c)Release mounting brackets and remove cover plate (7).
(3)Remove the two control rods (12) as follows :
(a)Remove and discard cotter pins (9).
(b)Unscrew, remove and retain nuts (10) and washers (11).
(c)Remove bolts (14) and spacers (13).
(d)Remove control rods (12), secure to structure.
(4)Disconnect the four hydraulic lines (3) and install blanking caps.
(5)Remove actuator (4) as follows :
CAUTION : ACTUATOR WEIGHS APPROXIMATELY 25KG (55LB). A SECOND MAN IS REQUIRED.
(a)Remove and discard cotter pins (16).
(b)Remove and retain the eight nuts (17) and eight washers (18) by beginning with those at the bottom.
(c)Remove and retain the eight bolts (19).
(d)Remove actuator (4).

B. Preparation of Replacement Component
(1)Make certain that component is clean and free from dents and corrosion and that securing screws are wirelocked.
(2)Make certain that component is conditioned for use on the aircraft (hydraulic fluid).
(3)Clean and inspect component interface and adjacent area.

C. Installation of Servo-Actuator
(1)Position actuator (4) with its base plate (8) on the aircraft mount.
(2)Install the eight bolts (19) from inner side towards servo-actuator.
(3)Install the eight washers (18) and nuts (17), starting by those at the top, tighten and safety with cotter pins (16).
(4)Remove blanking caps from hydraulic lines (3) and actuator (4), and connect the four hydraulic lines (3) making certain that they are correctly positioned, tighten lines.
(5)Install the two control rods (12) and adjust if necessary as follows :
(a)Cut and remove lockwire and loosen locknut (15) on rod (12).
(b)Unscrew end fitting to enable bolt (14) to be installed freely.
(c)Tighten locknut (15) and wirelock using corrosion resistant steel lockwire, dia. 0.7 mm (0.027 in.).
(d)Install spacer (13) and bolt (14).
(e)Install washer (11) and nut (10), tighten and safety with cotter pin (9).
(6)Install actuator cover plate (7) as follows :
(a)Engage mounting brackets on actuator.
(b)Install washers (6) and screws (5), tighten and wirelock with corrosion resistant steel lockwire, dia. 0.7 mm (0.027 in.)



Yaw Damper Servo-Actuator
Figure 403

(7)Remove blanking caps and connect electrical connectors (1), (2).
(8)Pressurize Blue and Yellow hydraulic systems (Ref. 29-14-00, 29-22-00, 29-23-00, P. Block 301) and make certain that there are no leaks on hydraulic connections.
(9)Remove rigging pin No.98A27307546000.
(10)Remove rigging pins 98A27207601000 and 98A27307545000.
(11)Remove safety clips and tags and close circuit breakers 306CC1, 306CC2, 305CC1 and 305CC2.

5. Test
Carry out an operational test of yaw damper (Ref. 22-26-00, P. Block 501, para. 1.B).

6. Close-Up

A. Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

B. Remove warning notices and/or safety barriers.

C. Close access door 312AR.

D. Remove access platform.

PITCH TRIM ACTUATOR 3CC1 (3CC2) - REMOVAL/INSTALLATION

WARNING : BEFORE PROCEEDING WITH MAINTENANCE WORK ON OR NEAR MECHANICAL FLIGHT CONTROLS OR PRIMARY FLIGHT CONTROL SURFACES, LANDING GEARS, ASSOCIATED DOORS OR ANY MOVING COMPONENT, MAKE CERTAIN THAT GROUND SAFETIES AND/OR WARNING NOTICES ARE IN CORRECT POSITION TO PREVENT INADVERTENT OPERATION OF CONTROLS.

NOTE : Removal/installation procedures for pitch trim actuators 3CC1 and 3CC2 are identical.

1. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform, 6m (19 ft. 8 in.)
B.	Torque Wrench 0.8 m.daN (71 lbf. in.)
C.	Circuit Breaker Safety Clips
D.	Blanking Plugs/Caps
E. Material No. 05-017	Special Materials (Ref. 20-31-00)
F. 97A27302009000	Locking Tool - Pitch Trim Control Wheel
G. 0U190360	Locking Pin - THS Actuator Mechanical Input Lever
H.	Lockwire - Corrosion-Resistant Steel Dia. 0.8 mm (0.032 in.)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 22-27-00, P. Block 501	Electric Pitch Trim-Adjustment/Test

2. Procedure

R (Ref. Fig. 401)

A. Job Set-Up

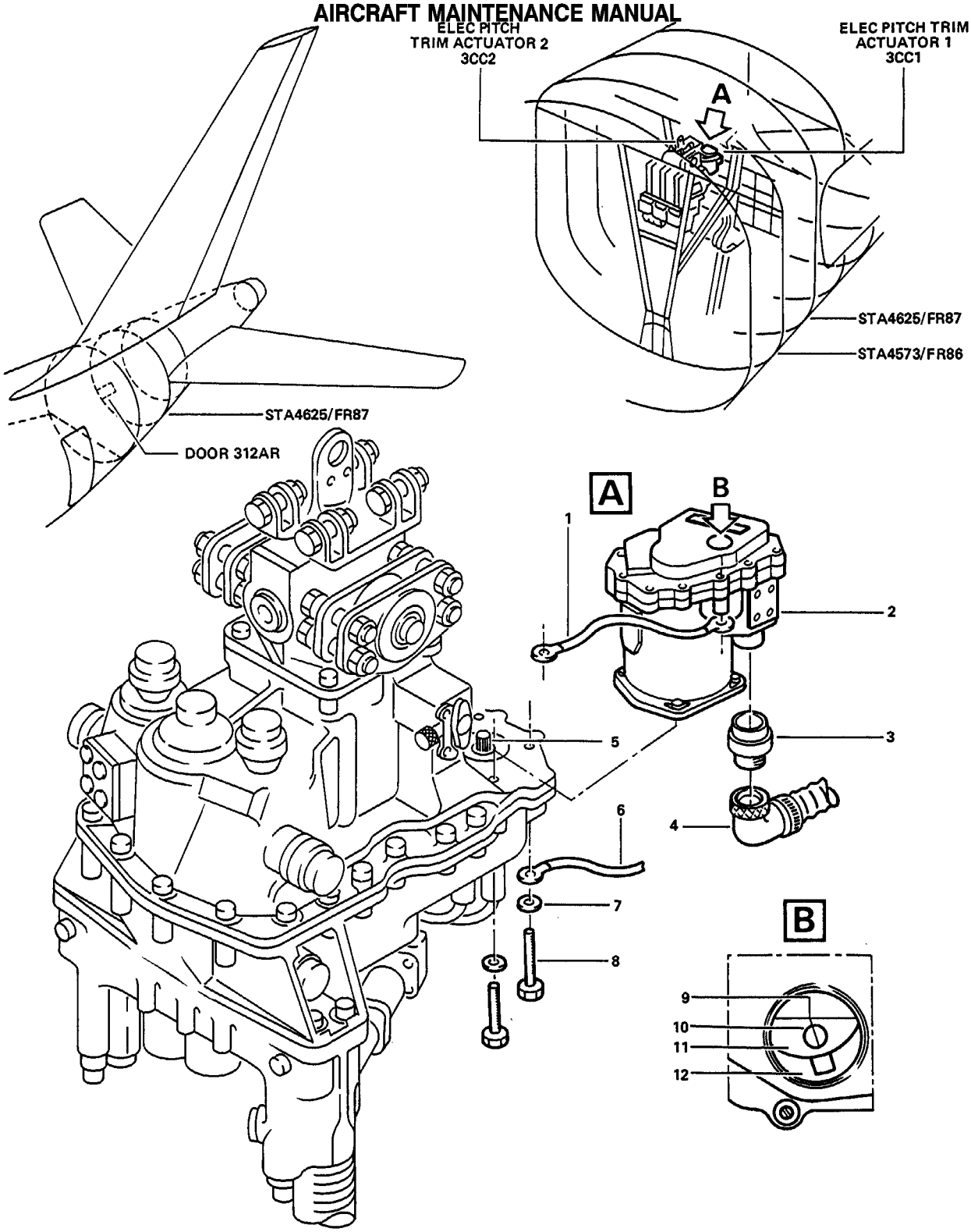
(1)Open, safety and tag the following circuit breakers :

(a)For pitch trim actuator 1 3CC1

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC 1/28VDC	306CC1	103/G 3
21VU	AFS/FAC 1/115VAC	309CC1	103/G 5
21VU	AFS/FAC 1/26VAC	305CC1	103/G 7

(b)For pitch trim actuator 2 3CC2

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC 2/26VAC	305CC2	103/G 8
21VU	AFS/FAC 2/115VAC	309CC2	103/G10
21VU	AFS/FAC 2/28VDC	306CC2	103/G12



Pitch Trim Actuator
Figure 401

R

(2)In flight compartment
- display a warning notice prohibiting operation of flight control surfaces.
- make certain that pitch trim control wheel is in neutral position and immobilize it with locking tool 97A2730200900.

(3)Position access platform at access door 312AR.
(4)Open access door 312AR.
(5)Install locking pin 0U190360 on THS hydromechanical actuator mechanical control.
(6)Depressurize the main hydraulic systems (Ref. 29-10-00, P. Block 301).

B. Removal
(1)Disconnect connector (4).
(2)Cut and remove lockwire. Remove the four mounting bolts (8) and washers (7).
(3)Cut and remove lockwire, remove bolts. Disengage bonding strips (1) and (6). Bonding strip (6) remains connected to the structure.
(4)Remove actuator (2) by raising upwards.
(5)Fit a protective cover on and cap electrical connector (4).

C. Preparation of Replacement Component
(1)Clean and inspect adapter (3) retained on removed actuator. Install adapter (3) of electrical connector.
(2)Install bonding strip (1) retained on removed actuator.
(3)Remove protective cover from actuator and check cleanliness of output shaft.
(4)Make certain, through window (12) that synchro setting is correct. For correct setting the red circle (10) with an oblique line (9) must appear in the yellow sector (11). If necessary, adjust output shaft in order to align synchros.

D. Installation
(1)Clean and inspect THS hydromechanical actuator interface and/or adjacent area for installation of pitch trim actuator.
(2)Remove cap and check connector end for correct condition.
(3)On mating surfaces of pitch trim actuator and THS hydromechanical actuator, apply Material No. 05-017.
(4)Install actuator (2) by lowering on to THS hydromechanical actuator so that hollow splined shaft of actuator (2) correctly engages in splined shaft (5) of THS hydromechanical actuator.
NOTE : Before engaging splines, make certain that pitch trim actuator is as close as possible to correct angular position with respect to THS hydromechanical actuator to avoid misalignment of the synchro reference position.
(5)Install bonding strips (1) and (6) the four washers (7) and bolts.
(6)Install the four mounting bolts (8) and hand-tighten.
(7)Make certain, through window (12), that synchro setting is correct. TORQUE the four mounting bolts (8) to 0.8 m.daN (71 lbf. in.). Safety bolts with lockwire.
(8)Connect electrical connector (4).
(9)Remove locking pin 0U190360.

(10)Remove locking tool from pitch trim control wheel.
(11)Pressurize the main hydraulic systems (Ref. 29-10-00, P. Block 301).
(12)Remove safety clips and tags and close circuit breakers :
- 306CC1, 309CC1 and 305CC1 for pitch trim actuator 1 3CC1
- 305CC2, 309CC2 and 306CC2 for pitch trim actuator 2 3CC2.

E. Test
Carry out TRIM disconnection override test (Ref. 22-27-00, P. Block 501, paragraph 1.B.).
NOTE : For PITCH TRIM actuator 1 perform test with PITCH TRIM 1 lever engaged.
For PITCH TRIM actuator 2 perform test with PITCH TRIM 2 lever engaged.

F. Close-Up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Remove warning notices.
(3)Close access door 312AR.
(4)Remove access platform.
(5)Depressurize Green, Blue and Yellow hydraulic systems.
(6)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(7)Restore system and aircraft to normal operating conditions.

YAW DAMPER ACTUATOR (304CC) SERVO-VALVE - REMOVAL/INSTALLATION

WARNING : BEFORE PROCEEDING WITH MAINTENANCE WORK ON OR NEAR MECHANICAL FLIGHT CONTROLS OR PRIMARY FLIGHT CONTROL SURFACES, LANDING GEARS, ASSOCIATED DOORS OR ANY MOVING COMPONENT MAKE CERTAIN THAT GROUND SAFETIES AND/OR WARNING NOTICES ARE IN CORRECT POSITION TO PREVENT INADVERTENT OPERATION OF CONTROLS.

1. Equipment and Materials

ITEM	DESIGNATION
A.	Access Plaftorm 6 m. (19 ft. 8 in.)
B.	Circuit Breaker Safety Clips
C.	Lockwire - Corrosion Resistant Steel, Dia. 0.7 mm (0.027 in.)
D.	Blanking Caps
E.	Warning Notices and Safety Barriers
F.	Torque Wrench
Referenced Procedures	
- 22-26-00, P. Block 501	Yaw Damper
- 29-10-00, P. Block 301	Main Hydraulic Power (Pressurization/ Depressurization
- 29-14-00, P. Block 301	Hydraulic Reservoir Pressurizing System
- 29-22-00, P. Block 301	Blue Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Lower Transfer Unit)

2. Procedure

A. Job set-up

(1)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC1/26VAC	305CC1	103/G 7
21VU	AFS/FAC2/26VAC	305CC2	103/G 8
21VU	AFS/FAC2/28VDC	306CC2	103/G12

(2)Place warning notices and/or safety barriers.

(3)Position access platform at access door 312AR and open it.

(4)Depressurize Yellow and Blue hydraulic systems (Ref. 29-10-00, P. Block 301 and 29-14-00, P. Block 301).

3. Removal of Servo-Valve on Yaw Damper Actuator
(Ref. Fig. 401)

A. Removal of Servo-Valve
NOTE : The removal of the other servo-valve is identical.
(1)Disconnect electrical connector (2) and install blanking cap.
(2)Remove the lockwire and remove the screws (3) with their flat washers (4).
(3)Disengage the servo-valve (1).
(4)Discard the four O-rings (5).

B. Installation of Servo-Valve
(1)Install the four O-rings (5) on the servo valve (1).
(2)Position the servo-valve (1) on a body.
(3)Install the four screws (3) equipped with their washers (4).
(4)Lock the screws (3) two by two with lockwire.

4. Test
Carry out an operational test of yaw damper (Ref. 22-26-00, P. Block 501, Para. 1.B.).

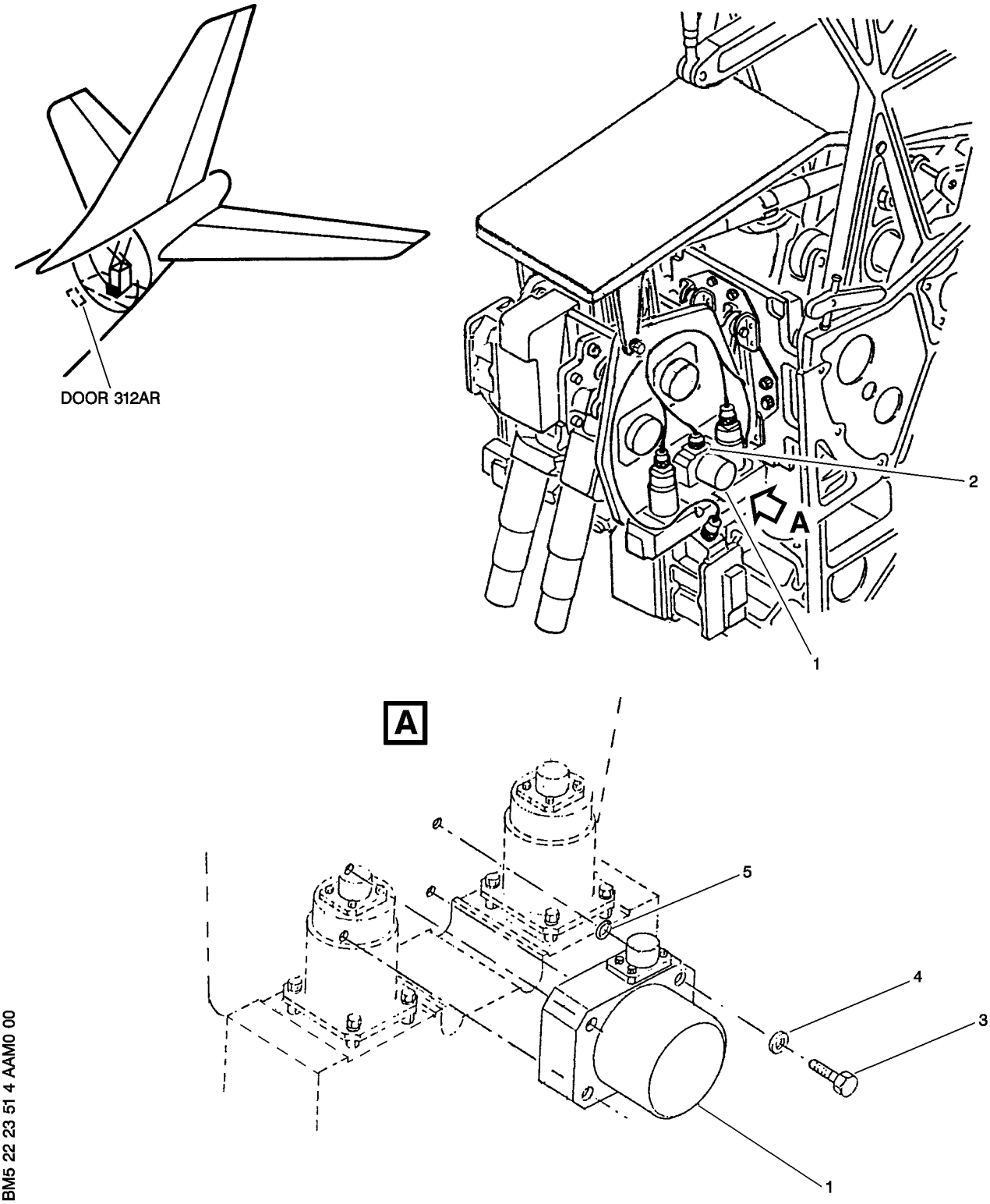
5. Close-up

A. Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

B. Remove warning notices and/or safety barriers.

C. Close access door 312AR.

D. Remove access platform.



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Servo-Valve of Yaw Damper Actuator
Figure 401

1. General

Three types of signals are used :
- Signals transmitted in digital form through ARINC 429 data buses
- Conventional analog signals (synchro type)
- 28V discrete signals - circuit open, ground.

2. ARINC Input Signals

Data bus connections for input signals are given below.
Transmitted data and their characteristics are given in the following tables.
All these signals are in ARINC 429 format.
Sensor signals are :

A. Command Channel
- IRS own (100 K)
- radio altimeter
- TCC in provision (own and opposite)
- MTP
- ADS own
- FAC opposite
- ADS opposite (in parallel, prior to demodulation, with monitoring channel)
- FCU own, in provision
- EFCU (after acquisition and demodulation in monitoring channel)
- monitoring FAC 100 K data bus.

B. Monitoring Channel
- ADS own (in parallel, prior to demodulation, with command channel)
- ADS opposite
- command FAC 12 K data bus
- FAC opposite
- FCU own, in provision
- SFCC own
- EFCU opp
- IRS 3 (100 K)

ARINC acquisition and management principle is described in 22-00 - AUTO FLIGHT-GENERAL

C. ARINC Input Signals - Definition

The following tables give for each parameter :
- the source
- the parameter name
- the octal label
- the function in the FAC.

(Ref. Fig. 001)

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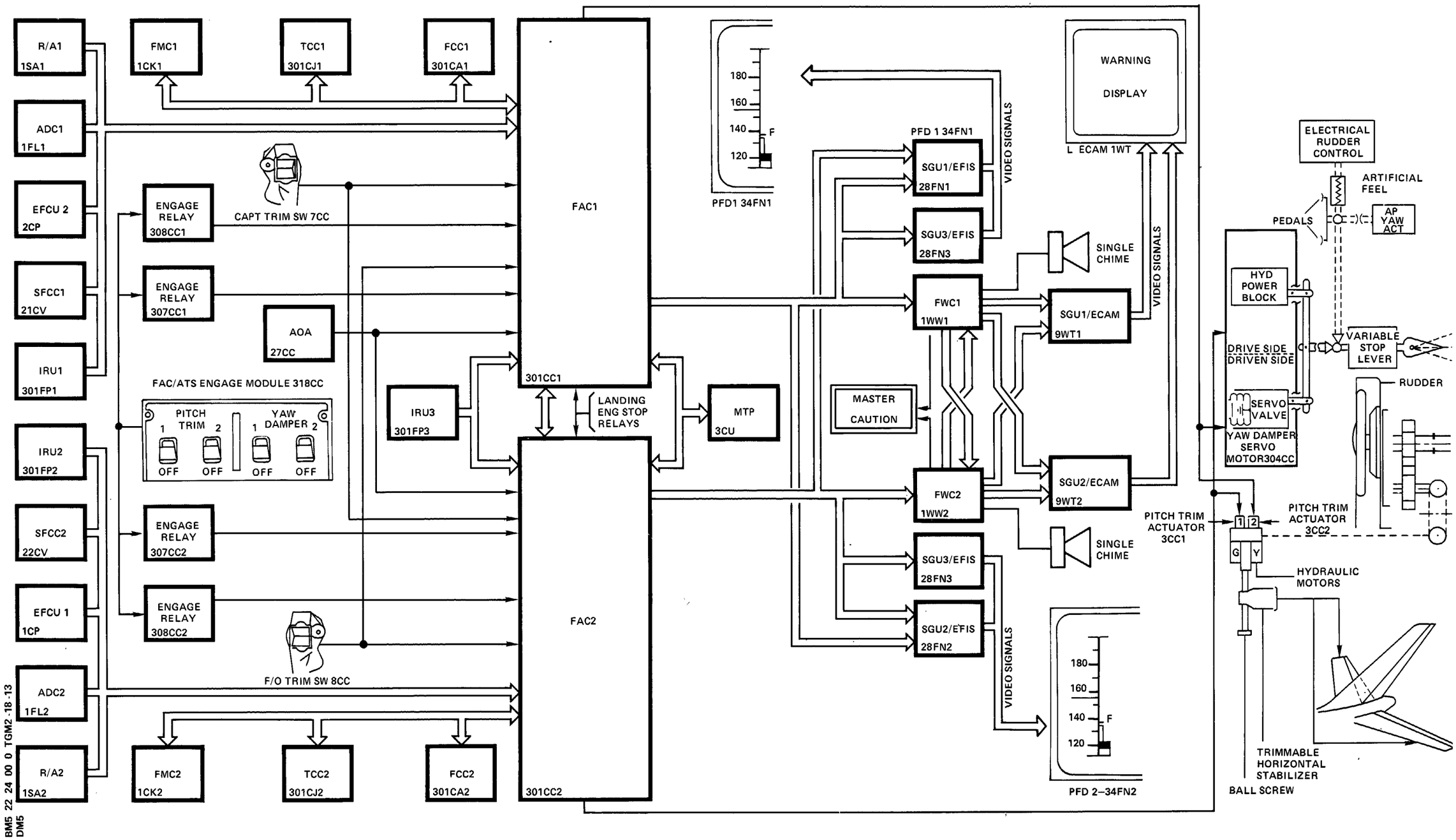

FAC - Block Diagram
Figure 001

Table 1 : ARINC Input Signals

SOURCE	PARAMETER NAME	OCTAL LABEL	FUNCTION	NOTES
RA OWN	RADIO HEIGHT	164	Minimum speed Alpha floor	
IRS OWN	Longitudinal acceleration	331	Windshear - ALPHA FLOOR	
	ROLL ANGLE	325	Yaw damper	
	Lateral acceleration Body Normal Acceleration	332 333	Yaw damper VMIN	(Acquisition at 30ms) (Acceleration in A/C axes)
	Vertical Acceleration	364	Windshear - ALPHA FLOOR	
	Yaw rate	330	Yaw damper - IRS monitoring	(Acquisition at 30ms)
	Pitch Angle	324	Windshear - ALPHA FLOOR	
	Ground Speed	312	Windshear - ALPHA FLOOR	
	Vertical Speed Baro/Inertial	365	Windshear - ALPHA FLOOR	
	Pitch Rate	326	Incidence trim	
ADS OWN	Maximum allowable Airspeed	207	Maximum speed,	
	Mach	205	Gains, Minimum Speed	
	Computed Airspeed	206	Gains, Windshear, Maximum speed,	
	Altitude 1013	203	Maximum speed,	
	True airspeed	210	Gains stab	

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Table 2 : ARINC Input Signals (continued)

SOURCE	PARAMETER NAME	OCTAL LABEL	FUNCTION	NOTES
ADS OWN	CORRECTED ANGLE-OF ATTACK	241	- INCIDENCE TRIM - Angle-of-Attack monitoring - Windshear - VMIN	
MTP	Maintenance word	352		K100AY910 (MTP)
FAC OWN	Average Angle-of-Attack	241	Average Angle-of-Attack monitoring	
	Voted Angle-of-Attack	156	Voted angle-of-attack monitoring	Crosstalk between
	YAW rate	330	IRS Monitoring	command and monitoring channels
	Lateral Acceleration	332	Yaw damper	
	Roll Angle	325	Turn coordination	
	Discrete 2	274	ARINC Monitoring	
	Discrete 1	146	ARINC Monitoring	
	AZ1	333	VMIN	
	Maintenance DATA 1	350		
	Maintenance DATA 2	351		
	Maintenance DATA 3	353		
	TRIMEQUC	155	MON channel equalization	
	Pitch rate	326	Incidence Trim	
	DIHC	107	Incidence Trim	
IRS 3	YAW Rate 3	330	IRS Monitoring	
ADS OPP	CORRECTED ANGLE-OF ATTACK	241	- INCIDENCE TRIM - Angle-of-attack monitoring - Windshield - VMIN	
	COMPUTED AIRSPEED	206	- Gains - VMIN	

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Table 3 : ARINC Input Signals (continued)

SOURCE	PARAMETER NAME	OCTAL LABEL	FUNCTION	NOTES
	Discretes 1	146	Monitoring	
	Discretes 2	274	Monitoring	
FAC	Discretes mon-cmd	157		Crosstalk between
OWN	YAM rate 3	330	IRS Monitoring	command and
INTL	VMIN	245	Characteristic	monitoring
BUS	VMAX	207	Speed Dispatching	channels
	Maintenance DATA 1	350		
	Maintenance DATA 2	351		
	Voted Angle-of	156		
	Attack			
	DIHCM	107	Incidence Trim	
	VMIN	245	12K ARINC OUTPUT	Reconfiguration on
				SFCC OWN loss
	VMAX	207	12K ARINC OUTPUT	
FAC OPP	YRATE OPP	330	IRS Monitoring	
	DISCRETE 2	274	FLAPS/SLAT FROM	
			SFCC OPP	
SFCC	SLATS	127	- YAW DAMPER	
OWN			- TRIM	
			- VMIN	
	FLAPS	137	- YAW DAMPER	
			- TRIM	
			- VMIN	
EFCU	SPEED BRAKES	072	- VMIN, TRIM	
OPP				
	SPOILERS	071	- YAW DAMPER	

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3. ARINC Output Signals

A. General

The FAC transmits SIGNALS in ARINC 429 form :
- through 12K main data bus to the FCCs, TCCs, MTP, the other FAC and the monitoring channel of the considered FAC
- through 12K specific data bus to the EFIS and FMC
- through 100K internal data bus from the monitoring to the command channel of the considered FAC.

The following tables show the characteristics of the transmitted signals :
- label
- parameter name
- transmission rate
- significant bits
- units
- range.

Discrete words as well as maintenance words are also given in details. It is to be noted as a reminder (see AUTO FLIGHT - GENERAL) that the 12 K main data bus is monitored through feedback to the monitoring channel. Specific monitorings concerning transmission are described in 22-30.

B. FAC Transmission Data Bus Monitoring

The FAC transmits a certain number of parameters to some users (FCC1, FCC2, TCC1 and TCC2) via its 12K data bus line, in ARINC 429 form. These parameters are transmitted with a status matrix whose contents means :
- if NCD : failure of detector used to compute the data.
- if F/W : data computation failure
- if No refresh : incorrect FAC transmission monitored by a looping method in the monitoring channel of 12K data bus.

A specific monitoring function has been integrated to enable reconfiguration of users on the healthy FAC data bus.
This function consists in cutting the data bus line for certain tests after a 1-second delay.
Cutoff of 12K line entails the following results :
- user advising
- associated FAC loss forcing through a loop (pitch trim, yaw damper part).

Another interest of line cutoff monitoring is to cancel failures which are only detected at this level.

C. Monitoring of Main 12K Line to FCC and TCC (Ref. Fig. 003)
This line is monitored in the command and monitoring channels. In the event of a fault detection, the line is cut after a 1-second delay to enable possible reconfiguration of users on the healthy FAC bus line.
This cutoff demand which is in fact a transmission interrupt is achieved following the conditions described below :
- internal power supply monitoring test (this type of monitoring is crossed between the command and monitoring channel so that the healthy channel can detect the failure). Delay is of 500 ms.
- real-time clock monitoring
- safety test concerning the digital central unit as well as the pitch trim and yaw damper common parts
- ARINC acquisition and transmission monitoring (ARINC management board = 12K in the command channel and 100K in the monitoring channel)
- refresh test on discrete word 2. This word is transmitted by the 12K data bus and fed back to the monitoring channel. Its check gives an indication of the transmission quality
- watchdog status test.

The same conditions and angle-of-attack monitoring are processed in the monitoring channel. The result is transmitted through a wired OR connection to condition the 12K ARINC transmission.
ARINC transmission can start again by attempting to place PITCH TRIM or YAW DAMPER engage lever in ON position if the lever had previously returned to OFF position.

D. EFIS 12K Line Monitoring (Ref. Fig. 002)
This line has a simplified monitoring :
- monitoring of the non-disabled characteristic of the EFIS coupler transmission. A test is performed between transmission request and transmission time.
  If a discrepancy is detected, the EFIS line is cut and it will only be enabled if the coupler is at right state again.
  Action on the engage lever is not effective.
- transmitted word status matrix monitoring
  In the event of a detection as described in the previous paragraph concerning the main data bus :
  . all the airspeed parameters are assigned a F/W code
  . discrete word 2 is assigned a F/W code.

(Ref. Fig. 003)

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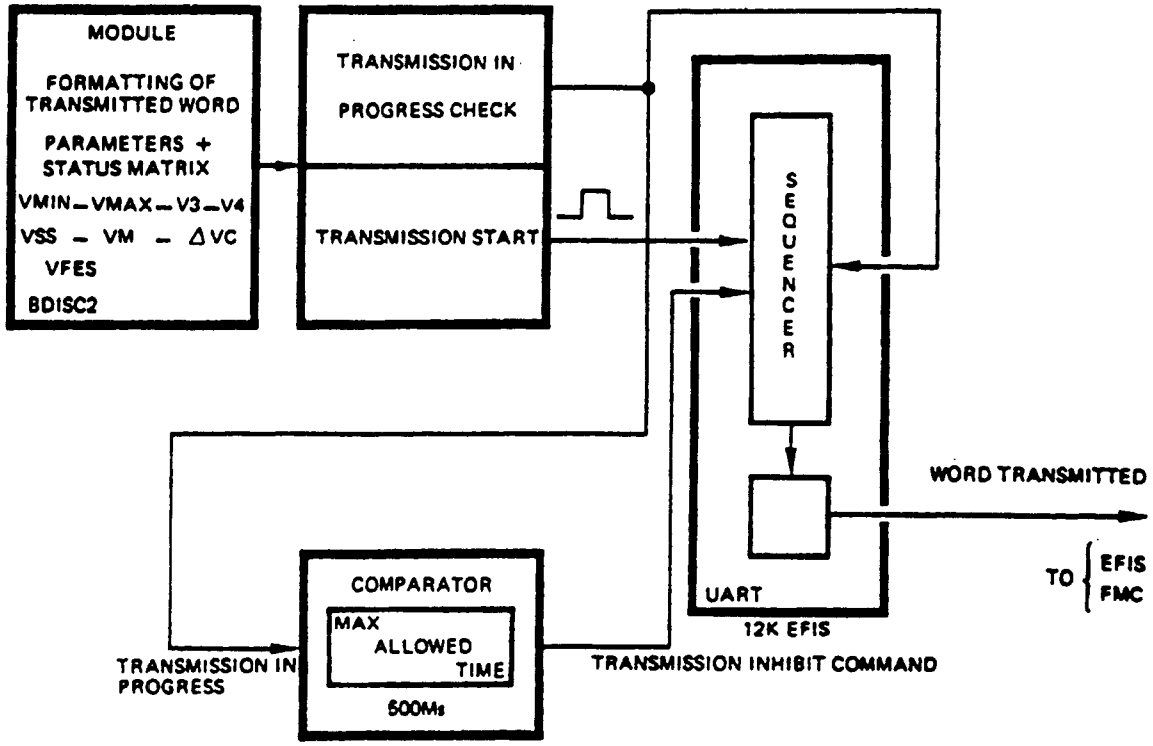
Table 4 : 12K ARINC Command Line Outputs



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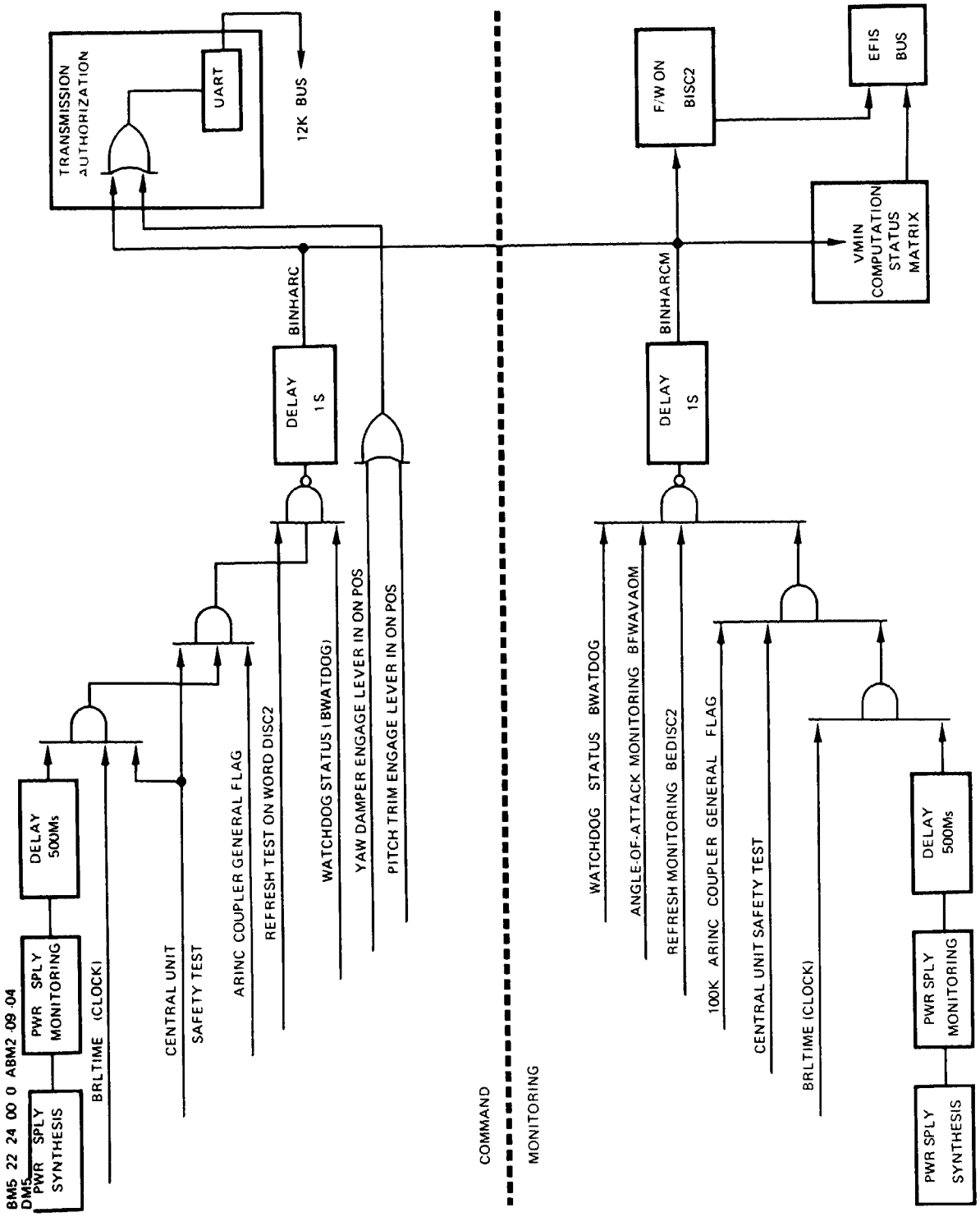
Monitoring of FAC-to-EFIS Transmission
Figure 002



12K Bus Transmission Monitoring
Figure 003

OCTAL LABEL	PARAMETER NAME	CYCLE OUTPUT	TRANSMIT INTERVAL (ms)	SIG BITS (NOT INC SIGN)	UNITS	RANGE	NOTES (resolution)
245	Minimum Airspeed	5	150	16	Knots		
207	Placard Speed	5	150	16	Knots	1024	
333	AZI	2	150	16	g	1	
330	Body Yaw Rate	All	30	16	°/s	±15	
332	Lateral Acceleration	All	30	16	g	±4	
325	Roll Angle	2	150	16	Deg	±45°	
146	AFS Discretes *	All	30				
241	Average Angle-of-Attack	3	150	16	Deg	±25°	
274	AFS Discretes 2 (FAC) *	1	150				
350	Maintenance Data 1 *	| - 1	150				Transmission of these discretes is not systematic
351	Maintenance Data 2 *	| - 1	150				Transmission of these discretes is not systematic
353	Maintenance Data 3 *	| - 1	150				Transmission of these discretes is not systematic
156	Voted Angle-of-Attack	3	150	16	Deg	±25°	
155	Trim Equalization	All	30	16	Deg	±20°	
107	DIHCC	4	150	16	Deg	±20°	
205	Mach	5	150	16	Mach	1	NOT ARINC
324	Pitch rate	1	150	16	°/s	±30°	

* See discretes definition complementary table

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Table 5 : EFIS FMC 12K ARINC Line Outputs (Monitoring Channel)

OCTAL LABEL	PARAMETER NAME	CYCLE OUTPUT	TRANSMIT INTERVAL (ms)	SIG BITS (NOT INC SIGN)	UNITS	RANGE	NOTES (resolution)
245	Minimum Airspeed		150	16	Knots	512	
207	Maximum Airspeed		150	16	Knots	1024	
262	Vc Trend		150	16	Knots	±256	
263	V3		150	16	Knots	512	Minimum flap retraction speed
264	V4		150	16	Knots	512	Minimum slat retraction speed
265	VM		150	16	Knots	512	Maneuvering speed
256	VSS		150	16	Knots	512	Stick shaker speed
274	Discrete 2 *		150				Discrete word

OCTAL LABEL	PARAMETER NAME	CYCLE OUTPUT	TRANSMIT INTERVAL (ms)	SIG BITS (NOT INC SIGN)	UNITS	RANGE	NOTES (resolution)

* See discretes definition complementary table

Table 6 : AFS Discretes 1 FAC ARINC Outputs (BFDISC1)

BIT No.	FUNCTIONS	ENCODING		NOTES
		ZERO	ONE	
1-8	Label			
9				at 0
10				
11	Yaw Damper Engagement	Not Engaged	Engaged	
12	Engine Failure Compensation	OFF	ON	
13	Turn Coordination	OFF	ON	
14	IRS Monitoring	Not Valid	Valid	
15				
16	Trim inhibit on limits	Not Inhibit	Inhibit	
17	Trim Engaged	Not Engaged	Engaged	
18	Synchronisation	OFF	ON	
19	Electric Trim	OFF	ON	
20	Auto Trim	OFF	ON	
21	Mach Trim	OFF	ON	
22	Incidence Trim	OFF	ON	

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23	Windshear Warning	Not active	Active	
24	Windshear warning not available	Available	Not available	
25				
26				
27				
28				
29				
30				
31				
32	Parity			

**ON A/C ALL

Table 7 : AFS Discretes 2 FAC ARINC Outputs (BFDISC2)

BIT No.	FUNCTIONS	ENCODING		NOTES
		ZERO	ONE	
1-8	Label			
9				
10	Slats/Flaps selector pos.			

BIT No.	FUNCTIONS	ENCODING		NOTES
		ZERO	ONE	
11	Position 1	only one bit and one at least set to the 1 state.		
12	Position 2			
13	Position 3			Only 1 bit at a time
14	Position 4			
15	Position 5			
16	AIRBRAKES OUT	IN OR FAILED	OUT	
17	Own SFCC validity	NOT VALID	VALID	
18	Both SFCC valid			
19	LEFT GEAR PRESSED	NOT PRESSED	PRESSED	
20	RIGHT GEAR PRESSED	NOT PRESSED	PRESSED	
21	GEAR SW VALIDITY	NOT VALID	VALID	
22	Own EFCU validity	NOT VALID	VALID	
23	Both EFCU validity	NOT VALID		
24	SPEED LIMITS VALIDITY	NOT ACTIVE	ACTIVE	If 5V pin programming
25				
26	Throttle Pusher	alpha < alpha 0	alpha ≥ alpha 0	
27	Windshear warning own	NOT ACTIVE	ACTIVE	
28	OPP Throttle Pusher	alpha < alpha 0	alpha ≥ alpha 0	
29	Windshear warning	NOT ACTIVE	ACTIVE	
30	|- SSM			
31	|			
32	Parity			

Table 8 : AFS Discretes 2. FAC EFIS output (BFDISC2)

BIT No.	FUNCTIONS	ENCODING		NOTES
		ZERO	ONE	
1-8	Label			
9				
10	Flap and slat control lever positions :			
11	Position 1	Only one bit, and at least one, set to 1		
12	Position 2			
13	Position 3			
14	Position 4			
15	Position 5			
16	Speedbrakes out	in or failed	out	
17	OWN SFCC VALIDITY	NOT VALID	VALID	
18	BOTH SFCC VALIDITY	NOT VALID	ONE VALID	
19	LEFT GEAR PRESSED	NOT PRESSED	PRESSED	
20	RIGHT GEAR PRESSED	NOT PRESSED	PRESSED	
21	GEAR SW VALIDITY	NOT VALID	VALID	
22	OWN EFCU VALIDITY	NOT VALID	VALID	
23	BOTH EFCU VALIDITY	NOT VALID	ONE VALID	
24	SPEED LIMITS VALIDITY	NOT VALID	VALID	
25	Windshear Warning	NOT ACTIVE	ACTIVE	
26	Windshear Warning not available	AVAILABLE	NOT AVAILABLE	
27				
28	Stall-trim	NOT ACTIVE	ACTIVE	
29				
30				
31				
32	Parity			

Table 9 : Maintenance Data 1 Discrete Word ARINC Outputs

BIT No.	FUNCTIONS	ENCODING	NOTES
1-8	Label		
9	-|	000 LRU table	
10	|- Command	001 Additional Data Transmission	
11	-|		
12	-|	010 Takeoff	
13	|- Number of transmitted LRUS		
14	-|		
15	Provision		
16	-|		
17	|		
18	|		

BIT No.	FUNCTIONS	ENCODING	NOTES
19	|- LRU1		
20	|		
21	|		
22	|-		
23	|-		
24	|		
25	|		
26	|- LRU2		
27	|		
28	|		
29	|-		
30	SSM		
31	"		
32	Parity		

Table 10 : Maintenance Data 2 Discrete Word ARINC Outputs

BIT No.	FUNCTIONS	ENCODING	NOTES
1-8	Label		
9	-		
10			
11			
12	- LRU3		
13			
14			
15	-		
16	-		
17			
18			
19	- LRU4		
20			
21			
22	-		
23	-		
24			
25			
26	- LRU5		
27			
28			
29	-		
30	SSM		
31	"		
32	Parity		

Table 11 : Maintenance Data 3 Discrete Word ARINC Outputs

BIT No.	FUNCTIONS	ENCODING	NOTES
1-8	Label		
9	-|	000 LRU table	
10	| -- Command	001 Additional Data	
	|	010 Takeoff	
11	-|	110 Synchronization	In the event of a
12	-|		synchronization
13	| -- Number of transmitted		control, only bit
14	-| LRU		Nos 12 to 18 are
15	Provision		used and contain
16	-|		the value of the
17	|		synchronization
18	|		counter
19	| -- LRU1		
20	|		
21	|		
22	-|		
23	-|		
24	|		
25	|		
26	| -- LRU2		
27	|		
28	|		
29	-|		
30	SSM		
31	"		
32	Parity		

4. Non Digital Input Signals (Ref. Fig. 004)

The following tables show :
- the input analog signals
- the input discrete signals

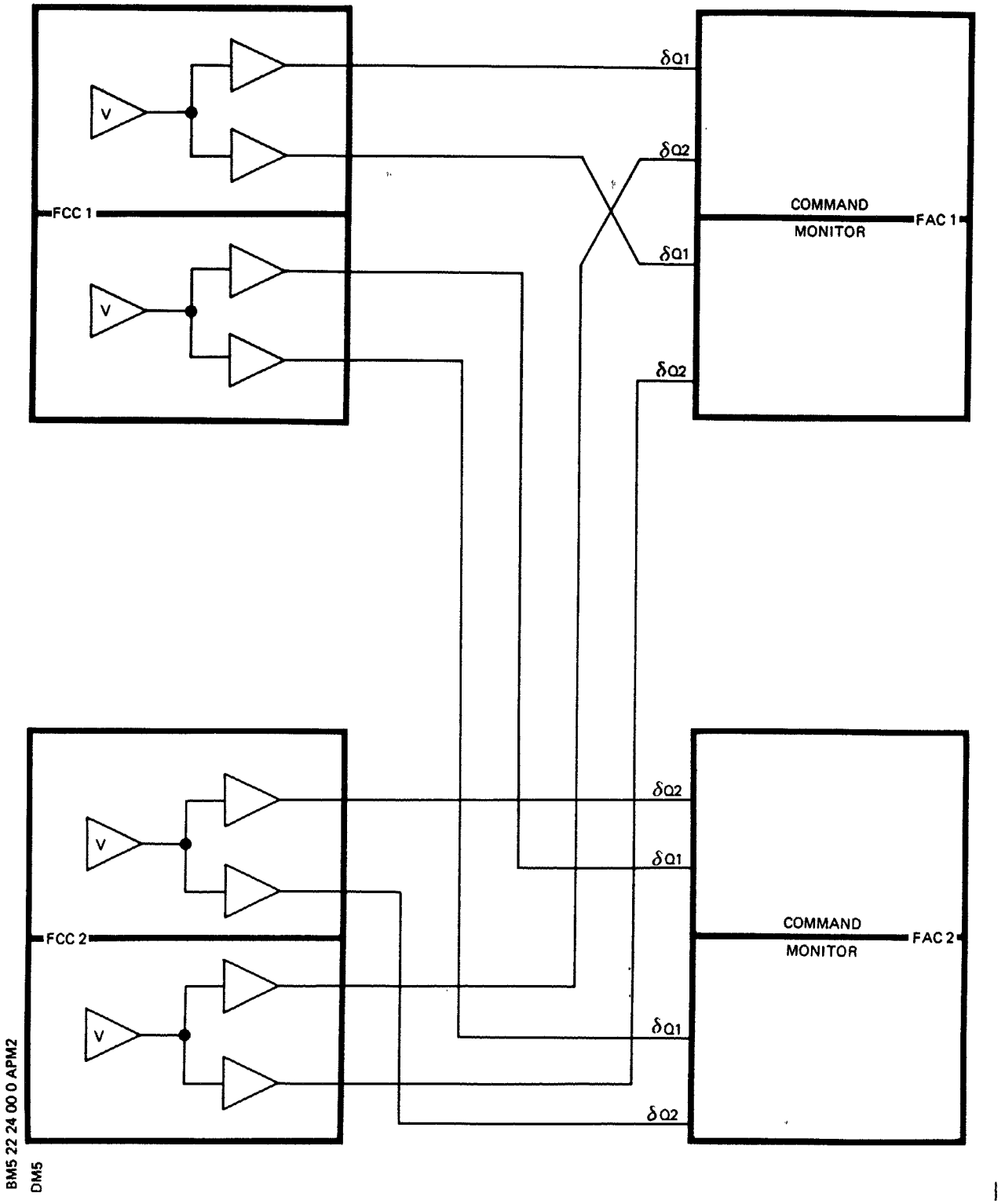
together with their characteristics.
These signals are used :
- in the power loop slaving (position feedback signals, ...)
- in the autotrim command generation. The non digital format is then chosen for safety reasons. (Ref. Fig. 005)
- in the monitoring logic generation (discrete signals).

FCC-FAC interconnection concerning autotrim commands is shown in figure (Ref. Fig. 013). In the event of an interconnection loss between the FAC and the FCC, the retained principle enables to lose one FAC only through FAC internal monitoring.
The AP voter is monitored by both the FCC voter comparators and power comparators.

BM5 22 24 00 0 ANM2 17 12
DM5

SIGNAL	TYPE	GRADIENT	RANGE	SENSE > 0 PARAMETER	REFERENCE	POLARITY PHASE SIGNAL FOR FUNCTION > 0	OUT OF PHASE	FUNCTION	COMMENTS
ANGLE OF ATTACK 3	4 WIRE RESOLVER	10.8 SIN α SYNCHRO 10.8 COS α SYNCHRO	± 60°	AOA IN NOSE - UP SENSE IN PHASE WITH 26V 400 HZ	NULL AOA = 0°		0.9 TO 24°	AOA PROCESSING	SEE PARA 7.B.(2)
TRIM ERROR 1 COMMAND	2 WIRE DC SIGNALS	500 MV/° CONTROL SURFACE	± 20 CONTROL SURFACE	TRIM COMMAND IN NOSE - UP SENSE	NULL TRIM COMMAND	POSITIVE SIGNAL ON PIN TP 9 A WITH RESPECT TO TP 9B		AUTO-TRIM	
TRIM ERROR 1 MONITOR						POSITIVE SIGNAL ON PIN MP 9A WITH RESPECT TO MP – 9B			
TRIM ERROR 2 COMMAND						POSITIVE SIGNAL ON PIN TP 13J WITH RESPECT TO TP 13K			
TRIM ERROR 2 MONITOR						POSITIVE SIGNAL ON PIN MP 13J WITH RESPECT TO MP 13K			
TRIM FU1	2 WIRE AC	200 MV/° SYNCHRO 2.65° SYNC = 1° THS	± 2° -15° THS	THS DISPLACEMENT WITH TENDENCY TO CANCEL OUT OF TRIM IN NOSE-DOWN SENSE (HEAVY NOSE)	-5.1° THS = NULL SYNCHRO SIGNAL	SIGNAL IN PHASE ON PIN : TP 8D WITH RESPECT TO TP 8E FOR THS POSITION LESS THAN – 5.1°		TRIM MONI-TORING (SOFTWARE) TRIM CONTROL (HARDWARE)	
TRIM FU2						SIGNAL IN PHASE ON MIN : MP 8D WITH RESPECT TO MP 8E FOR THS POSITION LESS THAN – 5.1°	8.5 ± 2°	TRIM MONI-TORING (SOFTWARE) PITCH (SOFTWARE)	
TACHO	AC	1.3 TO 1.6 VAC FOR 3000 RPM	0 - 11000 RPM	DISPLACEMENT SPEED WITH TENDENCY TO CANCEL OUT OF TRIM IN NOSE-DOWN SENSE (HEAVY NOSE)	THS NULL DISPLACEMENT RATE	SIGNAL IN PHASE ON PIN MP 15E WITH RESPECT TO MP 15F	0 ± 17°	TRIM CONTROL (HARDWARE)	
YAW DAMPER FU1	AC	533 MV/° CONTROL SURFACE AT LOW SPEED	+ 10°	RUDDER TO THE RIGHT WITH RESPECT TO AIRCRAFT REFERENCE LOOKING FORWARD	0	SIGNAL IN PHASE ON PIN TP 9D WITH RESPECT TO MP 9E	11	SERVO LOOP (HARDWARE)	GRADIENT DEPENDS ON THE SPEED FOLLOWING VARIABLE LEVER ARM LAW
YAW DAMPER FU2						SIGNAL IN PHASE ON PIN : MP 9D WITH RESPECT TO MP 9E		IMAGE LOOP COMPARA-TOR (HARDWARE)	

Table 12 : Input Analog Signals
Figure 004



FCC Output Voter Signal Interconnection Schematic
Figure 005

Table 13 : FCC Input Discrete Signals

SIGNAL	STATUS		FUNCTION	COMMENTS
	IN EFFECT	NOT IN EFFECT		
SRS MODE OWN	Ground	Open	Switch computation of the yaw damper engine failure compensation law	
AP OWN CWS	28VDC	Open	Auto trim term selection	CMD and MON duplicated signal
LD1 CWS OWN	28VDC	Open		CMD and MON duplicated signal
AP OWN ENGD	28VDC	Open	Trim mode logic	CMD and MON duplicated signal
SRS MODE OPP	Ground	Open	Switch computation of the yaw damper engine failure compensation law	
AP OPP CWS	28VDC	Open	Auto trim term selection	CMD and MON duplicated signal
AP OPP ENGD	28VDC	Open	Trim mode logic	CMD and MON duplicated signal
LD1 CWS OPP	28VDC	Open		CMD and MON duplicated signal
ANGLE-OF-ATTACK TEST	Ground	Open	Alpha probes test on the ground	
STALL WARNING OWN	Ground	Open	Yaw damper	
STALL WARNING OPP	Ground	Open	" "	

Table 14 : FAC Opposite Input Discrete Signals

SIGNAL	STATUS		FUNCTION	COMMENTS
	IN EFFECT	NOT IN EFFECT		
FAC ON FROM OPP	28VDC	Open	Ground energization circuits	
YAW DAMPER TOGGLE SW (CMD)	28VDC	Open	Yaw damper engage logic	
TRIM TOGGLE SW (CMD)	28VDC	Open	Pitch trim engage logic	
YAW DAMPER TOGGLE SW (MON)	28VDC	Open	Yaw damper engage logic	
TRIM TOGGLE SW (MON)	28VDC	Open	Pitch trim engage logic	
PRIORITY (CMD)	28VDC = TRIM 1 engaged	Open = TRIM 1 not engaged	Pitch trim engage logic	Only on FAC2
PRIORITY (MON)	28VDC = TRIM 1 engaged	Open = TRIM 1 not engaged	Pitch trim engage logic	Only on FAC2

Table 15 : Input Discrete Signals

SIGNAL	STATUS		FUNCTION	COMMENTS
	IN EFFECT	NOT IN EFFECT		
LEFT GEAR PRESSED SW	28VDC	Open	- Trim - Ground energization circuit	Data re-transmitted to other systems
RIGHT GEAR PRESSED SW	28VDC	Open	- Trim - Ground energization circuit	Data re-transmitted to other systems
NOSE GEAR SW	28VDC shock absorber compressed	Open	Ground energization circuit Ground test validity	CMD and MON duplicated signal
L/R ENGINE STOP	28VDC = Engine shutdown	Open = Engine running	Ground energization circuit	
GEAR DOWN	28VDC	Open	Flight envelope protection	
SIDE	28VDC = Side 2	Open = side 1	Standby alpha probe demodulation reference selection. Auto trim term selection	
HYD CIRCUIT HTY	Ground	Open	Yaw damper engage logic	
28VDC GEAR	28VDC	Open	Monitoring of landing gear shock absorber relay power supply	CMD and MON duplicated signal

Table 16 : Discrete Input Signals

SIGNAL	STATUS		FUNCTION	COMMENTS
	IN EFFECT	NOT IN EFFECT		
TEST POWER ON	28VDC	Open	Ground energization control circuits	
TCC OWN SELECT	28VDC	Open	Selection of the TCC ARINC line used by the FAC	In provision
TCC OPP SELECT	28VDC	Open		

Table 17 : Input Discrete Signals

SIGNAL	STATUS		FUNCTION	COMMENTS
	IN EFFECT	NOT IN EFFECT		
CPT TRIM SW UP	28VDC	Open	Electric trim term computation	
CPT TRIM SW DN	28VDC	Open	Electric trim term computation	
F/O TRIM SW UP	28VDC	Open	Electric trim term computation	
F/O TRIM SW DN	28VDC	Open	Electric trim term computation	
TRIM TOGGLE SW (CMD)	28VDC	Open		
TRIM CLUTCH RET (MON)	Ground	Open		
TRIM SW HOLD SUP	28VDC	Open	A/C relays power supply	
TRIM SW HOLD OUT	28VDC	Open		

5. Non Digital Output Signals

The following tables show the signals transmitted to the users, together with their characteristics, in :
- analog form
- discrete form.

These signals are used mainly for :
- pitch trim and yaw damper power loop slaving
- relays on engage or clutching winding control
- other system logic (FCC, TCC) for FAC data bus selection logics or engage logic.

Table 18 : Output Analog Signals

SIGNAL	TYPE	GRADIENT	RANGE	POSITIVE SENSE	MAXIMUM CURRENT	COMMENTS
TRIM MOTOR CONTROL	AC Signal	115VAC =	- 15° to + 2° THS	Opposite phase to 26 V = pitch-down	600 mA RMS	Trim motor control voltage is : 115 V, 400 Hz a certain number of alternations of which are allowed to pass depending on the slaving error. The phase of the signal + or -90° with respect to a reference voltage applied to a motor winding. The sign of the phase depends on the desired direction of rotation.
YAW DAMPER SERVO DRIVE	2-wire DC signal	0.2 mA/° /s control surface	+10° control surface	Positive = control surface deflection to right command signal (looking forward)	25 mA	This output operates in current generation

Table 19 : Output Discrete Signals

SIGNAL	STATUS		MAXIMUM CURRENT	COMMENTS
	IN EFFECT	NOT IN EFFECT		
YAW DAMPER ENDG OWN (CMD and MON)	28VDC	Open	100 mA	
YAW DAMPER ENGD OPP (CMD and MON)	28VDC	Open	100 mA	
TRIM ENGD OWN (CMD and MON)	28VDC	Open	100 mA	
TRIM ENGD OPP (CMD and MON)	28VDC	Open	100 mA	
YAW DAMPER CLUTCH COIL (CMD)	28VDC	Open	0.7 A	These signals are used as connections with engage unit
YAW DAMPER TOGGLE HOLD COIL (MD)	Ground	Open	0.7 A	
YAW DAMPER CLUTCH COIL (MON)	Ground	Open	0.7 A	
YAW DAMPER TOGGLE HOLD COIL (MON)	Ground	Open	0.7 A	
TRIM TOGGLE HOLD COIL (CMD)	Ground	Open	0.7 A	
TRIM CLUTCH COIL (CMD)	28VDC	Open	0.7 A	
TRIM CLUTCH (MON)	Ground	Open	0.7 A	
TRIM TOGGLE HOLD COIL (MON)	Ground	Open	0.7 A	
CPPSON	Open	Ground	40 mA	When in open condition, control box and engage unit energization is required
TRIM RELAY SUP	28VDC	Open		A/C relay power supply
TRIM 28V SUP	28VDC	Open		K151 power supply
FAC ON TO FCC	28VDC	Open	40 mA	FCC energization is required

SIGNAL	STATUS		MAXIMUM CURRENT	COMMENTS
	IN EFFECT	NOT IN EFFECT		
WHOOLER	28VDC	Open		Electric trim function
TCC PS INHIBIT	28VDC	Open	40 mA	TCC energization is required.

6. Interconnection between FACs and Alpha Probes

A. Received Data (Ref. Fig. 006)

The FACs receive angle-of-attack data signals from three different sources :

- in digital form through ADC1 data bus for left angle-of-attack
- in digital form through ADC2 data bus for right angle-of-attack
- in analog form from the standby alpha probe.

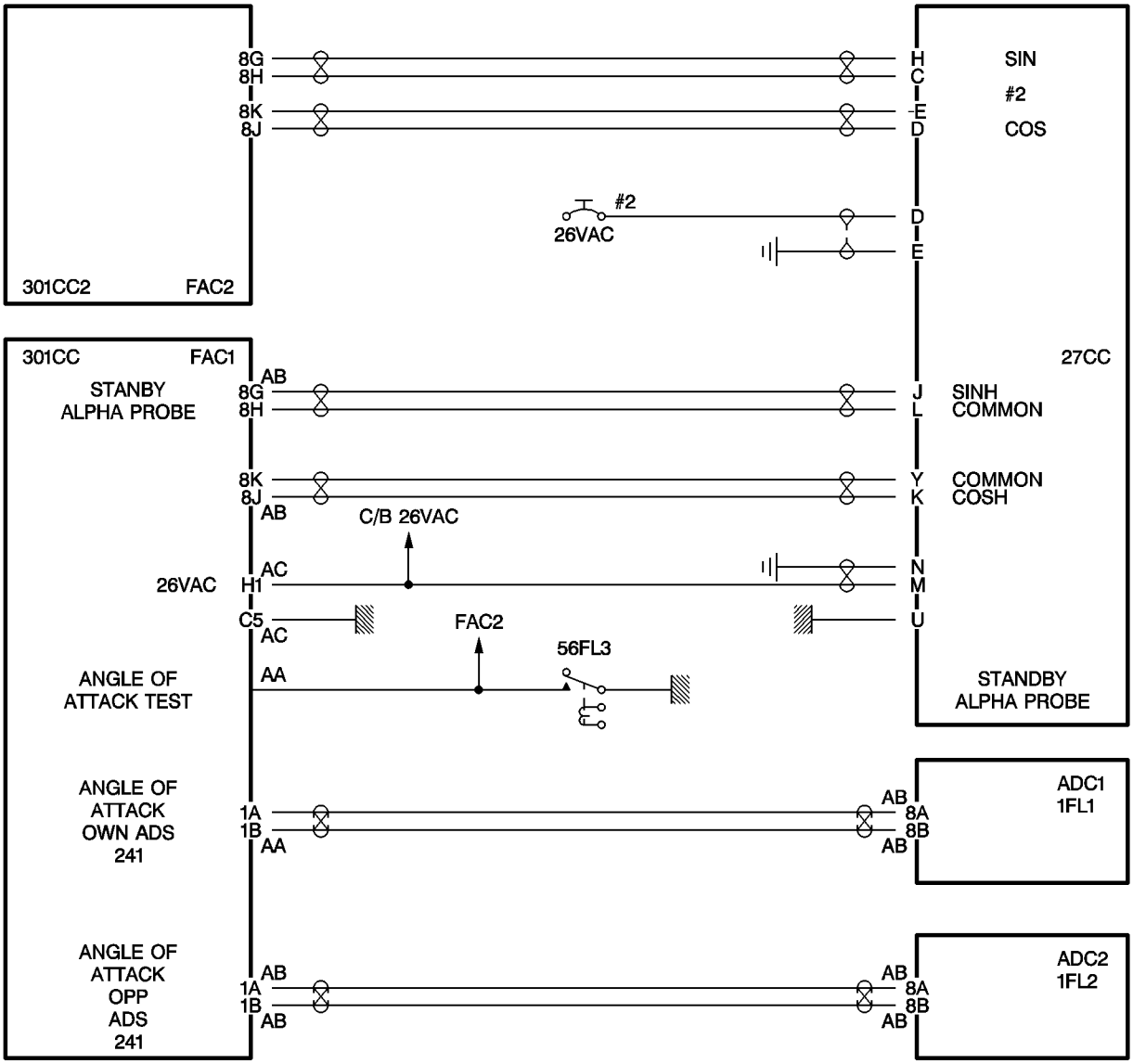
In addition, each FAC receives an alpha probe test initiation signal (tests concerning the three alpha probes : 1+2+3). This test can be activated on the ground on the lateral panel

- Type of signal :
  ground = probe test in progress
  open circuit = probe test not in progress.

The signals transmitted by the ADC are obtained in ARINC format.
The signals transmitted by the standby alpha probe are processed in a specific way as shown in the following figure (Ref. Fig. 006).
The FACs generate two types of signals in true angle-of-attack from received data :

- voted angle-of-attack
- average angle-of-attack.

These parameters are transmitted to the AFS.
On the ground the FAC transmits a null angle-of-attack signal except in test configuration during which the test value is re-transmitted.



BM5 22 24 00 0 BAM2 00

Angle-of-Attack Data - Interconnection Schematic
Figure 006

B. True Angle-of-Attack and Local Angle-of-Attack

Two different types of angle-of-attack data are received by the FAC :
- data delivered by the ADS
- data delivered by the standby alpha probe.

(1)ADS angle-of-attack data

The angle-of-attack data delivered by the ADS are directly given in true angle-of-attack value :

(2)Standby alpha probe data

Data signals are delivered in local angle-of-attack form and the FAC must calculate the true angle-of-attack as per the following formulas :
- local AOA = rotation of mobe resolver +25° :
- when flaps in : true AOAl ≤ 12° :

$$\text{True AOA} = \frac{\text{local AOA} + 2.29}{1.79}$$

True AOA > 12° :

$$\text{True AOA} = \frac{\text{local AOA} + 1.44}{1.72}$$

- when flaps out : True AOA ≤ 15°

$$\text{True AOA} = \frac{\text{local AOA} + 0.55}{1.73}$$

True AOA > 15° :

$$\text{True AOA} = \frac{\text{local AOA} - 3.2}{1.48}$$

C. Angle-of-Attack Monitoring

(1)General

The FAC generates the two following types of angle-of-attack data for its own utilization and transmits them to the TCC and FCC :
- voted angle-of-attack delivered by three different information streams
- average angle-of-attack delivered by two different information streams.

Data monitoring is carried out in the command and monitoring channels.

(2)Monitoring

(a)Voted angle-of-attack (Ref. Fig. 007)

A mean value voted angle-of-attack is obtained from the three information signals. The value is then compared with the input value for each angle-of-attack :
- if the result of the comparison is greater than 3.6° for at least 450 ms, the concerned value is forced to zero and the vote is made on the two remaining probes.
- this probe elimination logic is latched in flight and may be reset if the trim lever is stripped to OFF position and unlatched on the ground (by the landing gear signal) if the probes are in agreement.
  Failure is indicated on the MTP together with the location of the faulty probe.
- at second comparison of latched probes at wrong state :
  . the average angle-of-attack value is forced to zero (but not AOAV).
  The trim disconnects when the incidence trim mode is active.
  . if at least two comparisons are incorrect, flags are positioned :
- general flag of angle-of-attack monitoring defined below

- same flag with a delay of 150 ms.

This is latched in flight and overridden at right state on the ground at the level of the flag.

The flag of the angle-of-attack signals is taken directly to eliminate the faulty probe.

(b) Angle-of-attack voters

Monitoring is performed between the command and monitoring channels after data exchange through 12K line (command to monitoring channel) and 100K (monitoring to command channel) respectively in order to monitor the probe processing modules (elimination and votes). If the result of the comparison is greater than 2° true angle-of-attack for more than 600 ms, the general flags **BFAOA** and **BFAOAD** are set to the wrong state.

This results in :

- losing the trim in incidence trim mode
- transmitting the average angle-of-attack data (but not voted angle-of-attack) with a flag on the status matrix to the **FCC** and **TCC**.

(c)Voted angle-of-attack status matrix generation

Only the angle-of-attack value voted in the command channel is transmitted outside.

Consequently :

- command **AOAV** is delivered with a status matrix depending on the computed **BFAOD** flag
- monitoring **AOAV** is delivered with a status matrix forced to the right state.

The status matrices of the received voted angle-of-attack values are involved in the generation of flag **BFAOA** as they are typical of the bus transmission thus monitored (flag and **No** refresh).

Monitoring is cross coupled :

- **BFAOAM** for flag **BFAOA**, command channel
- **BFAOAC** for flag **BFAOA**, monitoring channel.

(3)Average angle-of-attack values (Ref. Fig. 008, 007)

(a)Computation

Each channel generates an average angle-of-attack signal whose computation is made by using probe inputs previously monitored in the voted angle-of-attack computation phase.

Computation in the command and monitoring channels is as follows :

- no faulty probe : $AVAOA = \frac{AOA + OAOA}{2}$

  (alpha probe 1 + 2)
- if one probe is faulty :

  $AVAOA = \frac{AOA + OAOA + AOA3}{2}$

In fact the concerned probe is forced to zero by the voted angle-of-attack monitoring.

- if two probes are detected faulty by voted angle-of-attack monitoring, **AVAOA** computation is forced to zero.
- on the ground without selecting any probe test position, the result is also set to zero.
- on the ground and in probe test (position 1 + 2 + 3) on the lateral panel, the result is in line with the test values so that alpha-floor can be checked on the ground.

(b)Monitoring

A comparison is made between the result of the computation in the command channel and the result in the monitoring channel (**AVAOAC** and **AVAOAM**). This computation is achieved in the monitoring channel only.

If the deviation is greater than 1.8° true angle-of-attack for more than 2 s, flag **BFWAVAOA** is set to wrong state (flag on average angle-of-attack comparison). This flag is overridden (unconditional right state) in the event of incorrect transmission, through 12K bus line, of signal **AVAOAC** required for comparison (test on **F/W** and **No** refresh) as the comparison is useless.

If flag **BFWAVAOA** which is representative of the **FAC** internal computation is set to zero, **ARINC** 12K bus line is cut by means of signal **BINHARC** after a 1-second delay.

(c)Average angle-of-attack matrices
AVAOAC status matrix depends on flag **BFAOAD** (Ref. paragraph **1.B.**) : it is conditioned by this flag and transmitted.
AVAOAM status matrix is forced to right state as this parameter is not transmitted but only used for comparison.

(4)Angle-of-attack flags
(a)Flag **BFAOA**
1 Significance
State is right if the following conditions are met :
- monitoring between probe voters correct
- transmission flag of opposite voted angle-of-attack correct
- no more than one probe faulty in flight (the case related to two faulty probes is overridden on the ground).

2 Function
The purpose of this flag is to disconnect the trim in incidence trim mode.

(b)Flag **BFAOAD**
1 Significance
Flag **BFAOAD** = flag **BFAOA** + 150 ms delay

2 Function
The status matrix of average angle-of-attack is set to wrong state towards external transmission.

(c)Alpha probe flags
1 Significance
These are the results of the status matrices delivered by the sensors (**ADC1, ADC2** and standby alpha probe).
These flags are :
- **BFAOA OWN** : associated probe flag via associated **ADC**
- **BFOAOA** : opposite probe flag via opposite **ADC**
- **BFAOA3** : standby alpha probe flag (Ref. paragraph below).

2 Function
These flags set the voted angle-of-attack monitoring function to wrong state and thus provide a means of eliminating the faulty probe(s) from the vote.

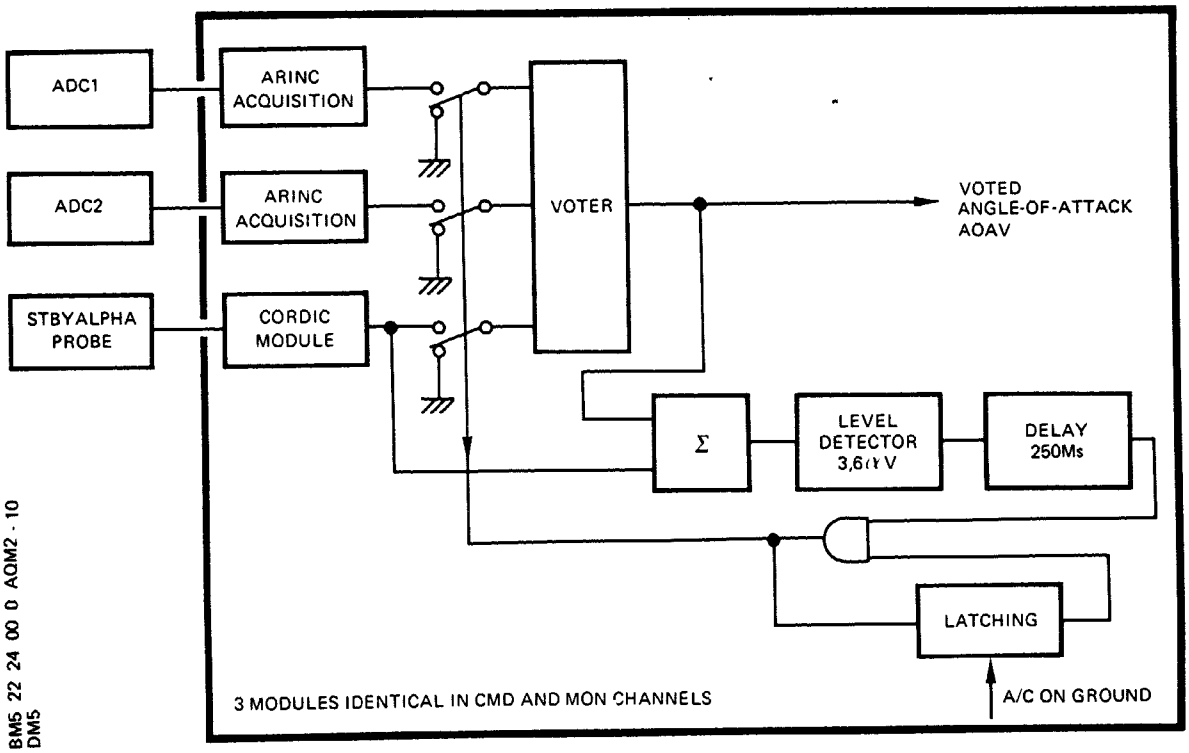
3 Standby alpha probe processing flag
- Significance
The alpha probe is processed and demodulated by a specialized module (**CORDIC** module) which performs a validity check on acquisition :
(sin2 alpha3 + cos2 alpha3 = 1 (at +20%).
When this condition is not met, flag **BFAOA3** is set to wrong state.
- Function
This flag is used in the voted angle-of-attack monitoring function for the elimination of the standby alpha probe from the vote.

(d)Flag **BFAVAOA**
1 Significance : check of average angle-of-attack internal computation.
2Function : 12K **ARINC** line-to-peripherals cutoff.

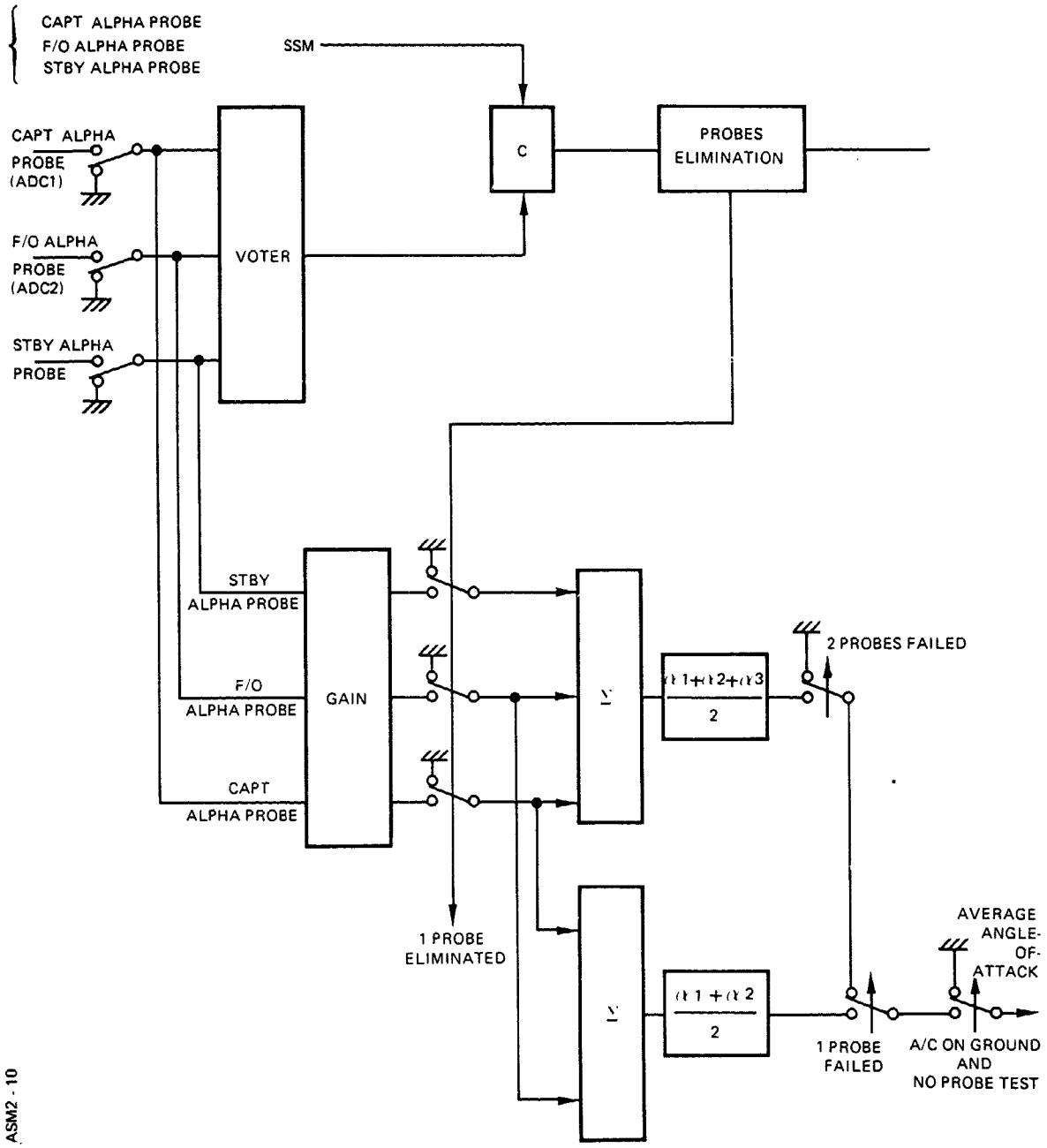
(5)Angle-of-attack processing - summary
(a)One alpha probe faulty
- no consequence
- average angle-of-attack reconfiguration.
(b)Two alpha probes faulty
- in flight : trim loss in incidence trim mode (loss of both trim systems)
flag in average angle-of-attack status matrix transmitted.
- on the ground : override at flag good state. No trim loss. Average angle-of-attack forced to zero without any flag.

Table 1 : Angle-of-Attack Monitoring Functions

SIGNIFICANCE	MONITORING TYPE	FUNCTION
Validation of received angle-of-attack values	Voter comparator between voted input and present input 3.6° threshold 450 ms delay - LATCHED IN FLIGHT - OVERRIDDEN ON GROUND	- Elimination of first probe from computation - Elimination of two probes if required together with : . trim loss in incidence trim . flag on average angle-of-attack sent to FCC/TCC
Validation of voted angle-of-attack computation	Comparator between command and monitoring voters 2° threshold 600 ms delay - NOT LATCHED	- Trim loss in incidence trim - Flag on average angle-of-attack is transmitted to FCC/TCC
Validation of average angle-of-attack computation	Comparator in MON channel only between CMD and MON average angle-of-attack 1.8° threshold 2 s delay NOT LATCHED	12K ARINC line cutoff Total loss of considered FAC through looping
Angle-of-attack acquisition	Status matrix specific to each probe and CORDIC processing monitoring for standby alpha probe	Faulty probe is eliminated from the vote



Voted Angle - of - Attack Monitoring
Figure 007



Average Angle - of - Attack Computation Principle
Figure 008

7. Interconnection between FACs and Actuators

The FACs are connected to the pitch trim and yaw damper actuators in order to execute the computed orders.
Clutching signals as well as slaving principles are dealt with in 22-26 for the yaw damper system and 22-27 for the pitch trim system.
The following table shows the type of the analog signals.
It is to be noted that the clutching signal for the pitch trim actuator wiring (side 1) is validited by the status of the FCC longitudinal torque limiter.
This additional monitoring enables to cover slow undetected trim 1 runaway in auto trim mode.
In this case the FCC will conteract this runaway through a deflection of the elevator until the torque limiter lever is replaced.
Trim 1 is then disconnected. This facility is overriden in gear down not pressed configuration in order to avoid spurious trim disconnection during go-around.

Table 25 : Analog Signals - Trim Actuators to FAC

SIGNAL	TYPE	GRADIENT	RANGE	POSITIVE SENSE	FUNCTION	COMMENTS
TRIM FU1	2-wire synchro	200 mV/° synchro or 600mV/° THS	-18° to +5° THS	In phase with 26VAC = nose-down	TRIM MONITORING TRIM CONTROL	Zero setting : -4.5° THS Out of phase 8.5° ±2°
TRIM FU2	2-wire synchro	200 mV/° synchro or 600mV/° THS	-18° to +5° THS	In phase with 26VAC = nose-down	TRIM MONITORING PITCH TRIM	Zero setting : - 5.1° THS Out of phase 8.5° ±2°
TACHO	2-wire AC	1.3 to 1.6 VAC 3000 rpm	0 to 11000 rpm	In phase with 26VAC = nose-down	TRIM CONTROL	Out of phase 0° ±17°
TRIM MOTOR CONTROL	AC	115VAC 1.2°/s. THS	-15° to +2° THS	In phase with 26VAC = nose-down	TRIM CONTROL	Voltage out of phase by 90° with respect to reference voltage 600 mA RMS

Table 26 : Analog Signals - Yaw Damper Actuator to FAC

SIGNAL	TYPE	GRADIENT	RANGE	POSITIVE SENSE	FUNCTION	COMMENTS
YAW DAMPER FU1	2-wire synchro	533 mV/° variable lever arm unit	±10° control surface	In phase with 26VAC = control surface deflection to the right	Power loop	Gradient depends on the speed according to the variable lever arm unit law. Out of phase 11° ±2°
YAW DAMPER FU2	ditto	ditto	ditto	ditto	Monitor (image loop)	ditto
YAW DAMPER SERVO DRIVE	2-wire DC	0.2 mA/°/s. control surface	±10° control surface	Positive = control surface deflection to right command signal	Yaw damper control	Current generation. Maximum comsumption : 25 mA

8. Interconnection between FACs and Stall Warning System

Each FAC receives stall warning data.

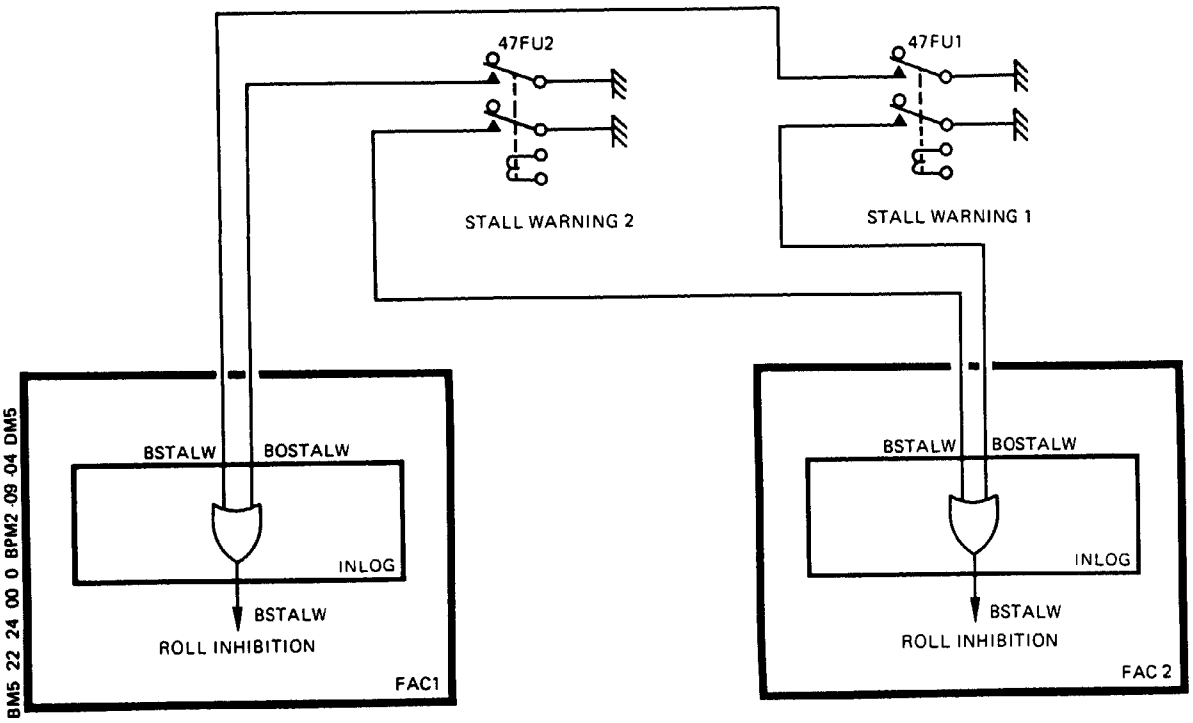
Relays 47FU1 and 47FU2 are controlled by FWC1 and FWC2 (flight warning computer) as soon as the aircraft angle-of-attack is greater than a certain angle-of-attack value.

Type of signal :

- ground : stall warning signal
- open circuit : no stall warning.

The signal is processed through an "OR" gate in each FAC. The resulting signal is used to inhibit the yaw damper roll angle terms.

(Ref. Fig. 009)



FAC - Stall Warning Interface
Figure 009

9. Interconnection Between FACs and EFIS

A. Introduction

The FAC processes the various speed and flight envelope protection data which are displayed in the left part of the PFDs on the speed scale.
Interface is achieved through an EFIS specific data bus which enables word transmission as indicated below.
The FAC data buses are connected to the symbol generator units (SGU) as follows :

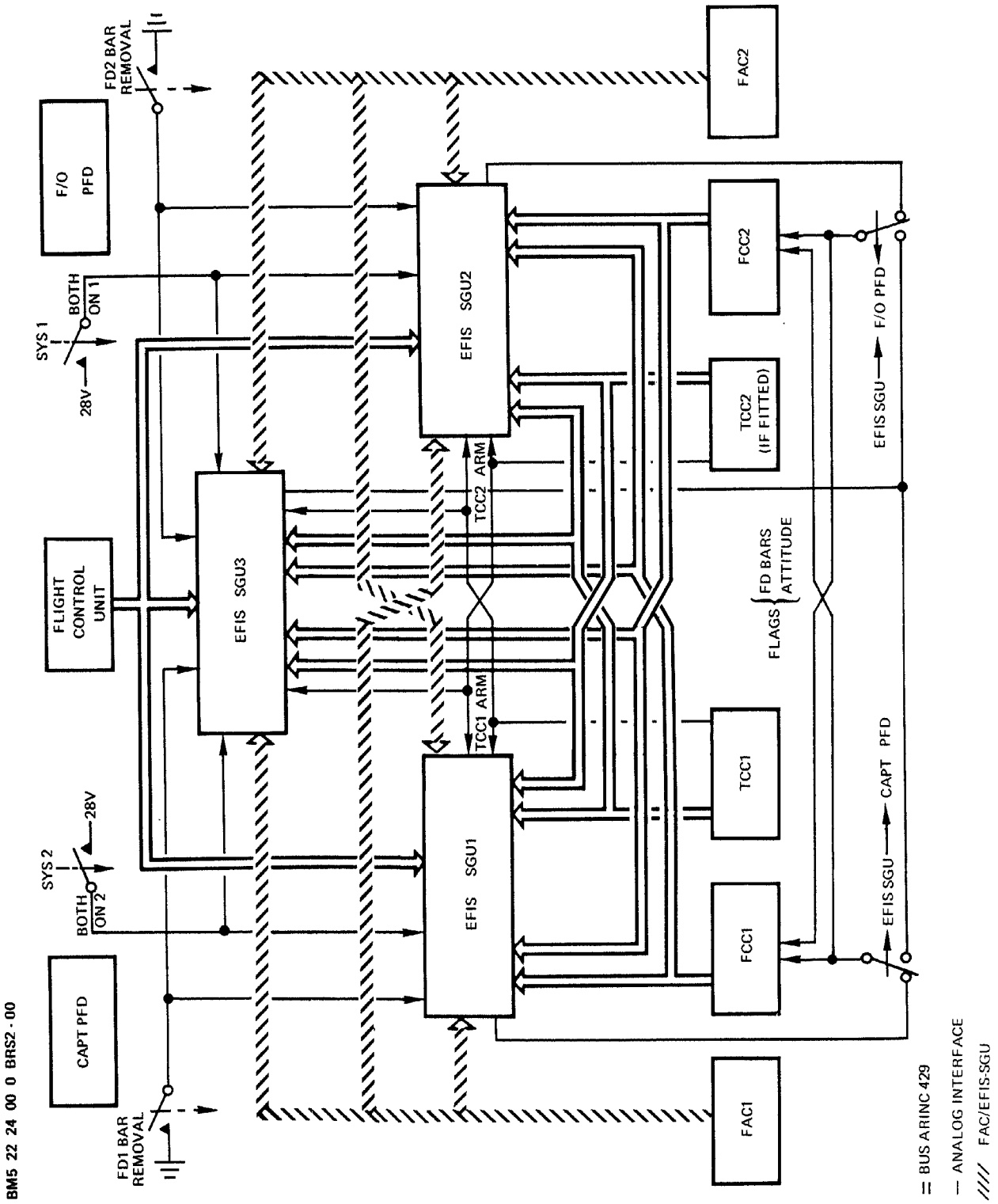
- EFIS SGU1 is fed with data from FAC1 in normal operation or from FAC2 in case of FAC1 failure (selection of FAC2 coupled with ADC2, through Captain ADC INST pushbutton switch located on left instrument panel).
- EFIS SGU2 is fed with data from FAC2 in normal operation or from FAC1 in case of FAC2 failure (selection of FAC1 coupled with ADC1 through F/O ADC INST pushbutton switch located on right instrument panel only if the Captain has not selected ADC2 switching over first).
- EFIS SGU3 is fed with data from FAC1 or FAC2 depending on whether it replaces SGU1 or SGU2.

These symbol generator units ensure :

- data switching to relevant PFD and PFD selection
- symbol display and disappearance logics processed according to configuration data delivered by the FAC on DISCRETES 2 word 274 and according to the status matrix of the words corresponding to the speed scale.

B. Transmission-to-EFIS Monitoring

FAC internal monitoring enables to detect any transmission discrepancy of the ARINC coupler which connects the system to EFIS. Any transmission whose rate exceeds 500 ms corresponds to a coupler inhibition (nominal rate :1,25 ms).
Therefore if the delay between the transmission start and a counter is exceeded, transmission is inhibited. Counter is reset when a new transmission starts.
(Ref. Fig. 010)



AFS - PFD Interface.
Figure 010

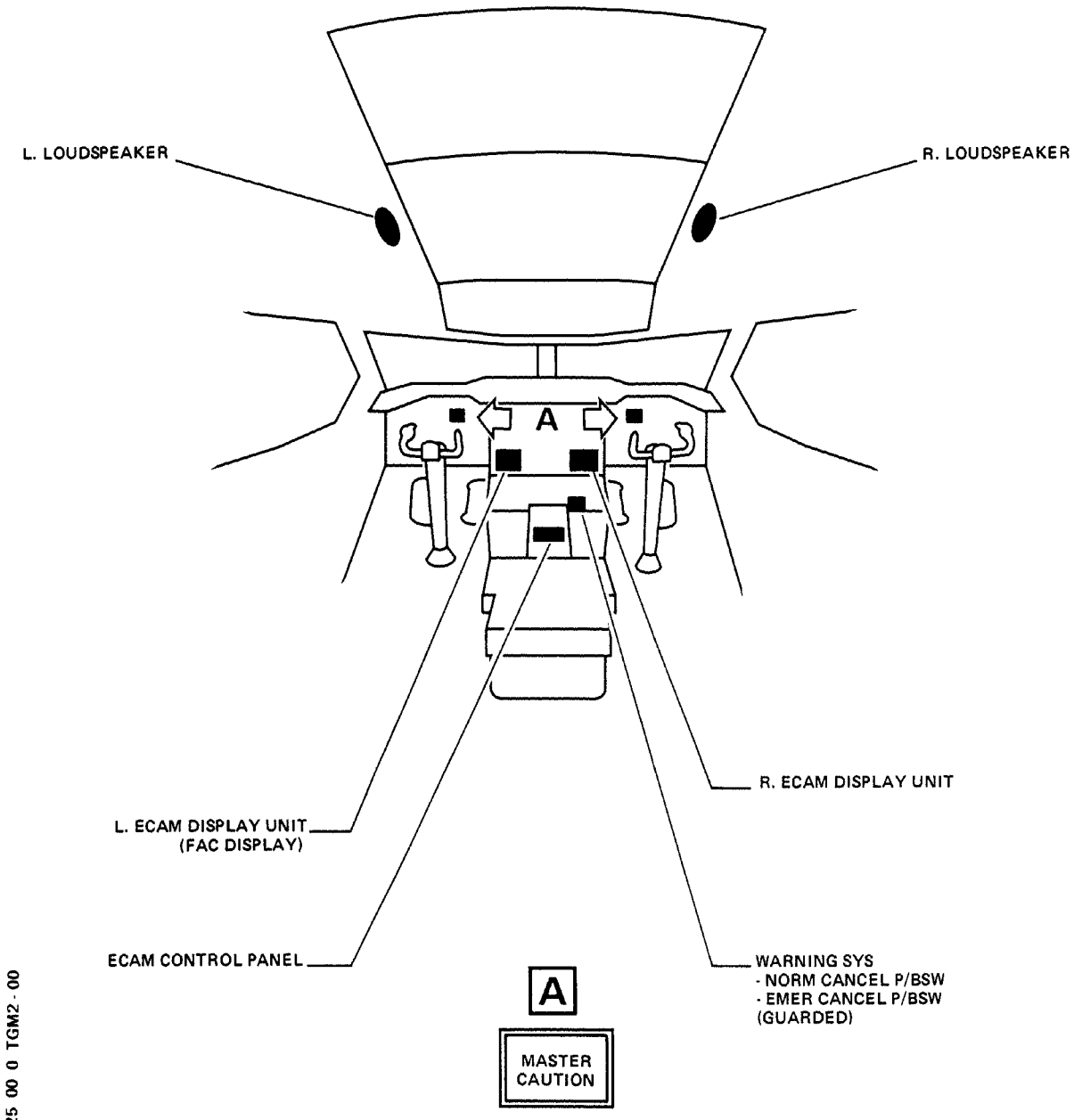
PITCH TRIM AND YAW DAMPER CONTROL UNIT - DISPLAYS, WARNINGS
DESCRIPTION AND OPERATION

1. Warnings
The engage lever position shows engagement status of the system.
Loss of a system is indicated by :
- the engage lever returning to the OFF position
- the ECAM system which generates :
  . an aural warning : 1-second single chime
  . a visual warning on the L ECAM display unit controlled by a SGU (symbol generator unit).

2. Display :
The visual warning is shown in amber and indicates an abnormal situation which does not require an immediate corrective action.
Warnings associated with the FACS are displayed automatically as soon as failures are detected. This warning display can be called manually by the crew.
In the event of a failure of the L ECAM display unit, failures can be displayed on the R ECAM display unit.
The MASTER CAUTION (amber) lights, on CAPT and F/O instruments panel, come on only when both systems (Pitch Trim 1 and 2 or Yaw Damper 1 and 2) are lost.
The following messages can be displayed on L.ECAM display unit :
FLT CTL :
YAW DAMPER 1 (2) (or and 2) OFF
PITCH TRIM 1 (2) (or and 2) OFF
(Ref. Fig. 001)

3. Trim Rotation Aural Warning
Trim rotation is indicated by an aural warning : the "Whooler".
This aural warning is activated upon operation of the pitch trim control switches whatever the THS deflection rate :
- aircraft airborne :
  1- in manual flight,
  2- if the AP engage lever is in CWS position and if the load applied to the control column is greater than the threshold.
- aircraft on the ground : if the AP is not engaged.

CAUTION : WITH AIRCRAFT ON THE GROUND :
IF THE AP ENGAGE LEVER IS IN CWS POSITION WHATEVER THE LOAD ON THE CONTROL COLUMN, THE TRIM ROTATES WITHOUT THE "WHOOLER" BEING ACTIVATED.



FAC Warning Indicating - Location in Flight Compartment
Figure 001

1. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Cart - Hydraulic Ground Power Supply
C.	Blanking Caps
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-22-00, P. Block 301	Blue Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

R **ON A/C 001-100,

(Ref. Fig. 401)

R **ON A/C 101-200,

(Ref. Fig. 402)

R **ON A/C ALL

2. Procedure

A. Job Set-Up
(1)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC2/28VDC	306CC2	103/G12

R **ON A/C 101-200,

21VU	AFS/TCC/SYS 1/28VDC	306CJ1	102/H 8

R **ON A/C 001-100,

21VU	AFS/TCC/SYS 1/28VDC	306CJ1	102/H8
21VU	AFS/TCC/SYS 2/28VDC	306CJ2	102/H12

**ON A/C ALL

132VU INTEGRAL LT/C/B &/OVHD/& PED 3LF 322/N77

B. Removal
(1)Unlock the four quick-release fasteners (5) on FAC/ATS engage unit (4).
(2)Withdraw FAC/ATS engage unit (4) from its housing (1).
(3)Disconnect and cap electrical connectors (2) and (3).
(4)Remove FAC/ATS engage unit (4).

C. Installation
(1)Clean and inspect FAC/ATS engage unit interface and/or adjacent area.
(2)Remove blanking caps from electrical connectors (2) and (3).
(3)Connect electrical connectors (2) and (3).
(4)Position FAC/ATS engage unit (4) in its housing (1).
(5)Lock quick-release fasteners (5).

**ON A/C 101-200,

(6)Remove safety clips and tags and close circuit breakers 306CC1, 306CC2, 3LF and 306CJ1.

**ON A/C 001-100,

(6)Remove safety clips and tags and close circuit breakers 306CC1, 306CC2, 3LF, 306CJ1 and 306CJ2.

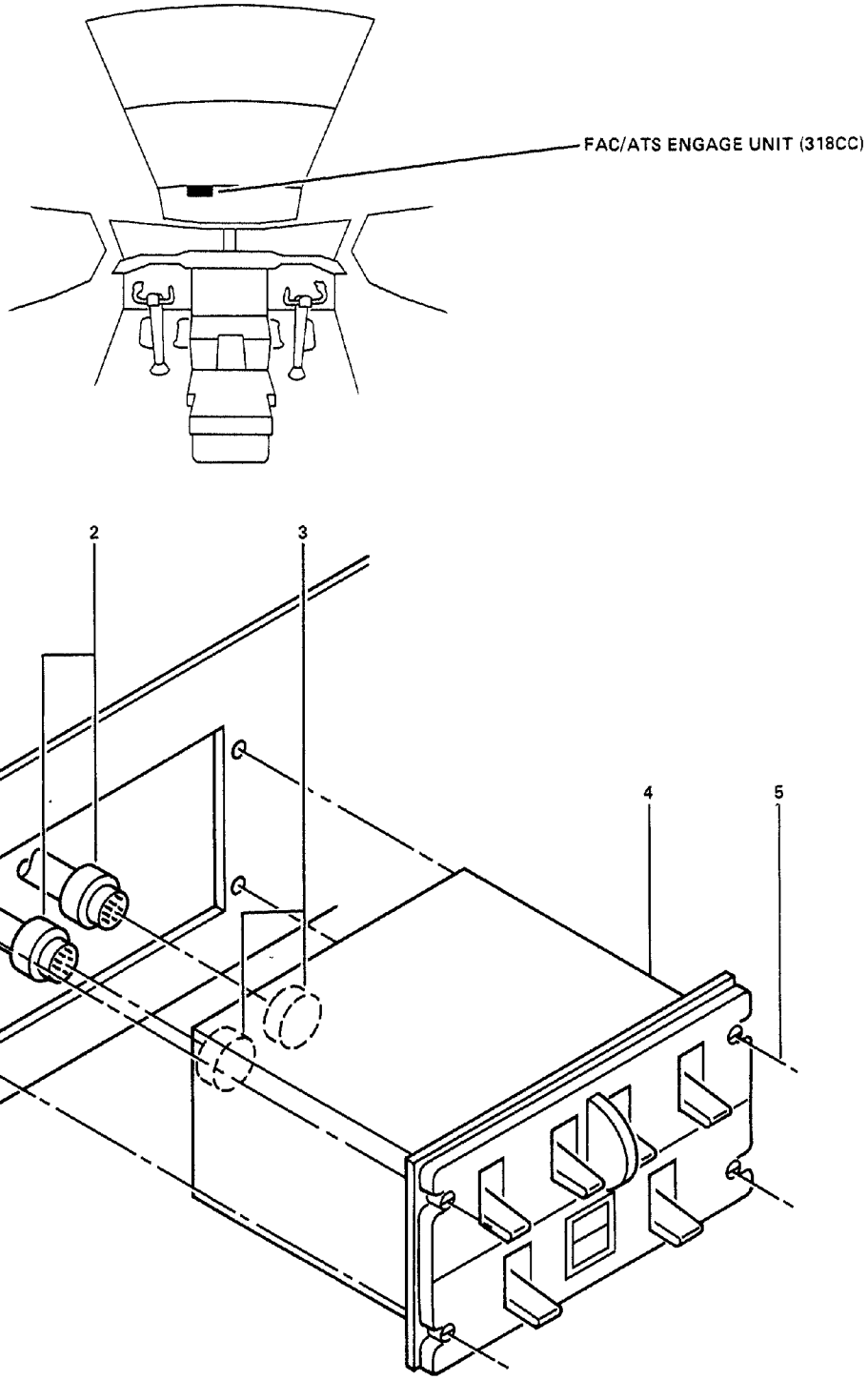
**ON A/C ALL

D. Test
Initial conditions :
- correct electrical power supply
- forward services ventilation system in operation
- hydraulic power available (BLUE and YELLOW)

NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-22-00, 29-23-00, P. Block 301).

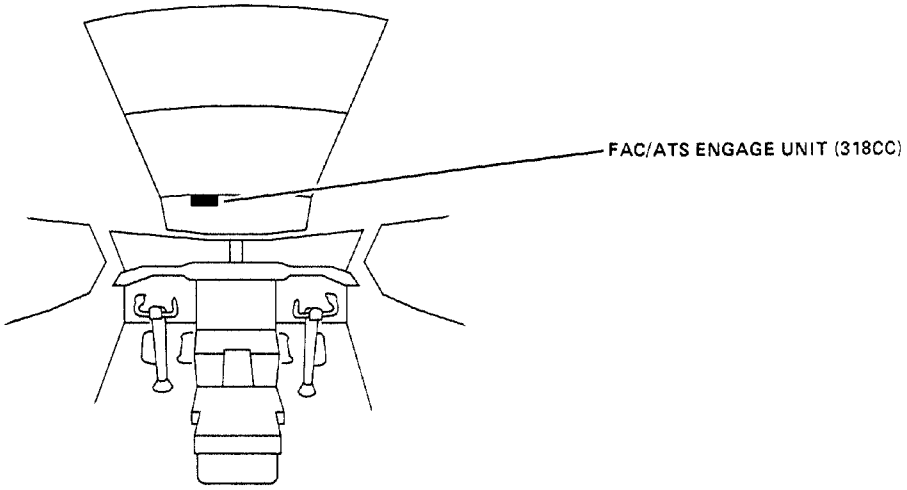
- overhead panel circuit breakers closed, particularly those associated with AFS, NAVIGATION, ADS and EFCU (panel 133VU).
- aircraft in ground configuration (L/G PROX DET & RELAYS circuit breakers on panel 133VU closed).
- AP not engaged
- ADS operating
- EFIS operating and adjustment, on panels 412VU and 413VU, of PFD image brightness.
- IRS 1, 2 and 3 aligned (perform ALIGN procedure until red ATT on PFDs disappear ; present position entering is not necessary).

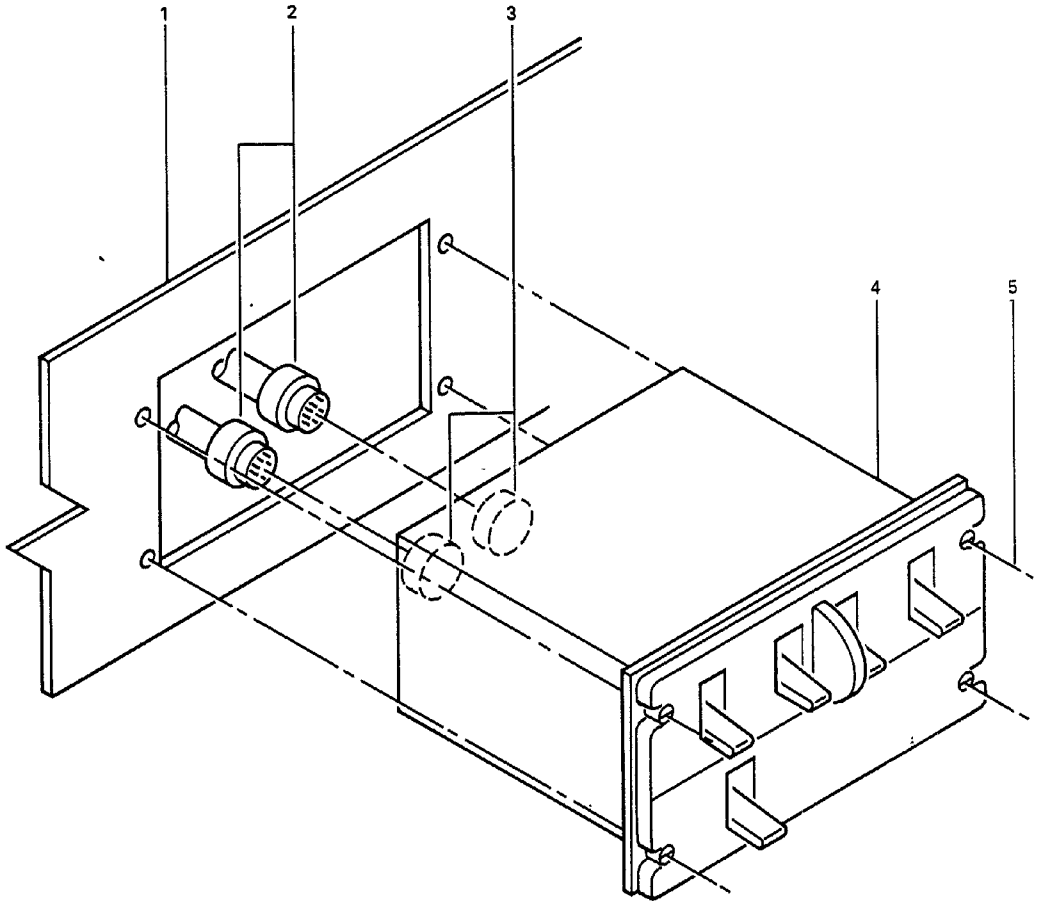
R NOTE : After FAC 1(2) electrical power supplying or after a FAC 1(2) reset
R longer than 2 seconds through circuit breaker G03 (G12) on panel



BM5 22 25 12 4 AAM2-00

FAC/ATS Engage Unit
Figure 401





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FAC/ATS Engage Unit
Figure 402

R 21VU, power up test of computer will restart with corresponding
R SPD LIM fault flag displayed for 1 minute. Wait 30 seconds after
R this flag has disappeared before engaging pitch trim or yaw damper
R levers.

ACTION	RESULT
**ON A/C 101-200,	
1. On overhead panel	
- place PITCH TRIM 1 lever in ON position	Lever remains in this position
- place YAW DAMPER 1 and 2, and ATS1 lever in ON position	Levers remain in this position
- place PITCH TRIM 1 lever in OFF position	After 12.5 seconds, YAW DAMPER 1 and 2 and ATS 1 levers return to OFF position
- place PITCH TRIM 2 lever in ON position	Lever remains in this position.
**ON A/C 001-100,	
1. On overhead panel	
- place PITCH TRIM 1 lever in ON position	Lever remains in this position
- place YAW DAMPER 1 and 2, ATS 1 and 2 levers in ON position	Levers remain in this position
- place PITCH TRIM 1 lever in OFF position	After 12.5 seconds, YAW DAMPER 1 and 2, ATS 1 and 2 levers return to OFF position
- place PITCH TRIM 2 lever in ON position	Lever remains in this position.
**ON A/C ALL	
2. On panel 21VU	
- open AFS/FAC 2/28 VDC circuit breaker 306CC2 (location 103/G12)	PITCH TRIM 2 lever returns to OFF position.
- close circuit breaker 306CC2.	

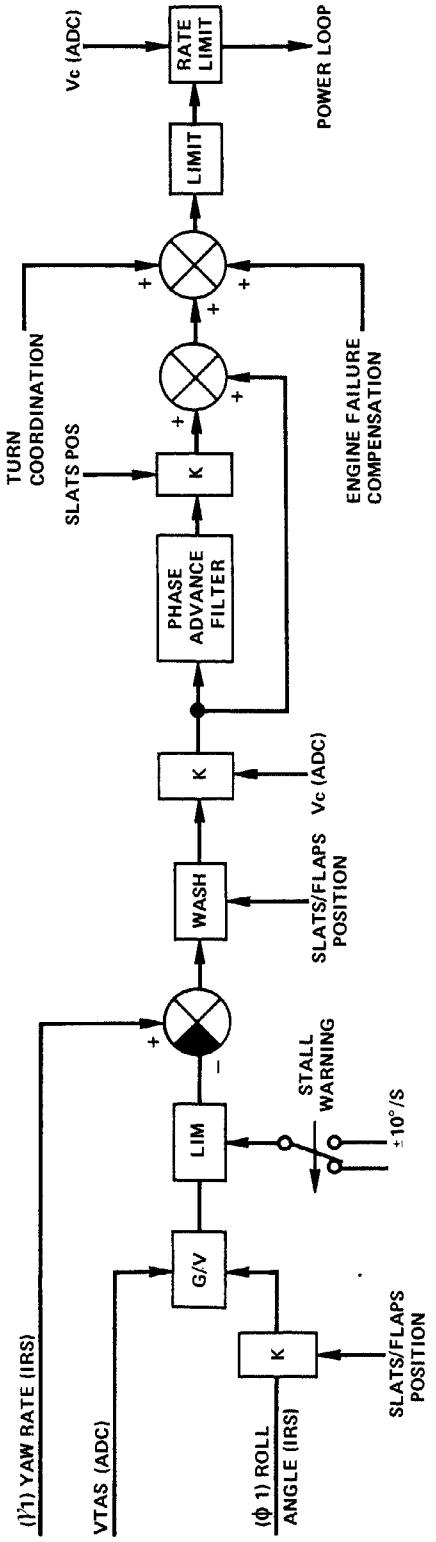
E. Close-Up
(1)Shut down and disconnect hydraulic ground power cart.
(2)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(3)Restore aircraft and system to normal operating conditions.

YAW DAMPER - DESCRIPTION AND OPERATION

1. General (Ref. Fig. 002)
The FACS includes a yaw damper sub-system which fulfills the following functions :
- Dutch roll damping throughout the flight envelope
- aid to lateral piloting through turn coordination
- assistance in counteracting effects of an engine failure.

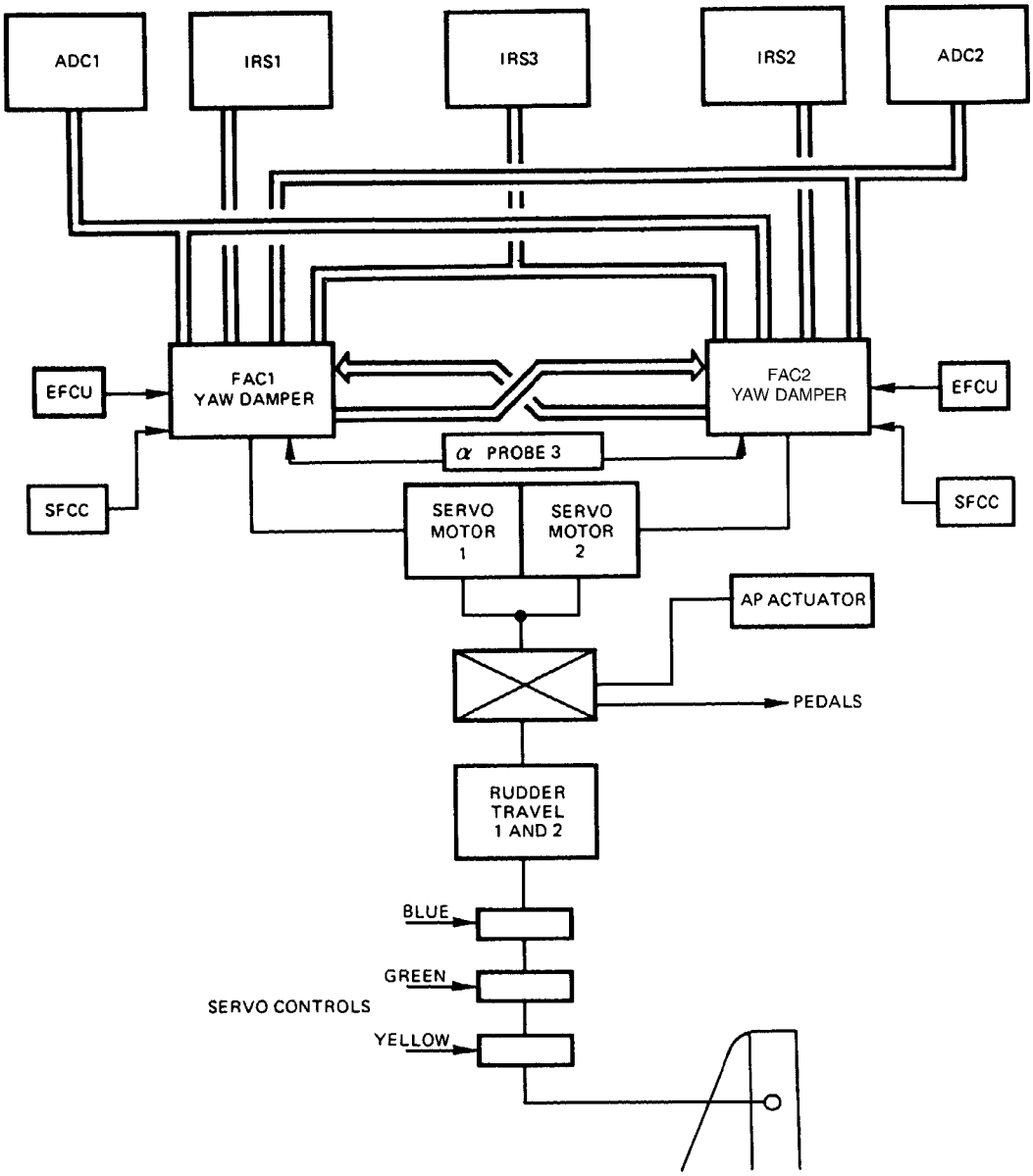
2. Dutch Roll Damping (Ref. Fig. 001)
The lateral flight qualities of the aircraft can be downgraded by the Dutch roll effects (yawing oscillations leading to a roll-yaw instability).
The main function of the yaw damper consists in providing enough oscillation damping throughout the flight envelope.
To this end, the yaw damper counteracts the yawing effect.
Damping can be performed by taking the yaw rate, easier to detect, as a slaving parameter.
As a yawing variation is indicated by a yaw rate variation, the damping effect can be reduced by modifying the gain of the yaw rate loop.
An additional term has been introduced in order to take into account the yawing mode stabilization in stabilized flight.
Limitations in speed and amplitude have been included in the law for safety reasons (limited runaways) :
- limitation in total amplitude at $\pm 10^{\circ}$ of rudder at low speed as a function of the corrected speed (2° only at high speeds)
- limitation in total speed at $\pm 39^{\circ}/s$ so that the actuator does not reach saturation level.

An analog filter has been added to the yaw damper loop in order to filter the sampling problems due to digital/analog conversion.
During stall warning detection, the phi term is set to zero so that the recovery maneuver is not hindered. This term is also cancelled in APCMD if the flap and slat control lever position is less than 4.
The Dutch roll damping function is used throughout the flight envelope, in manual and automatic operation.



BM5 22 26 00 0 AAM2 17 12
DM5

Dutch Roll Damping
Figure 001



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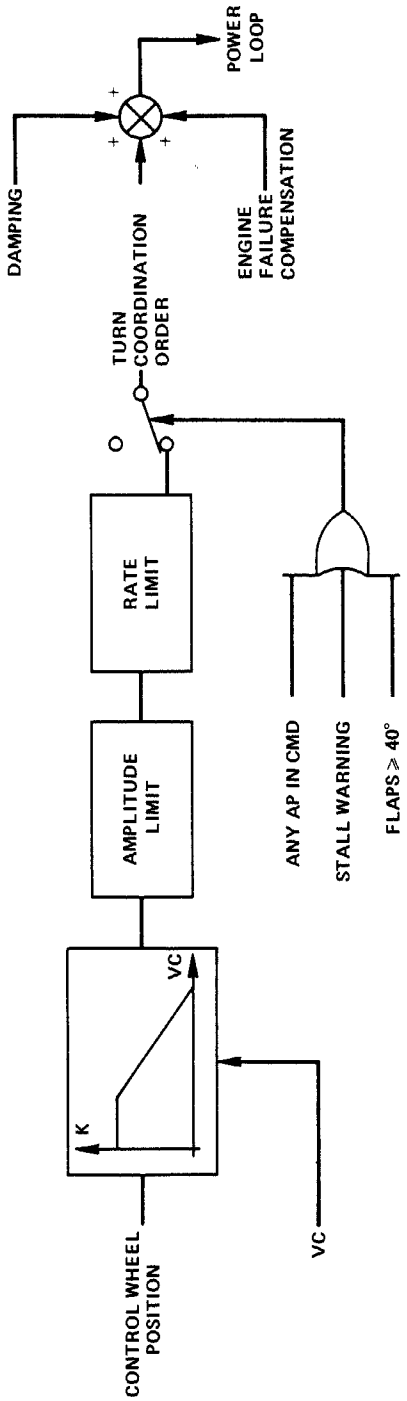
Yaw Damper - Block Diagram
Figure 002

R

3. Turn Coordination (Ref. Fig. 003)
The turn coordination function provides a means of assisting the aircraft lateral control and increasing the pilot comfort.
This function cancels out the induced sideslip when going into or coming out of a turn by transmitting an appropriate deflection command to the rudder.
This deflection command is proportional to the command performed by the pilot on the roll axis thus ensuring coordination between roll and yaw actions.
This function is used in manual flight only as the autopilot fulfills its own coordination function.
The coordination gain is contingent on speed.
Limitations have been included for safety reasons :
- limitation in amplitude at ±8° of rudder deflection
- limitation in speed at 5°/s of rudder deflection.
The coordination term is set to zero when in stall warning condition and in ≥30/40 configuration.

4. Engine Failure Automatic Compensation Function (Ref. Fig. 004)
This function reduces the sideslip induced by an engine failure resulting in thrust dissymmetry.
This function consists in providing a rudder deflection command which is already ensured by the yaw AP but with a limited transient authority.
The deflection command ensured by the yaw damper is added to the AP deflection command.
The induced sideslip results in a lateral acceleration which is the parameter cancelled by the yaw damper.
Any possible noise of the signal is eliminated by a low-pass filter.
A threshold is integrated in the channel so that an uncoordinated turn or a turbulence condition does not result in untimely triggering of the function.
This function is initiated in AP engaged in CMD, in go around and takeoff phases.



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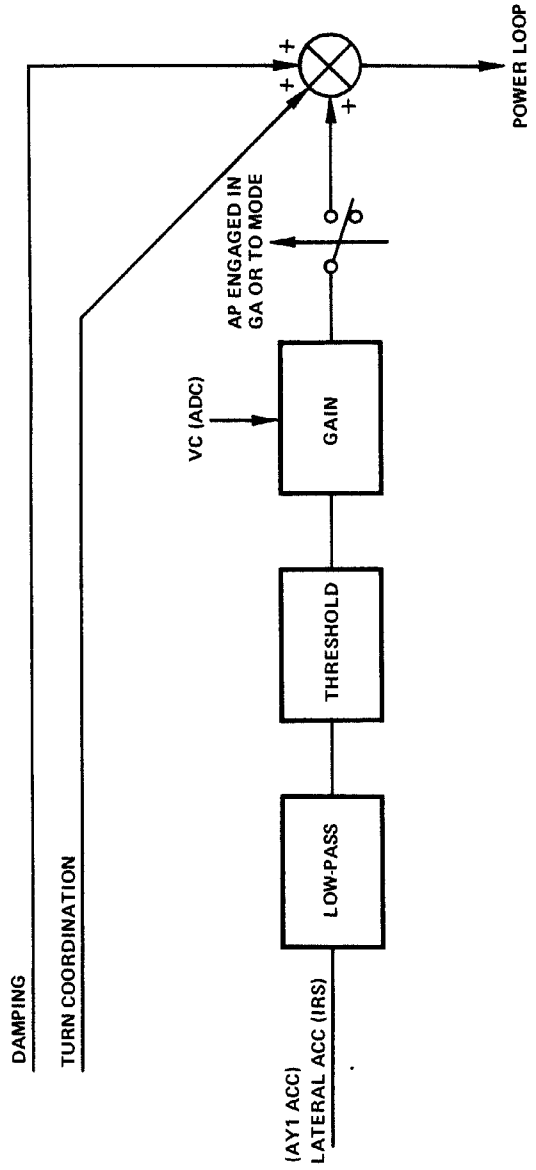
DM5

Turn Coordination
Figure 003



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DM5

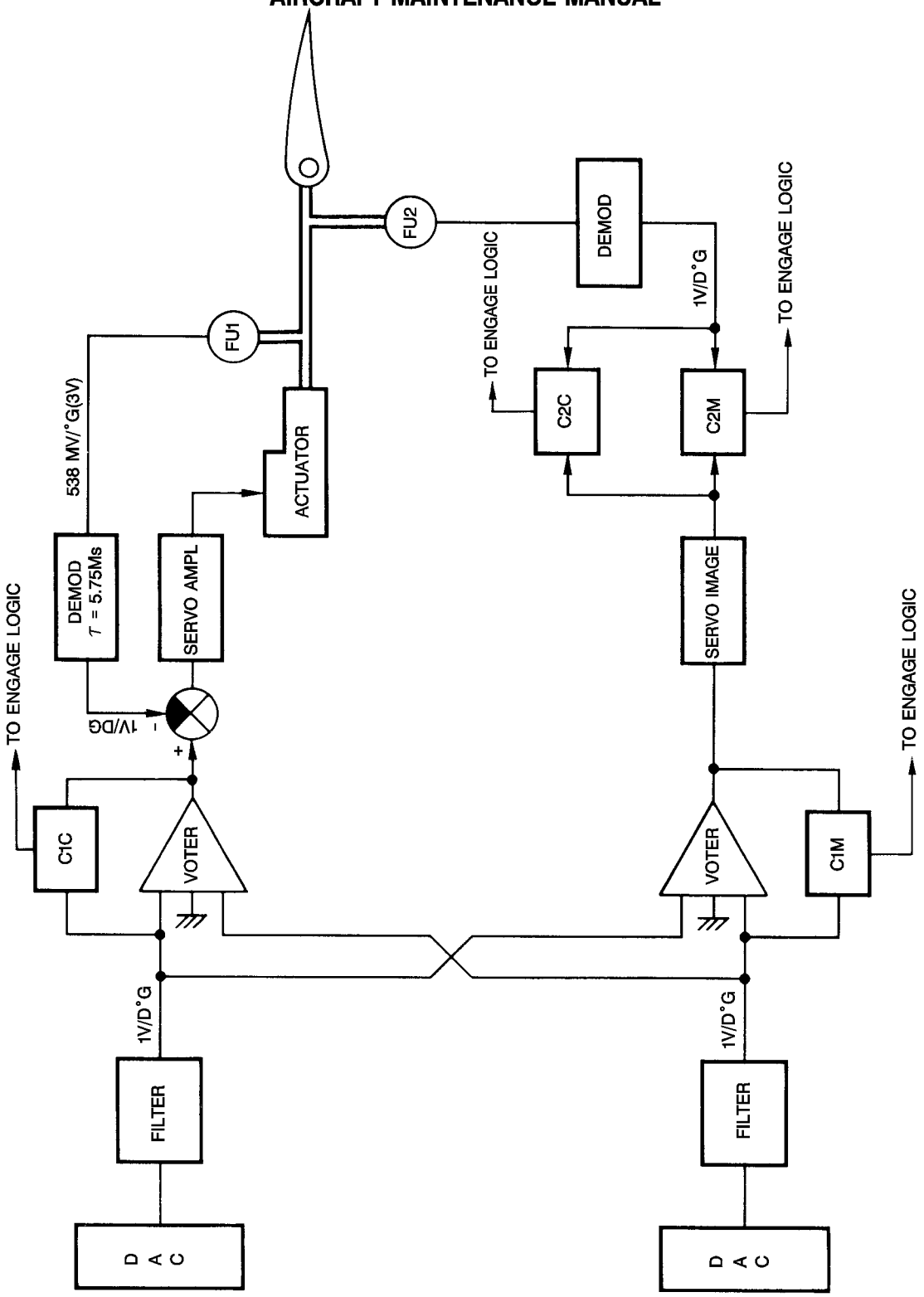
Engine Failure Compensation
Figure 004

5. Processing of the Three Function Signals
The signals resulting from the three functions described above are summed and processed on the same computation channel.

A. Authority Limit
The maximum command signal generated by the yaw damper is restricted to 10° of rudder deflection. This value has been determined as a trade-off between the maximum deflections required by the system performance on one hand and the safety problems caused by a rudder runaway on the other hand.
According to the rudder travel limiting system, the yaw damper command is limited as follows :
- 10° at low speeds (Vc ≤ 165 kt)
- $10 \times \left(\frac{165}{Vc}\right)^2$ of control surface for Vc > 165 kt

B. Speed Limit
The yaw damper command is limited to 39°/s with respect to the maximum speed of the actuator displacement which is 42°/s in order to protect the actuator against saturation (possible velocity error).

6. Power Loop (Ref. Fig. 005)
The power loop section enables servomotor control current to be generated in response to the voted deflection command.
This is achieved by a current amplifier enabling matching to the control of the servovalve located in the servomotor.
Position of the servomotor jack is transmitted by a synchro feedback signal which enables the error signal to be cancelled.
The analog input control signal to the servomotor is never removed and the servomotor is neutralized through hydraulic power cut-off.
Engagement is not possible unless it is undertaken under stable flight conditions.
In fact, slaving may not be achieved within the predetermined time (1 s) although all logic conditions are met.



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Yaw Damper Power Loop - Principle
Figure 005

R

7. Yaw Damper Engage Logic

A. General

Hold of the engage lever in the ON position depends on the validity of the command and monitoring logics.
It is to be noted that the electrovalves are powered and that the servomotor is pressurized if :
- command and monitoring logics are validated,
- the engage lever is in the ON position,
- the self-holding relay is energized.

System disconnection results in depressurization of the associated servomotor jack due to cut-off of the electrovalves and in return of the engage lever to the OFF position.

B. Engage Logic

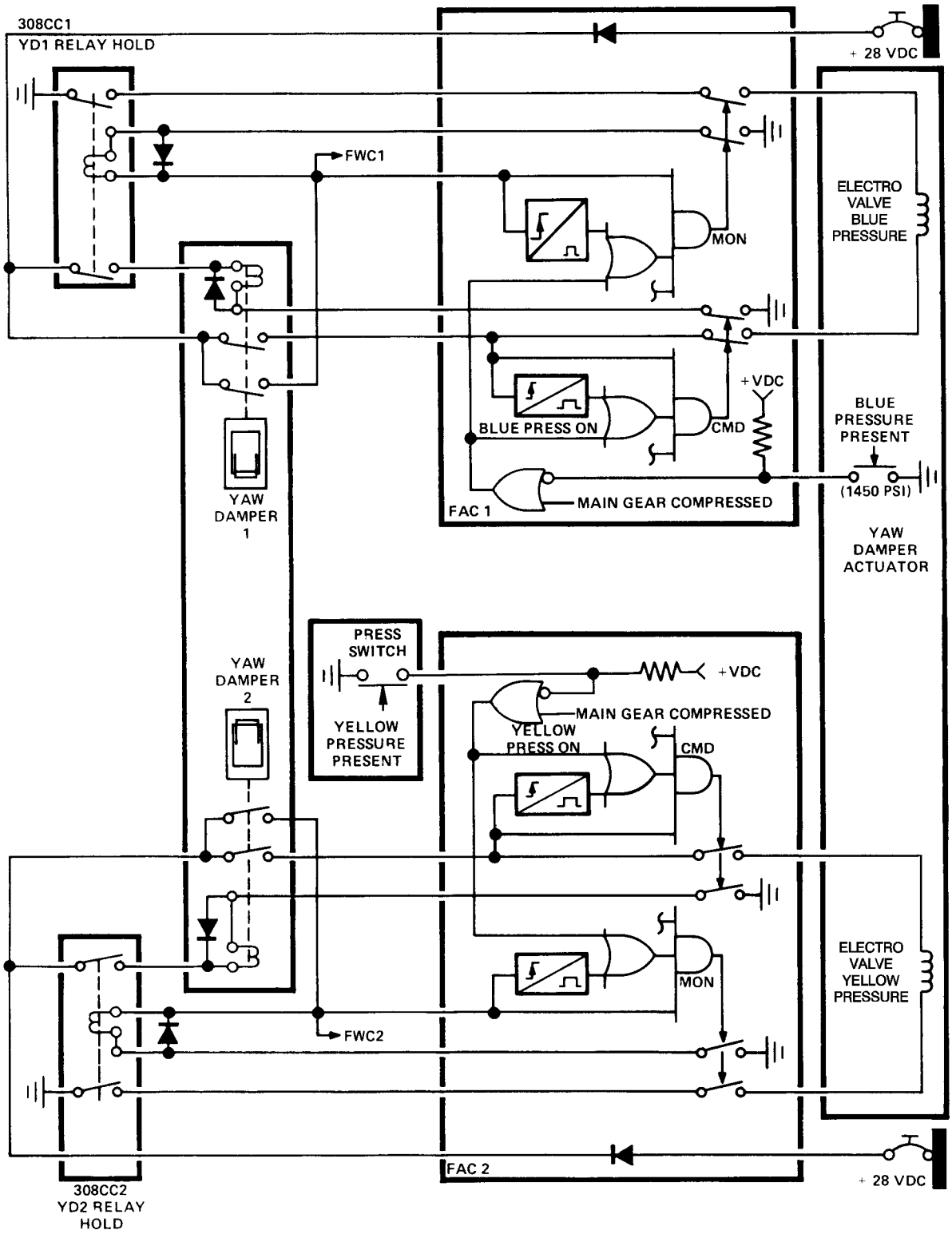
The yaw damper engage logic is shown in figure (Ref. Fig. 006).
The engage circuit is composed of :
- an engage lever for engagement selection (engage lever in ON position) and taking into account the engage logic reply in the command channel
- a self-holding relay 308CC installed in the aircraft wiring (electronics rack side). This relay ensures the engage logic reply in the monitoring channel as well as the actuator clutch line interruption.
- command and monitoring hard-wired logic ensuring the validation of the lever engagement and actuator clutch.

It is to be noted that the command or monitoring engage logic maintains the system engaged if the following conditions are fulfilled :
- voter comparator healthy
- power comparator healthy
- command and monitoring watchdog healthy
- mode logic correct
- engagement command (enables to override the power loop for 1 s)
- oscillation detector at right state.
- no permanent override of comparators with power cut.

A delay of 100 ms is inserted and if all the command and monitoring conditions are met, the following occurs :
- the engage lever on FAC/ATS engage unit 318CC is self-held
- the self-holding relay 308CC is energized
- the actuator clutch is energized.

All these conditions are accomplished twice :
- +28 V command through the command channel
- command by the ground through the monitoring channel.

Detection of a failure of one condition results in :
- the engage lever returning to the OFF position
- depressurization of the actuator through opening of the electrovalve even if the lever is not in the OFF position (mechanical jamming).



Engage Logic Principle
Figure 006

R

C. Yaw Damper Monitoring Logic

The yaw damper monitoring logic consists of :

- parameter specific monitoring functions (yaw rate monitoring through comparison ; average angle-of-attack monitoring) in monitoring side only.
- parameter conventional monitoring functions (status matrix monitoring)
- FAC general monitoring functions (power supply, software, etc.)
- monitoring functions specific to the yaw damper modes.

The monitoring logic is processed in software form.

It is to be noted that during 1 s after engagement command, the power loop conditions are overridden.

The following conditions are required by engagement :

- engagement command present
- hydraulic power correct at least 2 s when A/C is airborne (Ref. Fig. 007)
- transmission of discrete word 2 correct
- FAC main logic correct
- yaw damper safety test correct
- Vc valid except in the following configuration : flap and slat control lever position ≥ 3
- yaw rate valid (IRS)
- bank angle valid
- TAS valid in flight except in the following configuration : flap and slat control lever position ≥ 3
- turn coordination mode logic healthy
- engine failure compensation mode logic healthy
- yaw rate comparison correct
- average angle-of-attack computation monitoring correct in monitoring channel only (Ref. 22-24).

All these conditions must be fulfilled to generate a software signal which will supply the hard-wired logic.

This logic is duplicated (command and monitoring channel).

D. Yaw Rate Signal Monitoring

The yaw rate signal is the fundamental parameter of the yaw damper system.

The sensor which is used (IRS) does not provide the self-monitoring level necessary to the r signal (yaw rate signal). Consequently a monitoring sequence is performed by the FAC based on a comparison between three information signals delivered respectively by the associated IRS, opposite IRS and IRS3 which is used for double checking.

Monitoring is performed by the command channel only as shown in figures (Ref. Fig. 008).

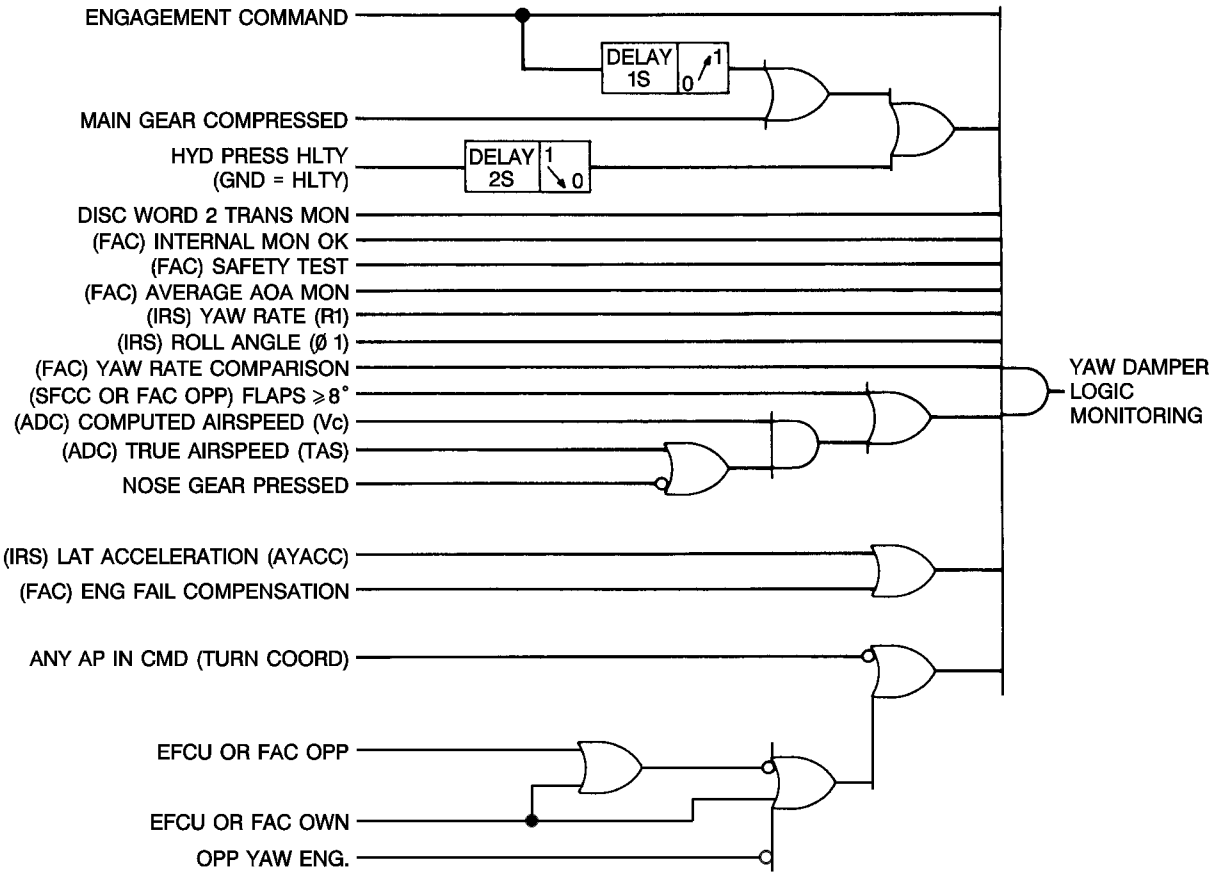
The command channel receives :

- associated yaw rate 1 (OWN) through the bus of associated IRS
- opposite yaw rate 1 through the bus of opposite FAC
- double-checked yaw rate 1 through the FAC internal bus as IRS3 data signal is processed in the monitoring channel.

A two-out-of-three data vote is performed by three modules comparing each input with the voted output.

The voted output is obtained by taking the mid value of the three values presented at the input.

When the deviation between this voted value and the input value exceeds



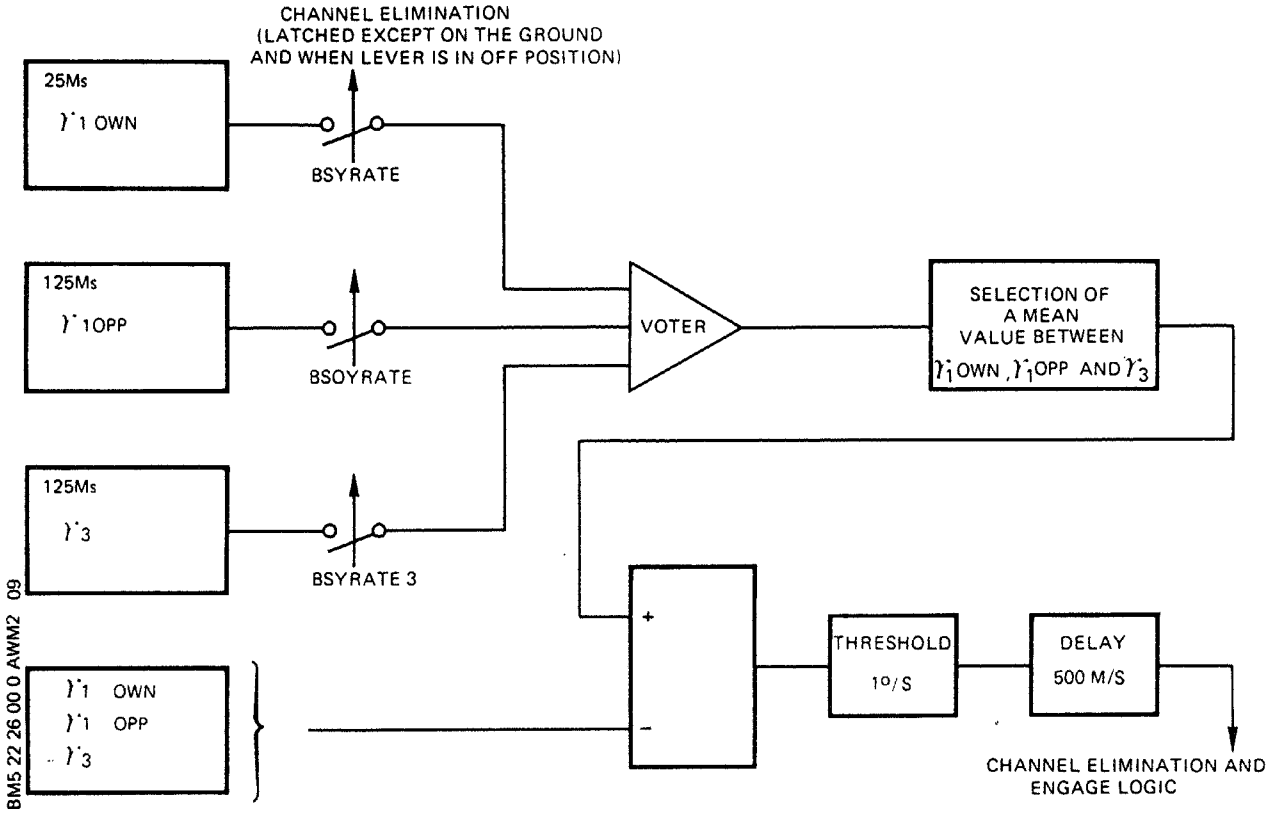
BM5 22 26 00 0 ASW2 00

Yaw Damper Monitoring Logic
Figure 007

a predetermined threshold (1°/s), for 500 ms a monitoring logic is performed :

- if associated yaw rate 1 parameter is concerned, the yaw damper is disengaged throughout the flight envelope.
- if opposite yaw rate 1 or yaw rate 3 is concerned, the relevant channel is eliminated from the vote. The system does not disconnect but a capability down-grading can occur.
- if a second source is concerned after previous elimination of a channel (opposite yaw rate or yaw rate 3) the system (FAC1 and FAC2) disconnects throughout the flight envelope.

Monitoring concerns the yaw rate digital values computed in the same cycle and the status matrix (NCD, F/W and No refresh).
Monitoring is latched in flight only.



Yaw Rate Monitoring - Principle
Figure 008

E. Monitoring of Other Parameters

(1)Roll angle signal monitoring

The roll angle parameter is used in the Dutch roll damping and turn coordination functions.

The parameter status matrix is checked (flag, NCD and No refresh) which results in the yaw damper loss throughout the flight envelope.

(2)Lateral acceleration monitoring

The lateral acceleration parameter is used in the engine failure compensation function.

The parameter status matrix is checked (flag, NCD and No refresh) which results in the yaw damper loss when the engine failure compensation function is active (AP in CMD in SRS mode takeoff or landing).

(3)Control wheel deflection monitoring

The control wheel deflection parameter is used in the turn coordination function. The parameter status matrix is checked (flag, NCD and No refresh) which can result in the yaw damper loss, in manual flight only (or CWS).

And if a reconfiguration is possible (opposite YAW engaged) the logic used is as follows :

- 1 EFCU failure only = loss of own FAC
- 2 EFCUs failed = no loss of FAC but turn coordination function inhibited.

Validity status of the EFCUs is exchanged between FACs by means of word 274 :

bit 22 = 0 loss of own EFCU

bit 23 = 0 loss of 2 EFCUs.

(4)Monitoring of ADC data utilized by the yaw damper

(a)Indicated airspeed (IAS)

This term is used for the yaw damper gain adaptation.

Its status matrix is checked (flag and No refresh). It is to be noted that the NCD is not taken into account in order to enable yaw damper engagement on the ground (as long as Vc < 30 kt, the ADC system transmits an NCD signal).

If the status matrix is not correct, the yaw damper is disengaged except in flaps extented configuration (flap and slat control lever position ≥ 3).

In the case of associated ADC detected failure on Vc, the different gains used by the yaw Damper are forced to fixed gains (low speed gains).

Furthermore, the value used for the gain computation is issued from a vote between Vc (ADC1 and ADC2) and 190kt in clean configuration or Vc (ADC1 and ADC2) and 0kt when slats are extended.

(Ref. Fig. 009)

(b)True airspeed (TAS)

This parameter is used in the roll angle term gain adaptation whose purpose is to counteract the effect of the stabilized turn (g/V phi).

The parameter status matrix is checked (flag, NCD and No refresh).

In the event of non validity condition, the yaw damper is lost except on the ground where the parameter is not significant, and also in flaps extended configuration (flap and slat control lever position ≥ 3).

In the event of TAS failure in this phase, the gain using this

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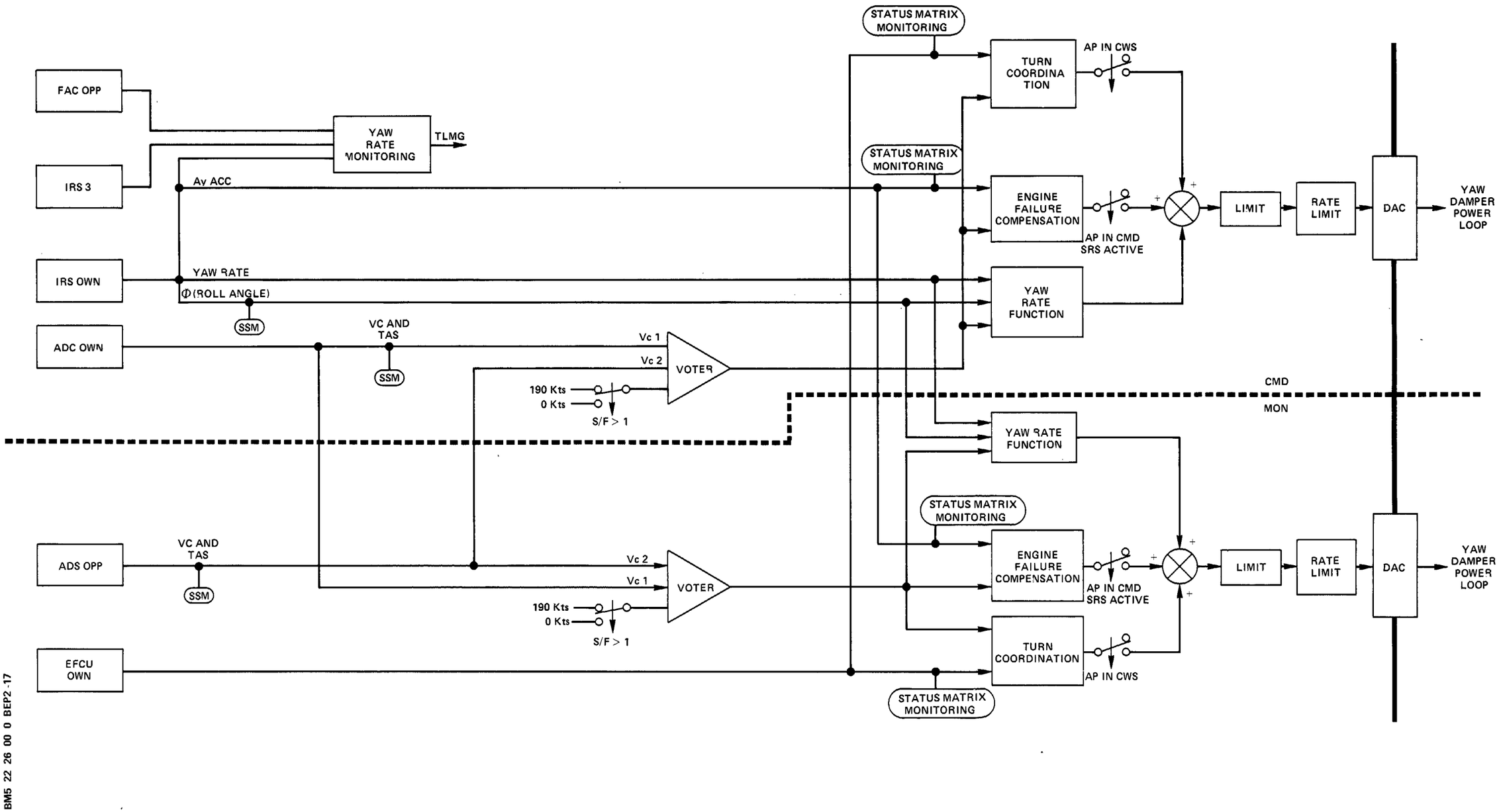

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Yaw Damper Monitoring
Figure 009

parameter is forced to fixed value (low speed gain).

Table 1 : Yaw Damper - Signal Processing

PARAMETER	ACQUISITION	MONITORING	VALIDITY
Direct yaw rate	Command channel through IRS own bus	- Flag - No refresh - NCD	- Throughout flight envelope - De-activated on ground and with lever in OFF position
Opposite yaw rate	Command channel through opposite FAC bus acquisition	- Two by two monitoring + flag, NCD No refresh	- Throughout flight envelope - De-activated on ground and with lever in OFF position
Double-checked yaw rate (IRS3)	- Monitoring channel - Retransmitted to command channel through 100 K internal bus	- Two by two monitoring + flag, NCD No refresh	- Throughout flight envelope - De-activated on ground and with lever in OFF position
Roll angle	Command channel through IRS OWN bus	- Flag - No refresh - NCD	Throughout flight envelope
Lateral acceleration	Command channel through IRS OWN bus	- Flag - No refresh - NCD	AP in CMD and SRS active
Indicated airspeed (IAS)	Monitoring channel and wiring to command channel which also processes ARINC bus acquisition	- Flag - No refresh - Vote	Throughout flight envelope except if flap and slat ctl lever pos ≥ 3.
True airspeed (TAS)	Monitoring channel through ADS bus also wired for command channel	- Flag - NCD - No refresh	Throughout flight envelope except on ground (5 s after shock absorber extension and if flap and slat control lever position ≥3).
Control wheel deflection	- Monitoring channel - Retransmission to command channel through 100 K internal bus	- Flag - NCD - No refresh	Manual and CWS

COMPARATOR	TYPE	PHYSICAL THRESHOLD	THRESHOLD (VOLTS)	DELAY (ms)	CONDITION OF OPERATION	COMMENTS
C 1	M	1.92° control surface	1.92	2000	Permanent	
C 2	M	2.08° control surface	2.08	100	Permanent	

1. Operational Test

A. Test of Mechanical Linkage between Yaw Damper Actuator and Rudder

(1)Reason for the job

To check correct control of rudder with YAW DAMPER 2 only engaged, then, without any YAW DAMPER engaged.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 34-25-00, P. Block 501	Inertial Reference System
- 29-22-00, P. Block 301	Blue Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

R - energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (BLUE, YELLOW)
R (Ref. 29-10-00, P. Block 301 ; 29-22-00, P. Block 301 ; 29-23-00, P. Block 301).
NOTE : As an alternate it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUS.
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION, ECAM and ADS.
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that ADS is operating
- align IRS1,2 and 3 (perform ALIGN procedure until red ATT flags on PFDs disappear; present position entering is not necessary) (Ref. 34-25-00, P. Block 501)
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that AFS components are energized (PITCH TRIM 1 engaged)
- on center pedestal, adjust ECAM display unit brightness and select FLT CTL page.

(b)Test

ACTION	RESULT
(1)Place **YAW DAMPER** 2 lever in **ON** position	Lever remains in engaged position,
(2)Operate rudder pedals to deflect rudder full left, then full right	On **ECAM FLT CTL** system page, check that **RUDDER** scale reflects rudder movement, and returns to zero when no action is made on pedals.
(3)Place **YAW DAMPER** 2 lever in **OFF** position	
(4)Operate rudder pedals to deflect rudder full left, then full right	Same result as step 2.
(5)De-energize **AFS** components by placing **PITCH TRIM** 1 lever in **OFF** position.	

(c)Close-up

1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

B. Turn Coordination

(1)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-22-00, P. Block 301	Blue Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(2)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (BLUE, YELLOW) (Ref. 29-10-00, P. Block 301 ; 29-22-00, P. Block 301 ; 29-23-00, P. Block 301).

  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUS.
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION, ECAM, ADS and EFCU
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that AP is not engaged
- make certain that ADS is operating

R
- place slat/flap control lever to the second notch.
- align IRS 1,2 and 3 (perform ALIGN procedure until red ATT flags on PFDs disappear ; present position entering is not necessary) (Ref. 34-25-00, P. Block 501)
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that AFS components are energized (PITCH TRIM 1 engaged)
- on center pedestal, adjust ECAM display unit brightness and select FLT CTL page.

NOTE : The test described below concerns side 1. For side 2, which is identical, refer to information between parentheses.

(b)Test

ACTION	RESULT
1. On overhead panel - place **YAW DAMPER 1 (2)** lever in **ON** position	Lever remains in engaged position.
R 2. Move control wheel full left	On **ECAM FLT CTL** system page, check that **RUDDER** scale reflects rudder movement to the left.
R 3. Release control wheel	**RUDDER** scale indicates 0° position.
R 4. Move control wheel full right	On **ECAM FLT CTL** system page, check that **RUDDER** scale reflects rudder movement to the right.
R 5. Release control wheel	**RUDDER** scale indicates 0° position.
R 6. De-energize **AFS** components by placing **PITCH TRIM1** lever in **OFF** position.	

(c)Close-up
1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

R

1. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Blanking Caps
Referenced Procedures	
- 22-40-00, P. Block 501	System Monitor
- 22-27-00, P. Block 501	Electric Pitch Trim
- 22-10-00, P. Block 501	Autopilot

(Ref. Fig. 401)

2. Procedure

A. Job Set-Up
(1)Open, safety and tag the following circuit breakers :
(a)For FAC1

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC1/115VAC	309CC1	103/G 5
21VU	AFS/FAC1/26VAC	305CC1	103/G 7
21VU	AFS/FAC2/28VDC	306CC2	103/G12
21VU	AFS/MTP	301CU	102/H 1
21VU	AFS/TCC/SYS1/28VDC	306CJ1	102/H8
21VU	AFS/FAC1/28VDC/BAT	323CC1	103/G2
106VU	AFS/FAC1/28VDC/STBY	322CC1	A23
133VU	L/G PROX DET & RELAYS/SYS1/RELAYS & RETRACT CTL	2GB	332/U51
133VU	L/G PROX DET & RELAYS/SYS1/FLT/GND	119GB	332/U52

(b)For FAC2

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/FAC2/26VAC	305CC2	103/G 8
21VU	AFS/FAC2/115VAC	309CC2	103/G10
21VU	AFS/FAC2/28VDC	306CC2	103/G12
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
**ON A/C 101-200,			
21VU	AFS/MTP	301CU	102/H 1
**ON A/C 001-100,			
21VU	AFS/TCC/SYS2/28VDC	306CJ2	102/H12

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301CU	102/H 1

**ON A/C ALL

133VU	L/G PROX DET & RELAYS/SYS2/FLT/GND	1GB	331/V52

(2)Open access door 121BL and gain access to FAC to be removed.

B. Removal (Ref. Fig. 401)
(1)Unscrew knurled nuts (3) at front of mount (4) and tilt them downwards.
(2)Withdraw computer (1) from mount (4).
(3)Cap electrical connectors (5).
(4)Remove computer (1).

C. Installation
(1)Clean and inspect computer interface and/or adjacent area.
(2)Remove blanking caps from electrical connectors (5).
(3)Check electrical connectors (5) for correct condition.
(4)Carefully slide computer (1) on mount (4).
(5)Engage knurled nuts in studs (2) and tighten.
(6)Remove safety clips and tags and close the following circuit breakers :
(a)For FAC1 :
306CC1, 309CC1, 305CC1, 306CC1, 322CC1, 323CC1, 301CU, 306CJ1, 2GB and 119GB.

**ON A/C 101-200,

(b)For FAC2 :
305CC2, 309CC2, 306CC2, 306CC1, 301CU and 1GB.

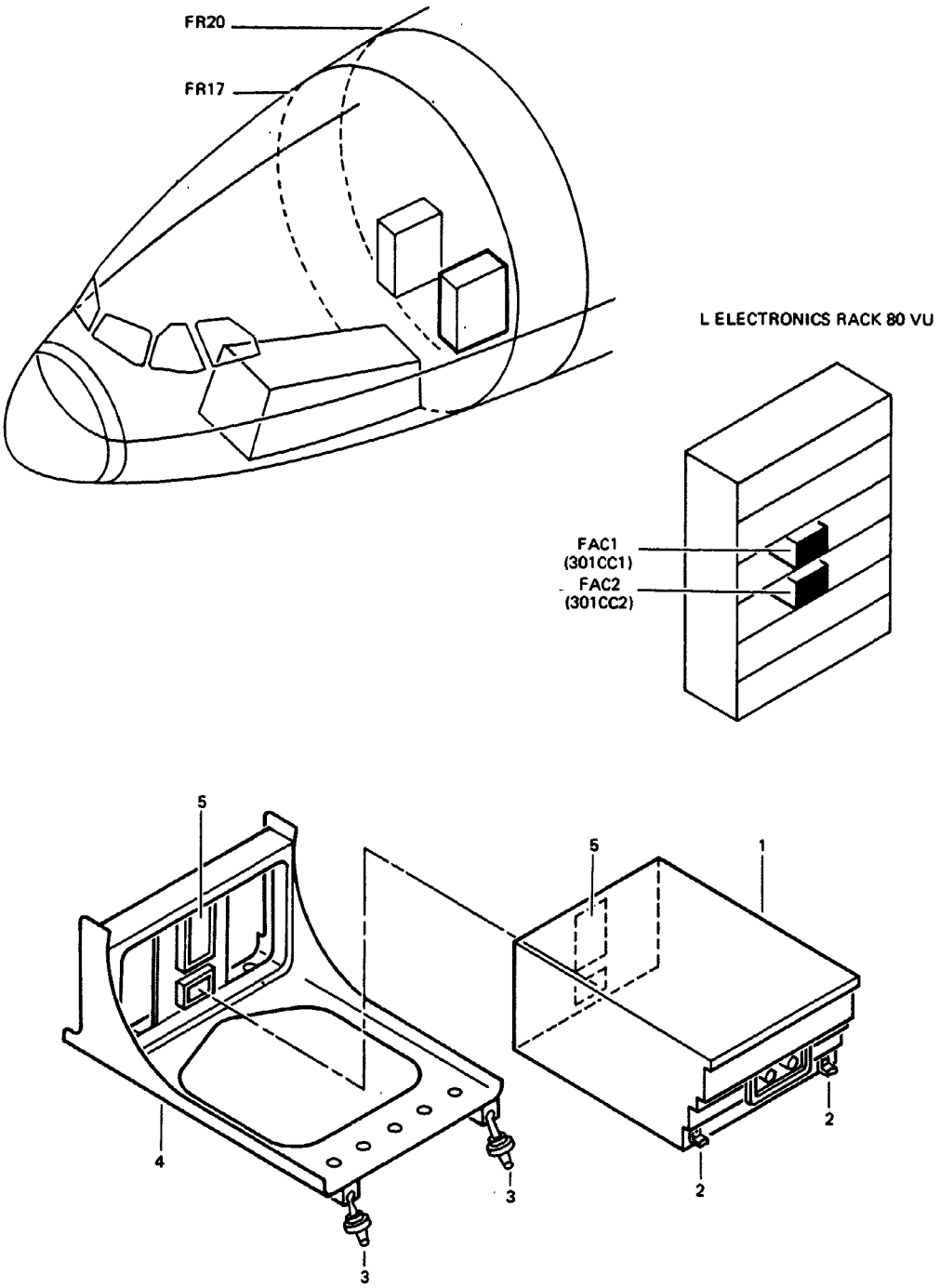
**ON A/C 001-100,

(b)For FAC2 :
305CC2, 309CC2, 306CC2, 306CC1, 301CU, 306CJ2 and 1GB.

**ON A/C ALL

D. Test
Carry out AFS test (Ref. 22-40-00, P. Block 501 paragraph 1.A.). Carry out LAND test (Ref. 22-40-00 P. Block 501, paragraph 1.B.), if aircraft is operated in actual AUTO LAND CAT 3 conditions.
R Carry out detailed visual inspection of the autotrim function
R (Ref. 22-27-00, P. Block 501, paragraph 1.G.).
Carry out autopilot operational test (Ref. 22-10-00, P. Block 501, paragraph 1.D. Check of Pitch Trim Autotrim circuits).

E. Close-Up



Flight Augmentation Computer
Figure 401

(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Close access door 121BL.

ELECTRIC PITCH TRIM - DESCRIPTION AND OPERATION

1. General

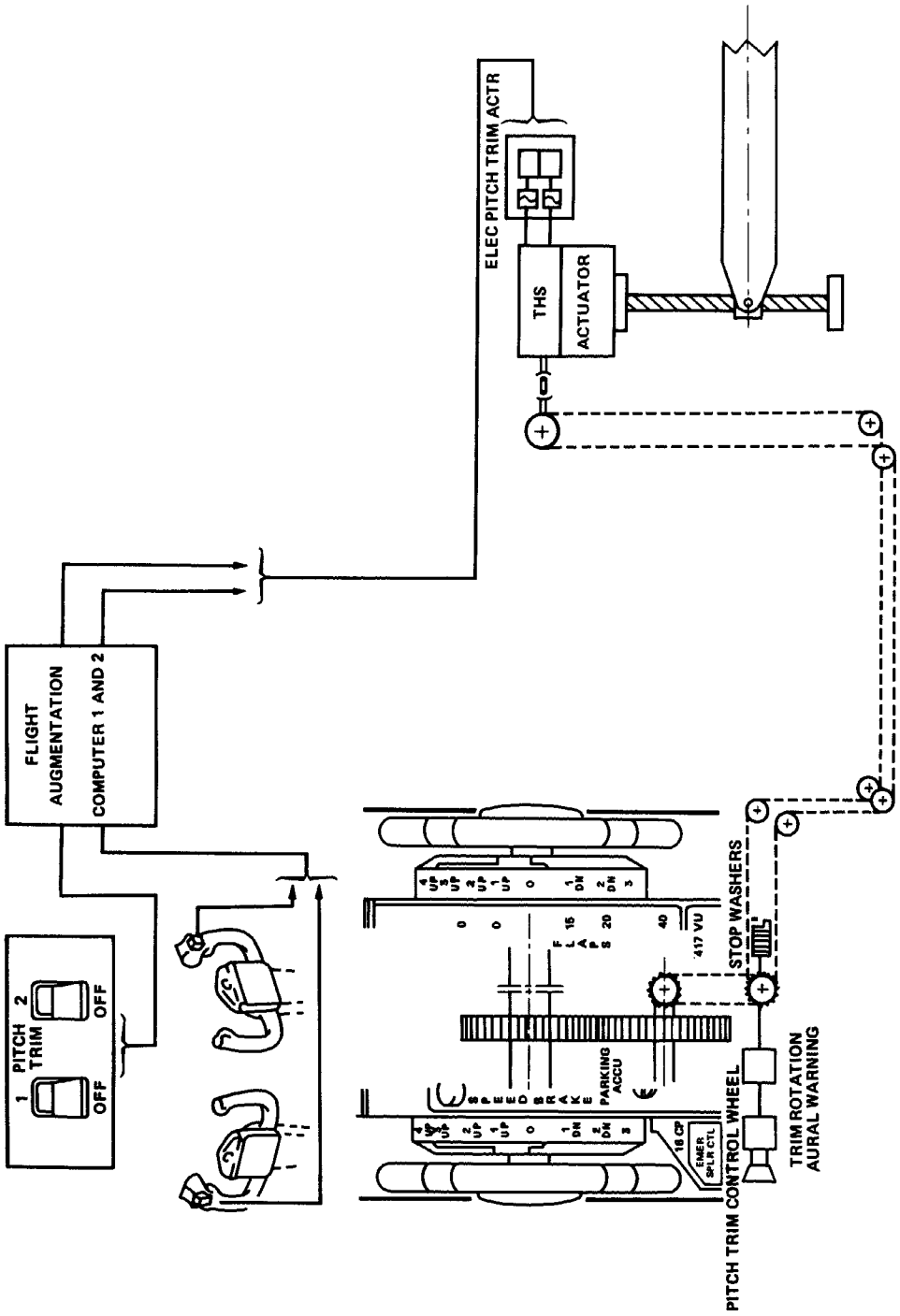
The FACS comprises an electric pitch trim sub-system which fulfills the following functions :

- electric trim
- autotrim
- static stability correctors (Mach trim, Vc trim)
- incidence trim
- flight envelope protection

The aircraft is equipped with two pitch trim systems which can be engaged simultaneously, priority being given to side 1.

The THS control principle is shown in following figure.

(Ref. Fig. 001)



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Trimmable Horizontal Stabilizer
Figure 001

2. Electric Trim Function

This is the basic function of the system. It is activated by means of trim control switches :

A. Manual Flight

It provides a means of obtaining a new stabilized position of the longitudinal axis by moving the THS. When stability is obtained in relation with the desired aircraft path, the force applied and held on the control column is null.

B. CWS on the Ground

It enables to adjust the takeoff trim without it being affected by an autotrim type loop.

C. CWS in Flight

It is used to enable adjustment of the reference attitude and associated stability trim.

The THS commanded displacement rate varies according to the flight envelope :

- (low speeds) Vc ≤ 200 kts
  commanded displacement rate = 0.9° THS/s
- (high speeds) Vc > 240 kts
  commanded displacement rate = 0.17° THS/s

Variation law is linear between Vc = 200 kts and Vc = 240 kts.

These are deflection limit speeds with constant force applied to the trim
R controls (Ref. Fig. 003).

Trim control is achieved by means of the pitch trim control switches mounted on the Captain's and First Officer's control wheel outer horns.

Each pitch trim control switch has three positions :

- rest (control switch is spring-loaded in this position)
- nose down
- nose up.

Two microswitches (trim 1 and trim 2) are associated with each working position.

They send the following reliable signals to each FAC :

- TRIM UP (nose up)
- TRIM DOWN (nose down).

The retained wiring principle enables to override any possible jamming of one microswitch and ensure complete segregation between trim 1 and trim 2.

The signal delivered by the pitch trim control switches is processed by an integrator in the FAC and the THS moves as long as action on the switch is maintained.

Any electric trim request which lasts longer than one second is indicated aurally by a "whooler".

The activation logic of the electric trim is as follows (Ref. Fig. 002):

- trim engaged and having priority (trim clutched).
- electric trim request when in one of the following conditions :
  - in manual flight (AP engage lever in OFF position)
  - in CWS with aircraft on the ground
  - in CWS in flight.

In the event of a contradictory order from the Captain and First Officer

on the pitch trim control switches, the commands are not taken into account.
In the event of a contradictory order from one pitch trim control switch (e.g. failure) trim disengages.

**ON A/C ALL

R

(Ref. Fig. 003)

**ON A/C ALL

R

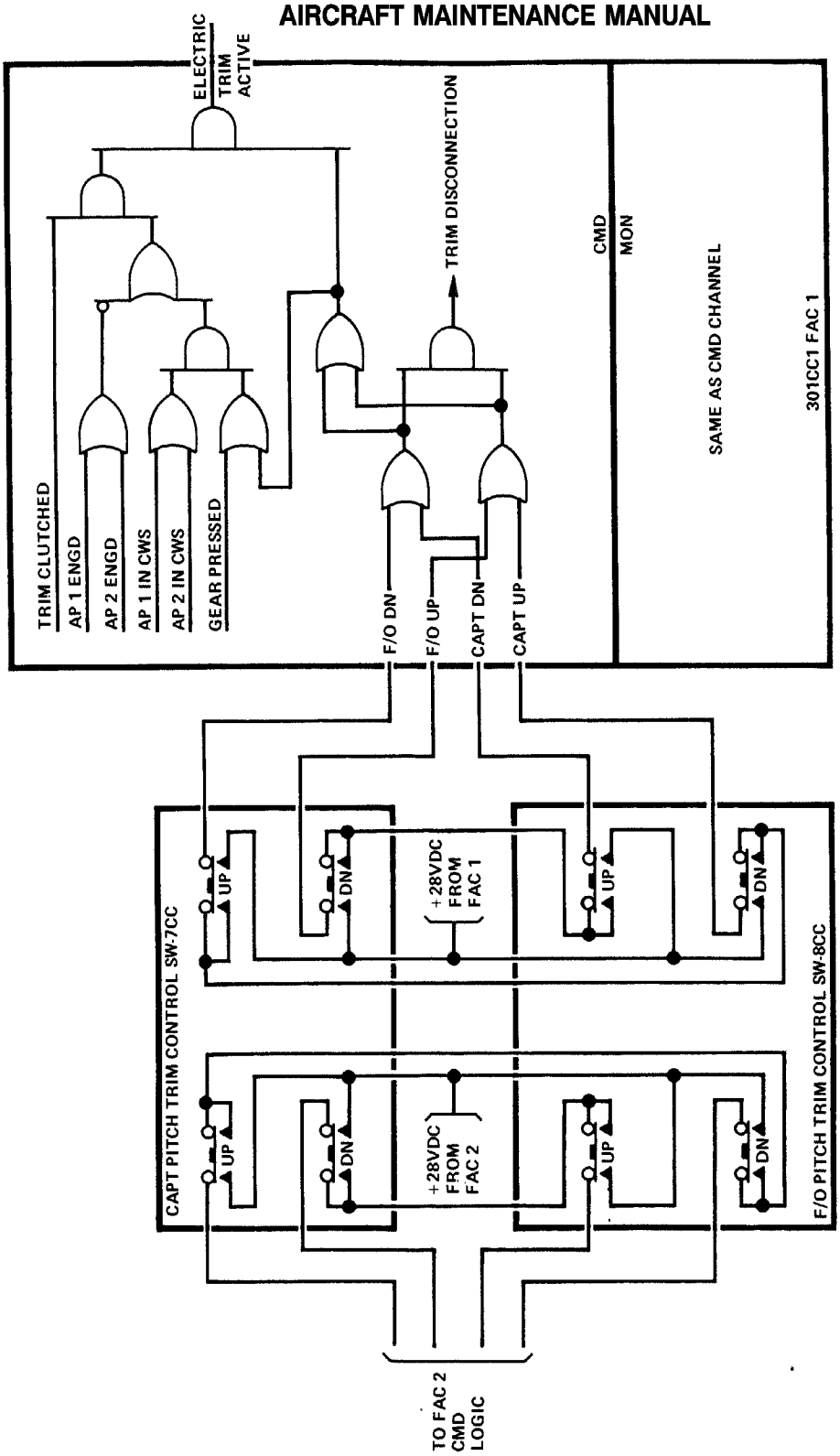
3. THETA TRIM Function
The purpose of the **THETA TRIM** function is to help the crew to control the pitch attitude of the aircraft during strong thrust variations in go around.
The **THETA TRIM** function sends a pitch order to the **THS**. This order is only a positive order (nose-down order). It is defined to target 25° of pitch attitude whatever the aircraft configuration (slat/flap configuration, weight and Center of Gravity).
When the **THETA TRIM** activation conditions are detected, a **THETA TRIM** order is generated. This order is cancelled after 15 seconds.
Subsequent activation of the **THETA TRIM** function is inhibited for 60 seconds.
This function is available :
- if the autopilot is disengaged
- when pitch attitude θ is greater than 10°
- when aircraft speed is below 180 kts and Mach is below 0.37.
- 10 seconds after lift-off.
This function is inhibited if the aircraft is on ground, 10 seconds after main gear is compressed.

**ON A/C ALL

R

(Ref. Fig. 004)

**ON A/C ALL



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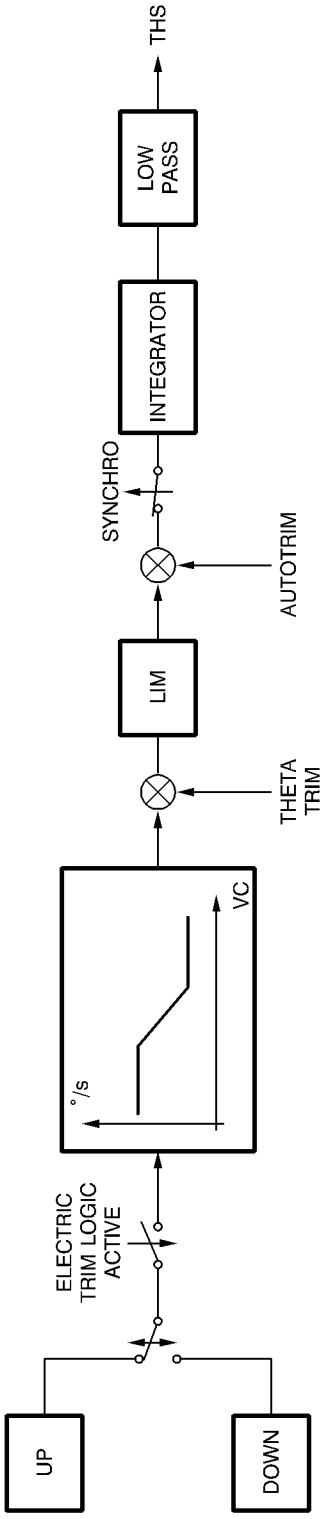
Pitch Trim Control Switches - Principle and Logic
Figure 002



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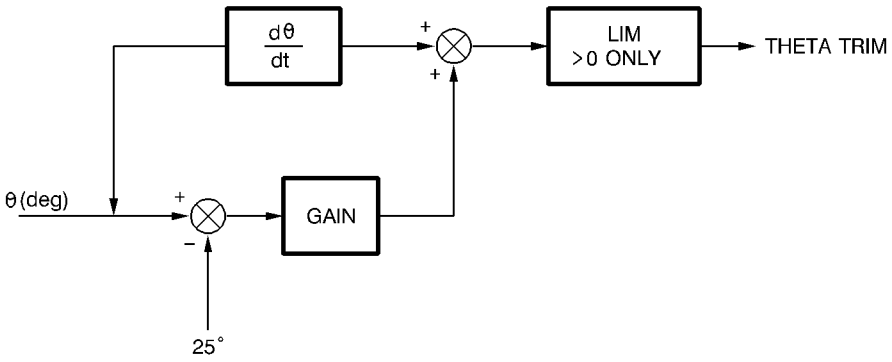
Electric Trim Low
Figure 003

R



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THETA TRIM Law - Principle
Figure 004

4. Auto Trim Function (Ref. Fig. 005)

The purpose of the auto trim function is to enable AP operation around a zero control surface position.

To this end, the AP elevator order from engaged AP will be integrated by the trim system, thus enabling the angle formed by the elevator with the THS to be permanently corrected to zero degree position. Working around a zero position also enables, in the event of AP disconnection, to have a correct trim setting and prevent too great load factors (> 0.1 g) due to modifications in stability conditions. The maximum THS commanded displacement rate depends on the flap configuration :

- flaps $\geq 15^{\circ}$ (S/F sel. pos ≥ 3) commanded limit rate = 0.5° THS/s
- flaps $< 15^{\circ}$ (S/F sel. pos < 3) commanded limit rate = 0.066° THS/s

These are deflection limit speeds with a constant command.

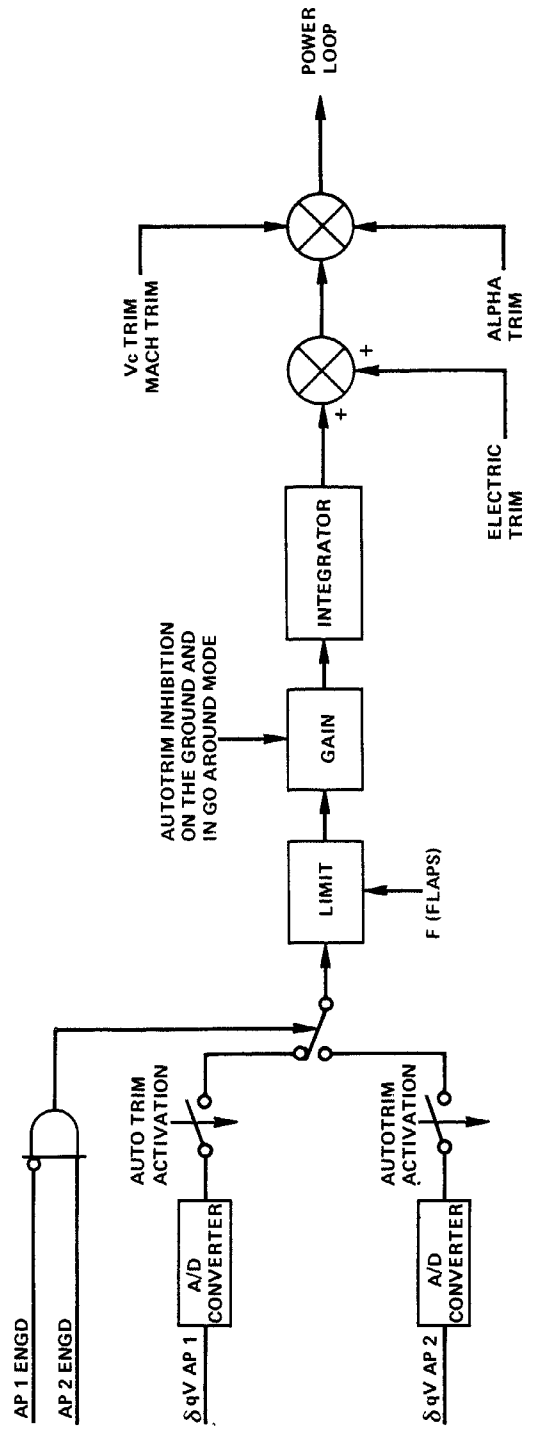
The auto trim command is delivered by the engaged AP in analog form for safety reasons. The transmitted command is the deflection command downstream of FCC voter. As each FAC is connected to one voter, the command distribution is such that any simple voter, output or wiring failure does not result in both FAC loss.

Monitoring of autotrim commands is ensured :

- at FCC level :
  - . by the voter comparator itself
  - . by power comparators in the event of a voter undetected failure.
- at FAC level :
  - . by comparison between command and monitoring channels (C3, C5, C6, C12 and speed monitor) and by reduced equalization of monitoring channel on command channel (limit and gain).
    The comparator C6 activation (active in clean configuration only) has to involve the loss of the trim system and disengagement of the trim engage lever.

The AP deflection signal Δq is processed by an integrator in the FAC and the THS moves as long as the AP command is not cancelled thus ensuring slaving of the longitudinal axis between control surface and THS. The law principle is given in following figure.

(Ref. Fig. 005)



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Auto Trim Law - Principle
Figure 005

5. Static Stability Correctors

Depending on the flight envelope which is being considered, various aerodynamic phenomena can modify the aircraft qualities with respect to normal envelope. The pitch trim system provides a compensation of any possible difference which could affect the control of the aircraft by fulfilling the following static stability augmentation functions :

- Mach trim
- Vc trim (speed trim)

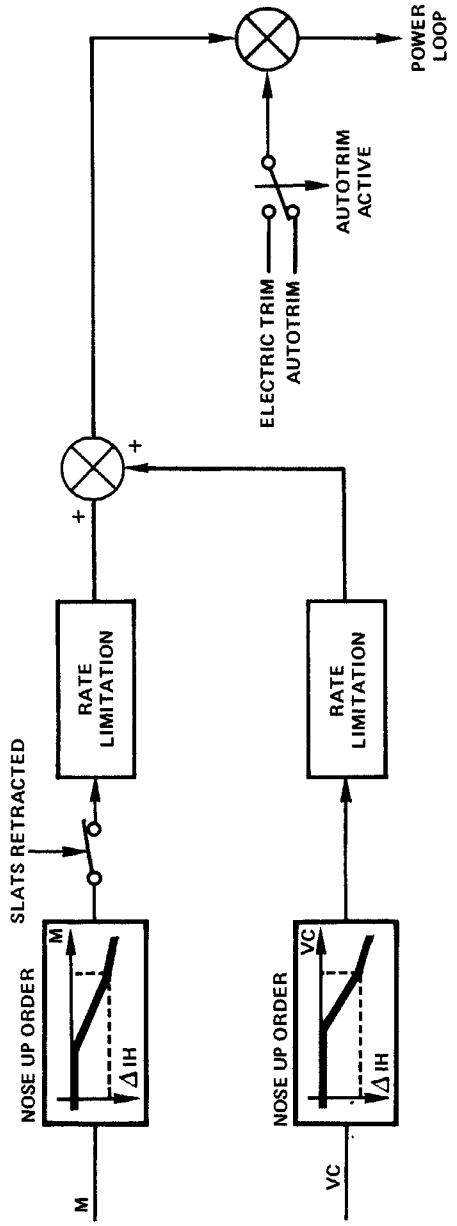
A law, function of the Mach number and corrected airspeed (Vc), enables to restore a static stability margin in high speed configuration (clean configuration). To this end, the THS position is modified according to a deflection law programmed as a function of the Mach number or Vc so that during an acceleration, the corrective action from the pilot is always in nose down direction. Consequently the aircraft must be in nose up direction (IH in nose up direction).

This correcting law is always active at high speeds and is added to the trim commands :

. Mach trim authority is -0.7° (nose up order)
. Vc trim authority is -0.82° (nose up order)

The common rate limitation is 0.05°/s THS.

(Ref. Fig. 006)



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Mach Trim and Vc Trim Law - Principle
Figure 006

6. Incidence Trim Function

A. General
Longitudinal stability augmentation is necessary to meet the certification requirements. This augmentation is given by a THS motion, function of a law dependent on angle-of-attack.

B. Principle
Longitudinal stability augmentation is obtained by transmitting a deflection command of the following type to the trimmable horizontal stabilizer :
ΔTHS = K (alpha - alpha 0)
with alpha 0 variable depending on configuration.
This additional deflection command must be applied or inhibited progressively. THS variations must be avoided during configuration modifications (flight, ground, extension or retraction of control surfaces). To this end, utilization of memorized values enables to take into account the deflections prior to configuration modification. THS motion is achieved by using the memorized value adjusted reference.

C. Standard Incidence Trim
The following laws provide protection against flight envelope deviations and enable stability margin augmentation by sending a nose down command as soon as the aircraft angle-of-attack signal which is not filtered exceeds a predetermined threshold. The following law is integrated : ΔIH3 = K (alpha - alpha 0) with the following characteristics : . law is active as soon as alpha > alpha 0 (alpha = aircraft angle-of-attack).

**ON A/C ALL

R

. alpha 0 is function of the Mach number varying between 11° and 1.6° in relation to increasing Mach numbers.
. K1 is a constant equal to 0.8° THS/° alpha (incidence trim again).
. K2 is a constant equal to 0.1° THS/°/s.
. authority limit = 1° THS (nose down only).
. maximum displacement rate = 0.8°/s (rate limit 1).

**ON A/C ALL

The (alpha) trim law is activated in CWS mode (clean configuration) if the pilot applies a force on the control column in order to increase the protection against high incidence caused by aerodynamic discontinuity.
The law is inhibited under one of the following conditions :
- slats extended (S/F Sel pos > 1)
- AP engaged in CMD
- speedbrakes extended.
The latter conditions avoid pitch oscillation risks.

D. Not Applicable

E. Anti Stall Function

The purpose of this function is to assist the stall recovery through a predetermined nose down command to the THS.

The conditions of activation and deactivation are dependent on the angle of attack value and the slats/flaps configuration

- activation :

(alpha ≥ alpha threshold - $\frac{1.5\ s}{1 + 0.5\ s}$ alpha). (alpha ≥ alpha min).

(S/F selector position ≥ 4)

- deactivation :

alpha< alpha min.

ANTI-STALL CHARACTERISTICS	SLATS/FLAPS SELECTOR POSITION				
	1	2	3	4	5
ALPHA THRESHOLD				21°	21°
ALPHA MIN				16°	16°
NOSE DOWN COMMAND AMPLITUDE				4°	4°

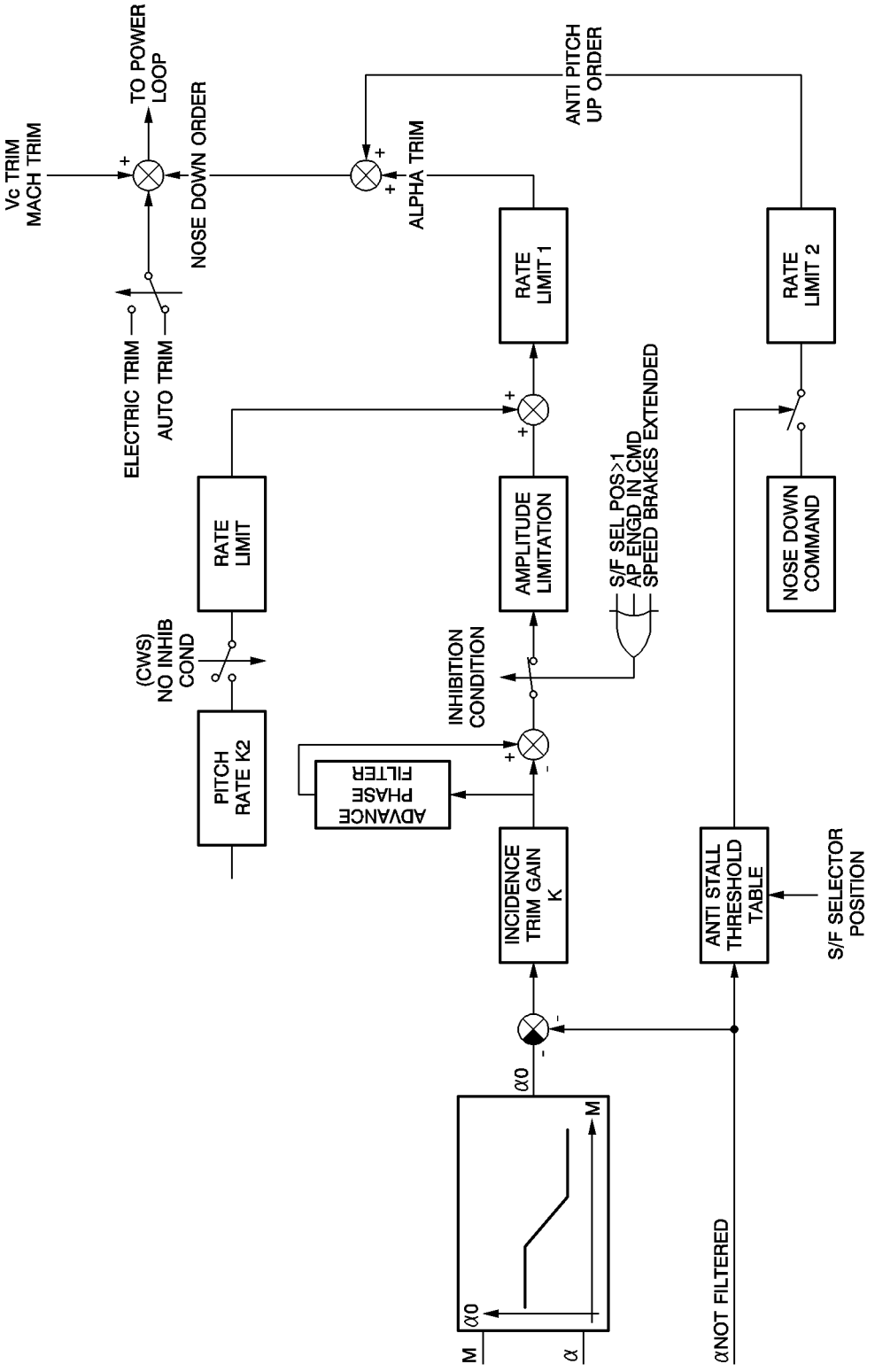
The nose down command maximum rate (rate limit 2) sent to THS actuator is 0.9°/s for every configuration.

**ON A/C ALL

R

(Ref. Fig. 007)

**ON A/C ALL



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Incidence Trim Law - Principle
Figure 007

7. Pitch Trim Engage Logic

A. General

Hold of the engage lever in the **ON** position depends on the validity of the command and monitoring logics.
It is to be noted that the clutch is supplied if :
- command and monitoring logics are valid,
- the engage lever is in the **ON** position,
- the self-holding relay is energized.

The self-holding relay is power supplied via the torque limiter microswitch. System disconnection results in return of the engage lever to the **OFF** position.

B. Engage logic

The engage logic principle is given in figure (Ref. Fig. 008).
The engagement circuit is composed of :

(1) Engage lever

The engage lever ensures :
- the engagement command (lever placed in **ON** position)
- the self-holding relay internal power supply after having passed through the torque limiter microswitch and self-holding relay **307CC1** of the aircraft wiring
- the engage logic status acquisition in the command channel.

(2) Self-holding relay

The self-holding relay **307CC1** of the aircraft wiring (electronics rack side) ensures :
- the engage logic status acquisition in the monitoring channel
- the engage logic status acquisition in the command channel.

(3) Hard-wired engage logic

The hard-wired engage logics in the command and monitoring channels are independent and fulfill the two following functions according to the mode and status logic (**TRIM LOG** signal) and to the computer hardware monitoring (comparator **C3**, command and monitoring watchdogs, power comparator inhibition monitoring) :
- engage lever cutoff
- clutch line cutoff.

(4) Status and mode logic

The status and mode logics in the command and monitoring channels are processed in the software part and ensure mode and computation monitoring.
All these conditions must be validated in the command and monitoring channels to enable self-holding of the engage lever and clutching of the actuator.
Possible mechanical locking of the lever is taken into account and clutch cutoff is also provided.
The hard-wired logic enables validation of :
- the actuator clutch
- self-hold of the engage lever on **FAC/ATS** engage unit **318CC**
- self-hold of self-holding relay **307CC1**.

The following conditions are required :
- trim software monitoring

- status of comparator C3 located at the extremity of the channel and comparing the command and monitoring computation
- state of command and monitoring watchdogs
- cross-check of oscillation sensors at TRIM LOG output.

These hardware conditions are delayed by 100 ms before they control :

- the lever
- the relay
- the clutch.

It is to be noted that :

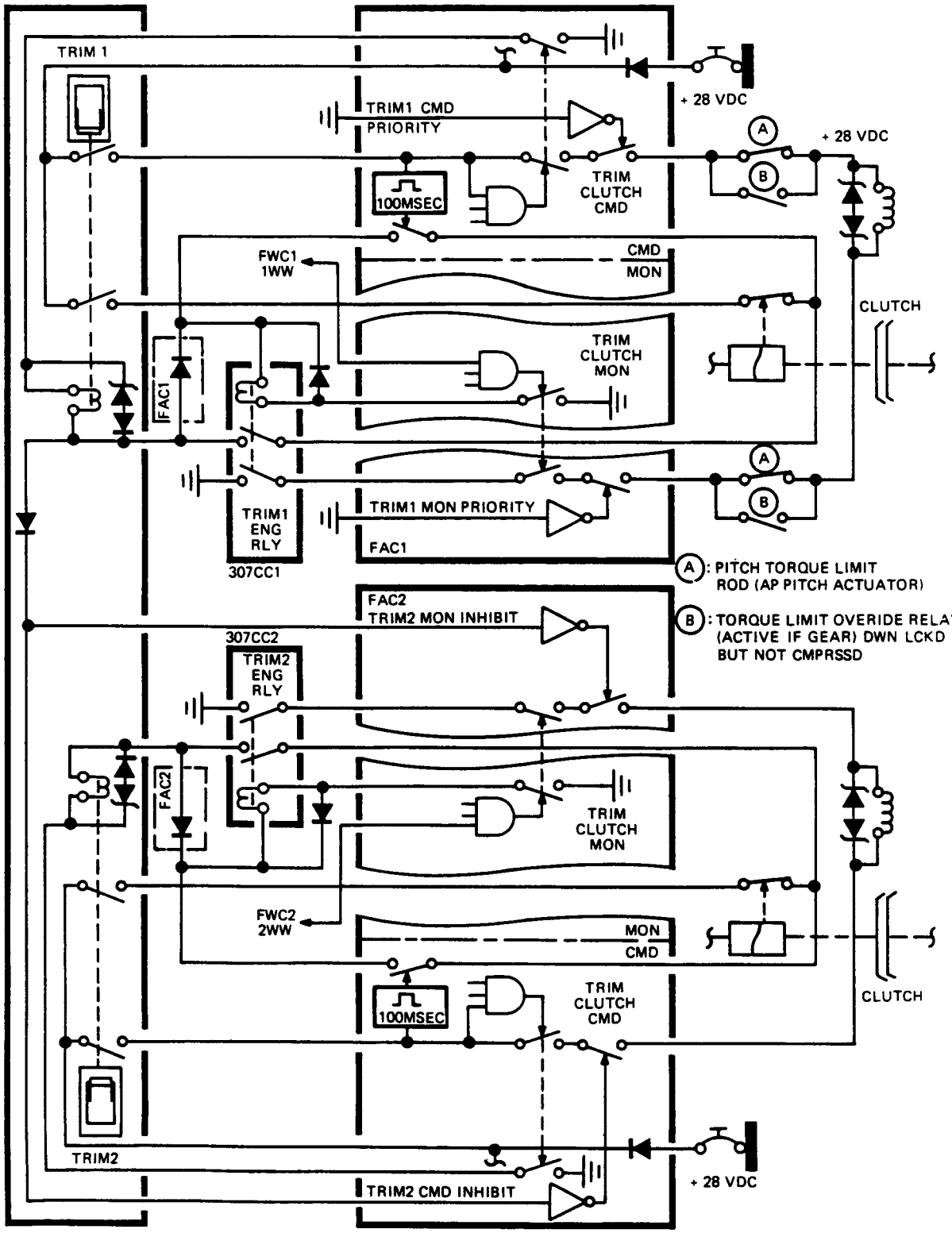
- on side 1 only, the AP torque limiter conditions trim 1 clutch line.
- on both sides the trim torque limiter conditions the engagement self-holding. Any opening of this limiter (e.g. control wheel locking in trim rotation) disconnects the trim.
- mechanical locking of the trim engage lever does not prevent trim clutch cutoff.

NOTE : Anti-buzzing circuit

To avoid trim runaway in a case where failure is detected but lever does not return to initial position (hard spots), a specific circuit has been included in the engage logic. This prevents system reengagement, entailing a change from synchro to active mode as long as the lever is not returned to initial position and reset (from OFF to ON). To this end, a command channel relay calibrated to 100 ms ensures locking in the non engaged state as soon as a failure is detected by a comparator ; this entails actuator clutch cutoff even if lever remains engaged (trim changes to synchro mode).

Relay is reset when lever is released and then attempted to be engaged again.

(Ref. Fig. 008)



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Pitch Trim 1 and 2 Engage Logic
Figure 008

C. Pitch Trim Mode and Status Logic

(1)General

The mode and status logic is accomplished in the software part independently for the command and monitoring channels. It ensures pitch trim monitoring in its various functions. The result of this logic **(BTRIMLOG)** is taken into account in the hard-wired engage logic.

(2)Mode and status logic principle

Various conditions must be met simultaneously so that the trim can be engaged and remain engaged.

(a)Mode conditions

The following trim modes are taken into account with their associated monitoring functions :

1 Electric trim
- pitch trim control switch monitoring (no **UP** or **DOWN** contradictory order)

2 Autotrim
- acquisition of the **THS** displacement rate monitoring upon electric trim non demand **(TTRATE)**.
  Operation of this comparator is delayed at :
  - electric trim de-activation
  - flap retraction
  - trim clutching.
- acquisition of **C12** external monitoring ($IH^{\circ} - \Delta q$) in relation to the auto trim mode, on the active trim only.
- acquisition of **C6** : comparison of the control channel trim demand and monitoring channel trim demand delivered by the active **AP** (Active only in clean configuration).
  NOTE : Comparator **C12** is overridden by various conditions which are delayed in order to prevent untimely transient activations due to this comparator :
  - at activation of the autotrim mode (1-second delay)
  - when switching to synchro mode, active mode (1-second delay)
  - at inhibition on stops (1-second delay).

3 Static stability correction
- Mach and speed monitoring in cruise (clean configuration)
- Angle-of-attack monitoring in incidence trim (flag **BFAOA** representing angle-of-attack vote processing).

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- comparison of incidence trim demand **(DIHC)** of each channel through comparator **C7**.

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(b)Status conditions
- monitoring of safety tests associated with the trim part

- power channel monitoring (comparator C4 and C5)
- monitoring of 12K bus transmission operation (No refresh, word BDISC2)
- monitoring of cancellation of inhibition conditions at the end of certain delays of trim power comparators C4, C5, C12, BTTRATE
- FAC general monitoring (RTC, internal power supplies, ARINC management, watchdogs, safety tests)
- landing gear monitoring
  . 28 V validity check
  . landing gear dissymmetry check (15-second delay)
- monitoring of angle-of-attack data processing performed on the average angle-of-attack computation in the monitoring channel only
- engagement command monitoring (engage lever in ON position)
- monitoring of power supply automatic cutoff on ground
- monitoring of correct transmission of equalization order from the command channel to the monitoring channel (in the monitoring channel only).

All these conditions are necessary to engage the trim.
(Ref. Fig. 009)

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(Ref. Fig. 010)

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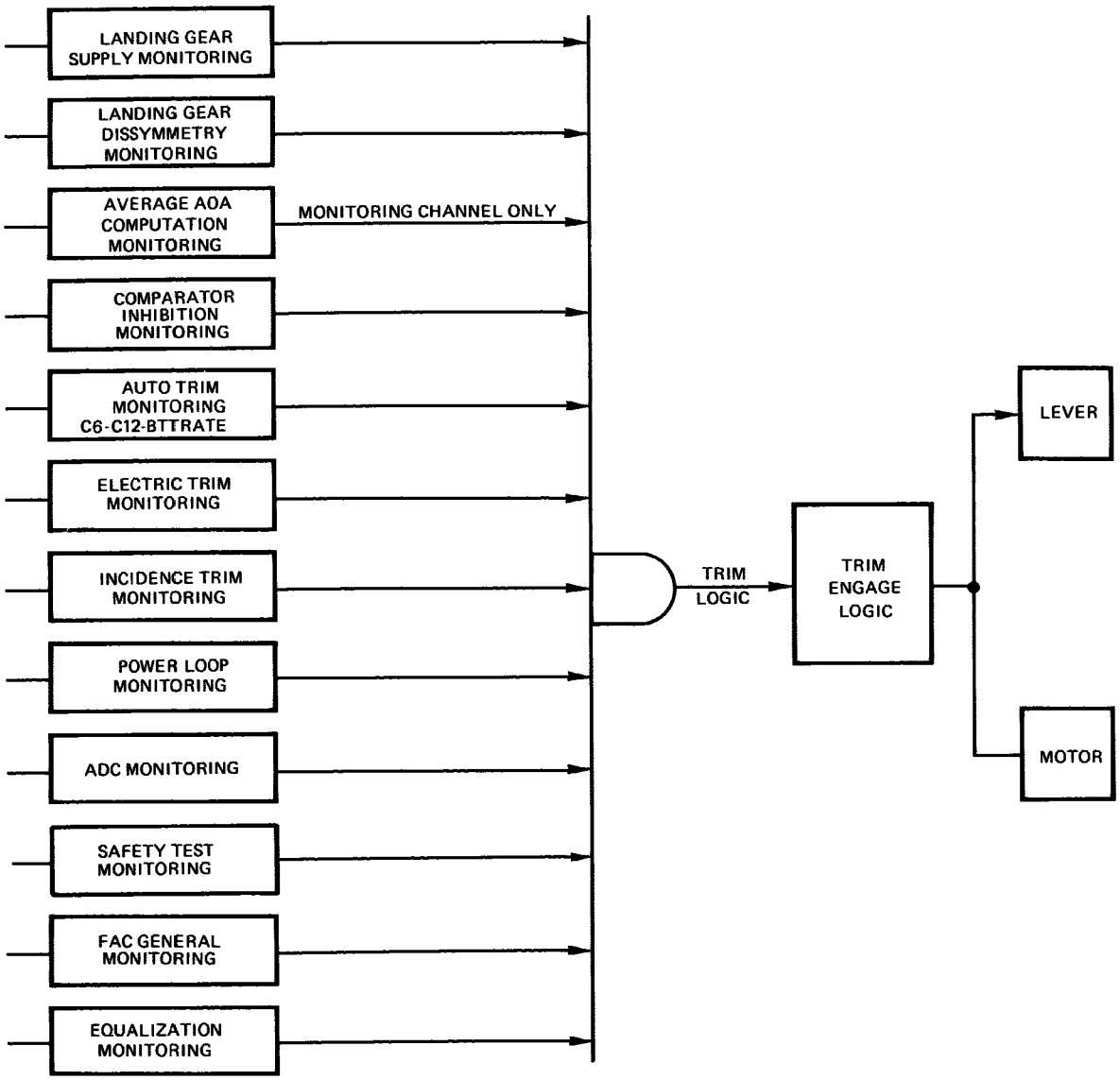

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Pitch Trim Mode and Status Logic
Figure 009

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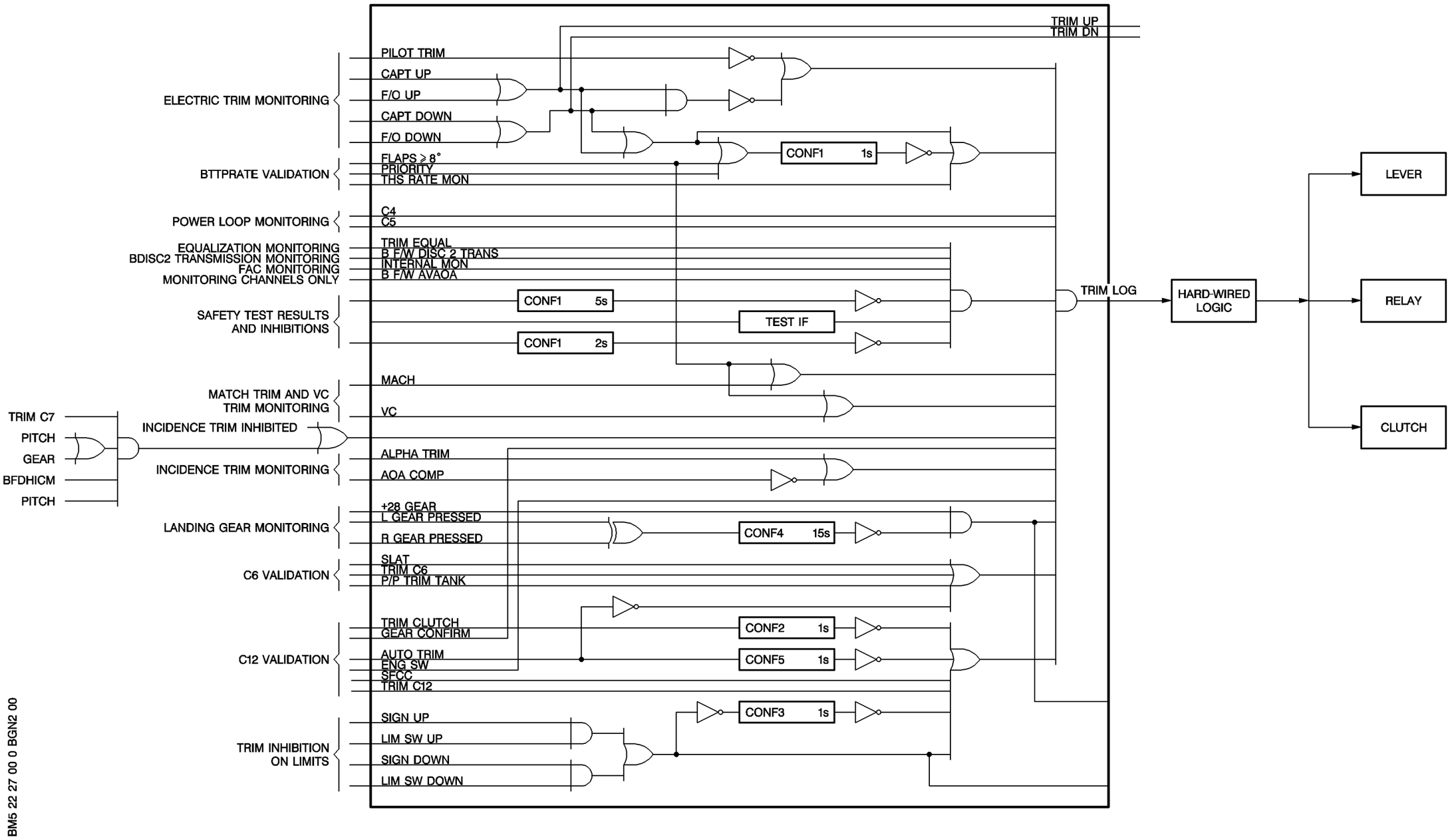

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Mode Logic - Schematic
Figure 010

D. Pitch Trim Function Activation Logic

The activation of the various pitch trim functions depends on flap and slat extended configuration conditions and FCC engagement status. When the trim is not in electric or auto trim mode it is in synchronization mode with actuator declutched.

(1)Electric trim

The electric trim is activated :

- upon pilot's command (UP or DOWN) and if the trim is clutched and if at least one of the following conditions is met :
- no AP engaged
- at least one AP in CWS mode and action on trim control switch.

(2)Auto trim

The auto trim function is activated :

- when the electric trim is not activated (both functions are incompatible)
- when the trim is effectively clutched (signal BTCLUTCH) and if at least one of the following conditions is met :
  - . at least one AP engaged in CMD mode in flight. In single AP configuration both FAC's select auto trim order from engaged AP. In dual AP configuration each FAC selects auto trim order from associated FCC.
  - . at least one AP engaged in CWS mode without electric trim demand (pitch trim control switches) and without the following conditions :
    - on ground and 5s after take off.
    - after 60s in flight if landing gear extended and no FD, in SRS or GO AROUND mode when an effort is detected on control column.

  In any of these cases autotrim function is forced to zero to facilitate maneuver.

(3)Static stability correctors

The static stability correction functions are available if the trim is clutched, in these condition mach trim and Vc trim functions are activated as soon as the slats are retracted (clean configuration).

(4)Incidence trim functions

(a)Anti pitch-up function

(See para 5.D.)

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(b)Alpha trim function

The alpha trim function is activated if necessary, in clean configuration (slats and speedbrakes retracted) and in CWS mode (AP not in CMD).

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(c)Anti stall function

The anti stall function is activated if necessary (high angle of attack condition) depending on the slats//flaps configuration through the control lever position. (See table in para. 5.E.).

8. Monitoring of Parameters Used by the Pitch Trim System

A. General
Some parameters undergo specific monitoring. These are :
- landing gear data
- angle-of-attack data.

The other parameters are monitored at the level of their status matrix or by software or hardware comparators.

B. Input Parameter Monitoring
(1)Computed airspeed
Flag and No refresh monitoring is achieved at the level of the status matrix.
The NCD is not taken into account as a warning.
This type of monitoring is overridden when the slats are extended (S/F sel : pos > 0) . Consequently, ADC loss does not lead to trim loss in approach.
(2)Mach number
Flag and No refresh monitoring is achieved at the level of the status matrix. The NCD is not taken into account as a warning.
This type of monitoring is overridden when the slats are extended (S/Fsel : pos > 0) in order to maintain the trim in approach upon ADC loss.

C. Trim General Monitoring
The software comparators C4 and C5 check the power loop :
- C4 = FU1 signal - FU2 signal
- C5 = trim command signal - FU1 signal.

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The software comparators C6, C12 and CTTPRATE check the auto trim function :
- C6 = delta q AP from CMD channel - delta q AP from MON channel (from active AP).
- C12 = autotrim command signal - THS response (delta q - iH)
  NOTE : A 0.5 second delay is implemented on C12, in CWS mode, clean configuration, speedbrakes retracted, when a stick force is applied to the column. This delay is also implemented in CWS mode or AP off clean configuration, speedbrakes retracted when alpha reaches alpha 0 - 0.5° in the monitoring channel only.
- CTTPRATE = THS displacement rate due to auto trim command only.

The hardware comparators C3 are located at the extremity of the channel, after A/D conversion, and monitor the whole trim computation :
- C3 = final command in CMD channel - final command in MON channel.
- C7 = standard incidence trim command in command channel. Standard incidence trim command in monitoring channel.

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(Ref. Fig. 011)

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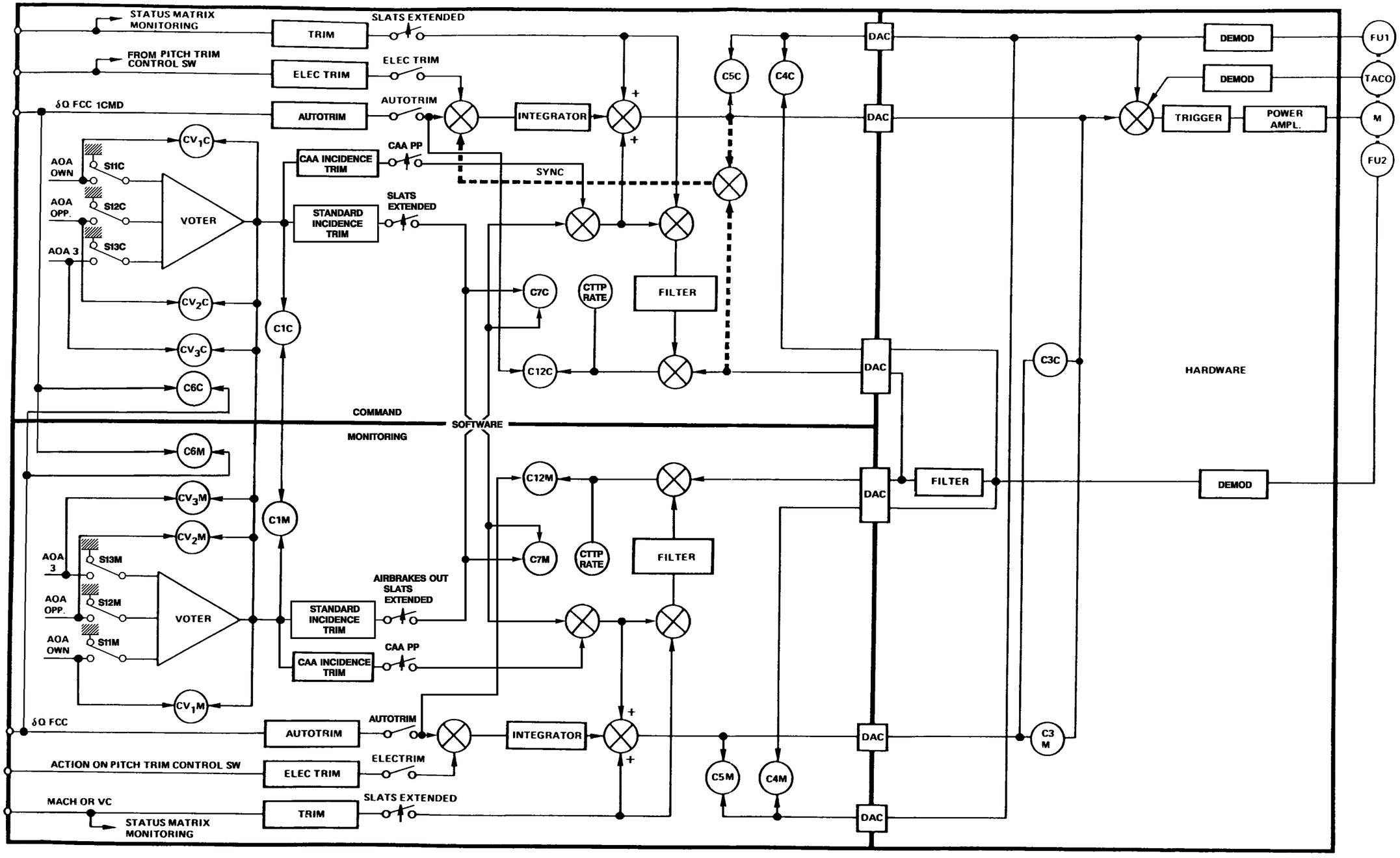
The trim does not disconnect on its mechanical stops the following electrical limits are set :
- in nose-down direction = +2.5°
- in nose-up direction = -13°.

Limit detection is performed in FAC by follow up data processing.
Detection protection is ensured by CMD and MON independent computation and by channel exchange via bus.
Inhibition is effective when detection is performed by both channels.



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Trim Monitoring - Block Diagram
Figure 011

C.	*TYPE S H	PHYSICAL THRESHOLD	DELAY (ms)	CONDITION OF OPERATION	COMMENTS
CV1	S	3.6° AOA	375	Permanent	S11 open after comparator tripped
CV2	S	3.6° AOA	375	Permanent	S12 open after comparator tripped
CV3	S	3.6° AOA	375	Permanent	S13 open after comparator tripped
C1	S	1° AOA	450	Active if slats retracted	
C3	H	0.25° THS	100	Permanent	
C4	S	0.85° THS	100	Permanent	
C5	S	0.25° THS	160	Permanent	
C6	S	0.35° Δ q	600	Autotrim and slats retracted.	
CTTP RATE	S	0.2° THS/ s	600	Flaps retracted and no action on pitch trim control switches.	
C12	S	2.5°	100	Auto trim and nose-up or nose-down limits are not reached.	

* TYPE S = SOFTWARE
  TYPE H = HARDWARE

9. Equalization Principle

In fact the trim is an integrator.
Consequently an equalizing circuit is provided to prevent deviations of values and more precisely differences between the command and monitoring channels which could affect general monitoring (C3).
The monitoring integrator is slowly synchronized on the command integrator output.
Due to the high rate of the anti-stall signal (0,9 °/s) and the fact that the processing of this function is made during a long cycle (150 ms), a dissymetry may appear between control and monotoring channels in high incidence configuration, causing trim disconnection (C 3 or C5). For this reason the anti-stall signal is taken into account in the equalization principle.
(Ref. Fig. 012)

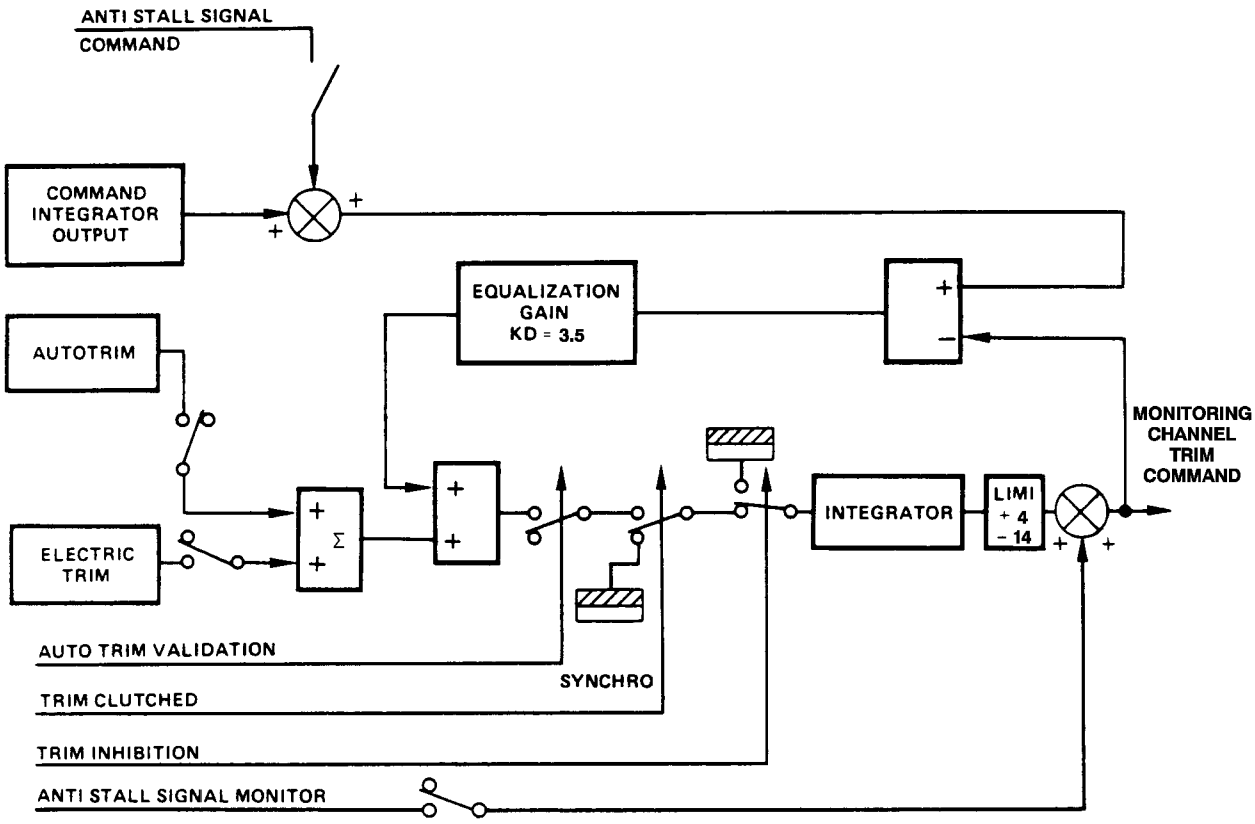
10. Pitch Trim Operation - Change-Over Principle

A. General

The electric pitch trim function is fulfilled by a double-checked system. Double checking is accomplished by two independent channels (FAC1 and FAC2). As the purpose of the system is to ensure maximum availability of this function, any loss of a channel must be compensated by the remaining channel in a time short enough so that no appreciable movement of the aircraft is perceptible.
To this end, the retained principle could be either simultaneous operation of both channels or instantaneous priority transfer operation upon loss of a channel.
The integration of the pitch trim controls in the flight controls does not allow simultaneous operation of both channels. The change-over principle is retained : it consists of a simultaneous engagement of both channels and ensures priority of channel 1 which is the only one to be clutched although both trims are engaged.

B. Operation

Both pitch trim systems can be and must be engaged at the same time and both engage levers are self-held if the logic conditions are met.
Nevertheless only the clutch of pitch trim actuator 1 is activated.
The other channel is thus engaged but its clutching is in standby.
The clutching command will be achieved when PITCH TRIM 1 engage lever returns to the OFF position.
Accomplishment is made by creating an inhibition command of pitch trim 2 actuator via PITCH TRIM 1 engage lever circuit.
(Ref. Fig. 013)



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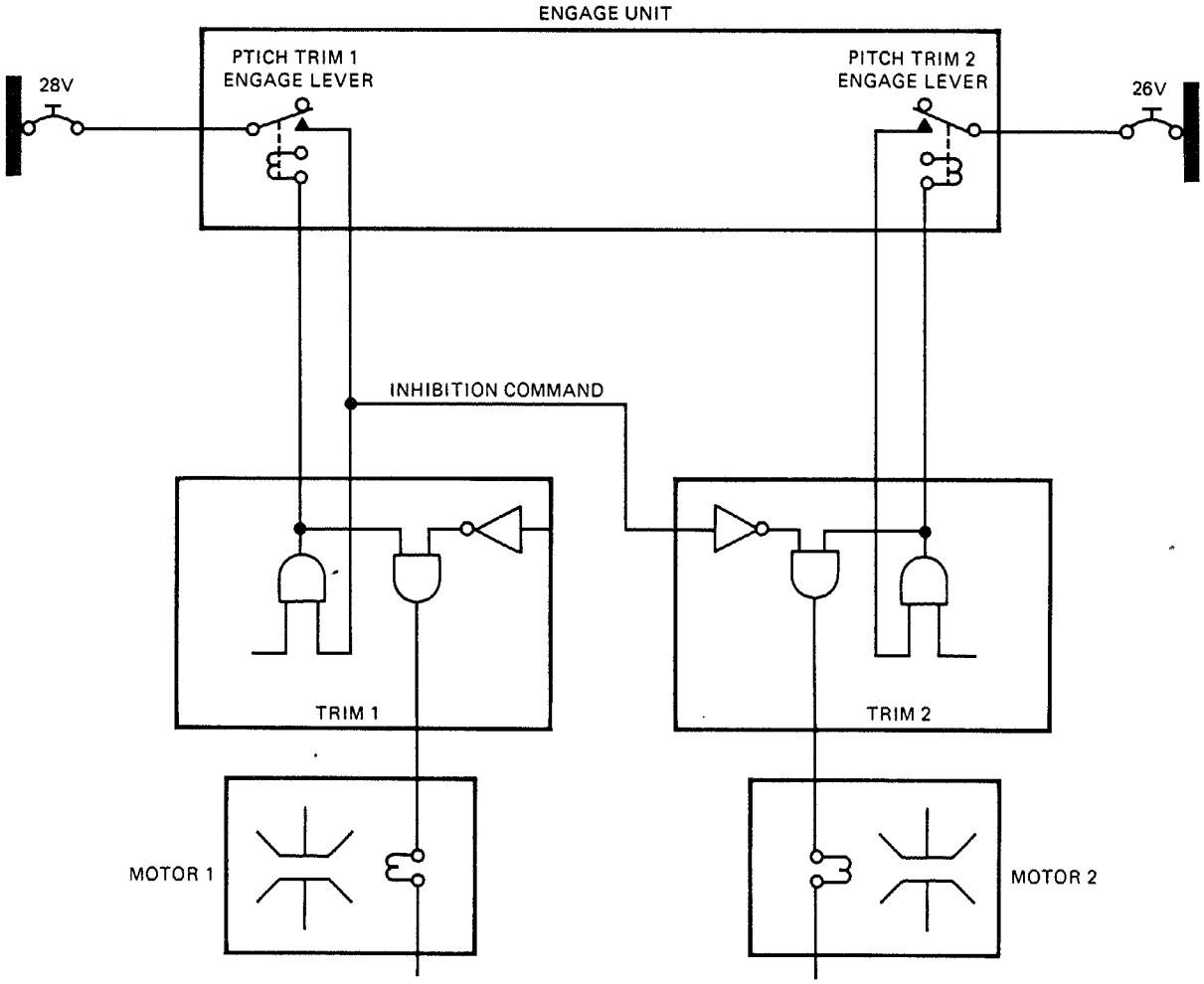
Trim Equalization Principle
Figure 012



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DM5

Trim 1 Priority
Figure 013

11. Pitch Trim Power Loop

A. Pitch Trim Power Loop Principle

(1) General

Trim command is accomplished by an electric actuator of single-phase motor type with squirrel-cage rotor.

The stator comprises two windings :

- phase A : control voltage generated from the error command. This is the check phase.
- phase B and C : motor energizing voltage generated from the 200 V/ 400 Hz in quadrature with phase A. This is the reference phase.

The operating principle is shown in figure (Ref. Fig. 014).

The circuit is composed of :

- a direct channel ensuring pitch trim actuator control after the input commands are changed into control pulses whose duration is proportional to the input command.
- a feedback channel ensuring :
  - . slaving stabilization through a tachometric feedback
  - . position slaving through a synchro feedback.

This is a "discrete" type operation ; as long as a command signal is present, pulses are applied to the check phase which is cut when the input command signal is inhibited.

These commands are proportional to the slaving error signal.

The winding power supply is checked by two thyristors. Either thyristor is controlled to deliver 115 V whose phase authorizes rotation of the motor in one direction or the other.

The rotation direction is detected by comparators. A control threshold is integrated (0.015° THS).

When the control winding is continuously supplied with power (theoretical case) the motor operates at nominal constant speed. In fact the control commands are of short duration and the motor maximum speed is contingent on the control command duration.

(2) Synchronization

When a trim system is not active, it is synchronized with the THS position. The clutch is disengaged, FU2 repeats the THS position and FU1 repeats the motor position.

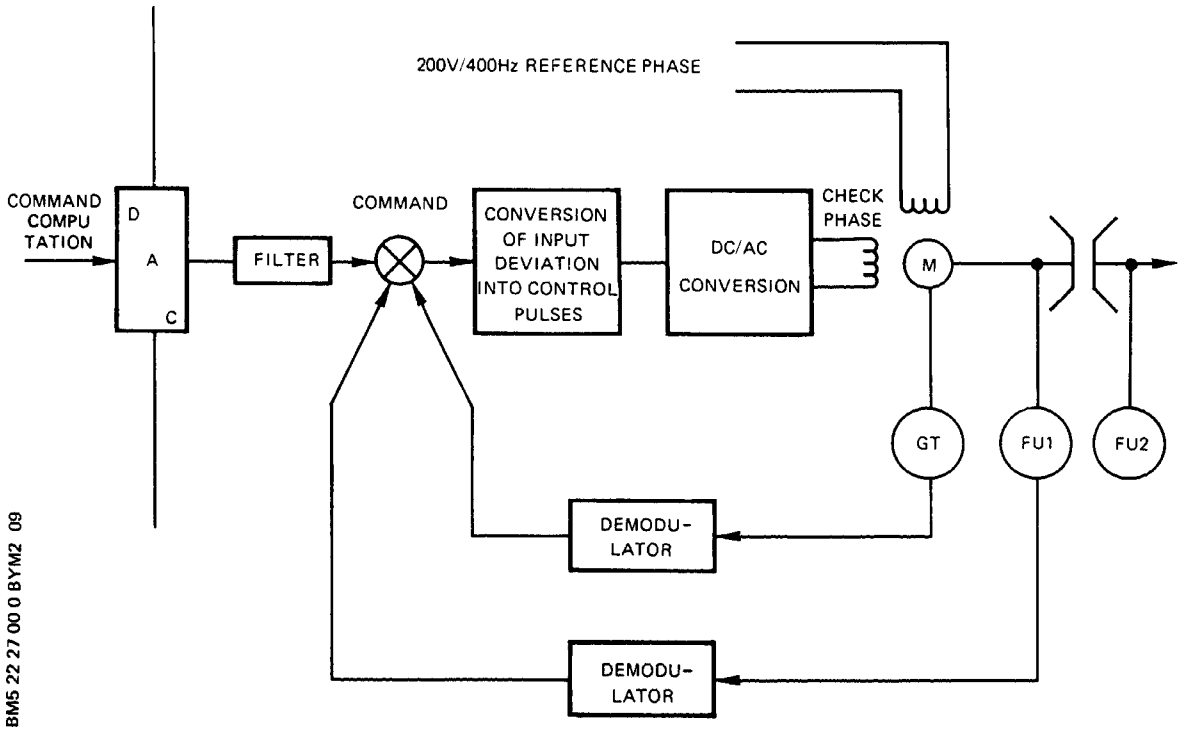
When a difference is detected between the two positions, a command is transmitted to the motor in order to cancel the difference between FU1 and FU2.

This difference can come from a THS position modification performed either by the pilot or by the other trim system.

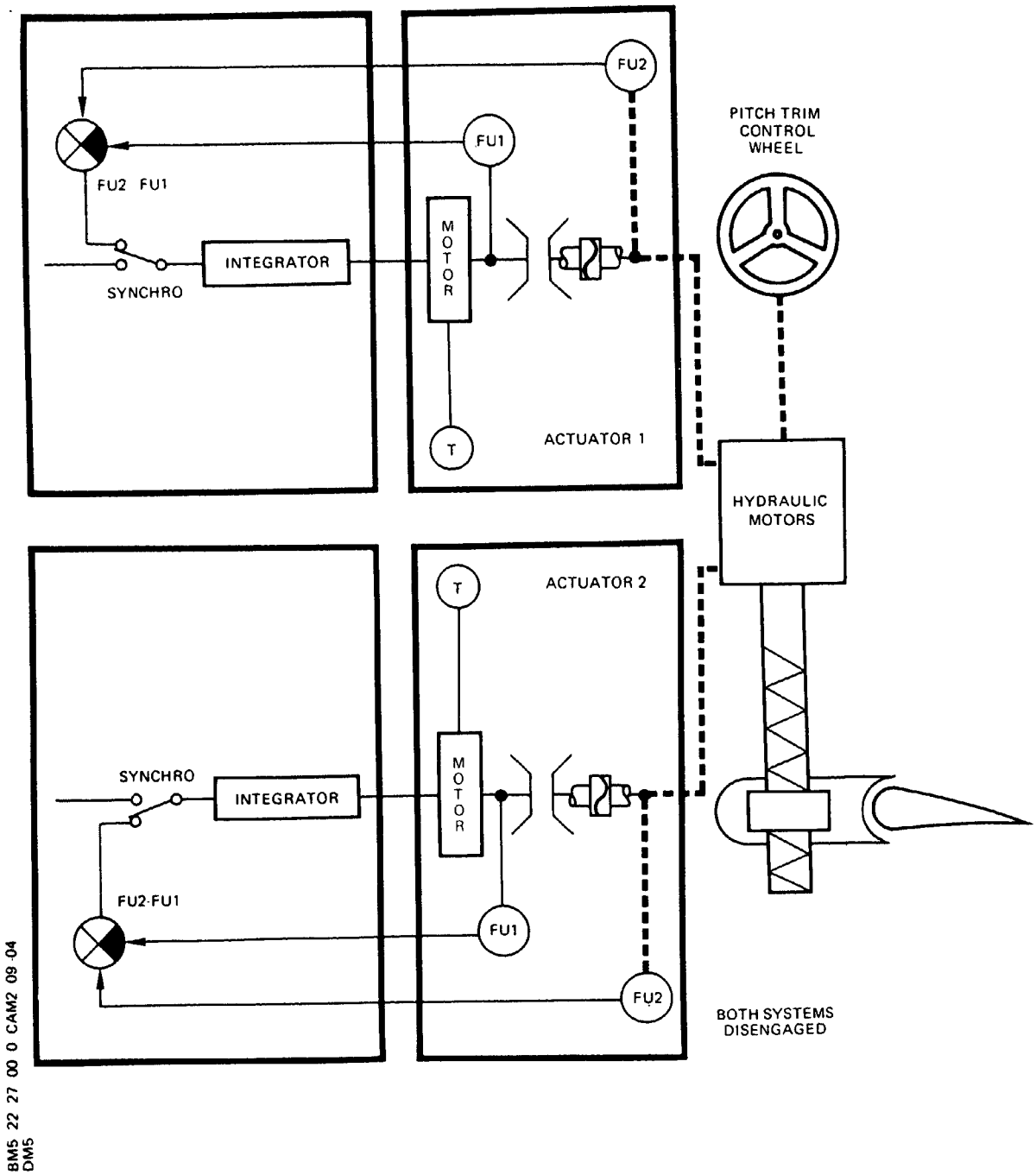
NOTE : The trim actuator operates even when the system is not engaged, for synchronization purposes.

(Ref. Fig. 014)

(Ref. Fig. 015)



Trim Power Loop Principle
Figure 014



Pitch Trim Synchronization
Figure 015

B. Power Loop Characteristics

(1)THS displacement range and position feedback gradients

The THS position which gives a zero FU synchro command corresponds to -5.1° THS (UP).

The THS displacement range on either side of this position is :

. -5.1° to +3° in nose down command that is a range of 8.1°

. -5.1° to -15° in nose up command that is a range of 9.9°.

The ratio between FU feedback and the actuator shaft is 1/112.

Relationship between synchro position and THS displacement is as follows :

- 14 turns of actuator correspond to 17° THS
- the synchro gradient is 530 mV/°THS
- 1° THS = 2.65° synchro
- trim command output gradient (measure in J3-C, J2-C)
  . 0.855 V for 1° THS
  . 0 V for -5.1° THS
  . positive for nose up direction.

(2)Actuator data

- check phase : 115 V/400 Hz, phase A and neutral
- reference phase : H = phase B
  C = phase C
- clutch signals : +28 V or open circuit (+28 V clutched).

(3)Power channel feedback signals

(a)Tachometric feedback signals

- 26 V/400 Hz energization with in-phase signal, displacement in nose up direction
- 3.6VAC gradient for 1° THS/s at FAC input
- gradient after demodulation (J3-C)
  . 69 mV/°THS/s
  . negative if nose up direction.

(b)FU1 and FU2 position feedback signals

- 26 V/400 Hz energization with in-phase signal with 26 V giving a nose up direction
- gradient at FAC input, 530 mV for 1° THS
- 0 V for -5.1° THS
- gradient after demodulation in FAC (measure in J3d for FU1 and J2d for FU2)
  . 0.879 V for 1° THS
  . negative for nose up direction.

R

1. Operational Test

A. TRIM Engage Logic

(1)Reason for the job

To check TRIM 1 priority over TRIM 2.

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart - Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Make certain that hydraulic power is available (GREEN)

NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps (Ref. 29-21-00, P. Block 301).

- Close overhead panel circuit breakers particularly those associated with AFS and ADS
- Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- Make certain that ADS is operating.
- Make certain that AP is not engaged.
- Make certain that pitch trim control wheel is in neutral position.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 and 2 levers in ON position.	Levers remain in engaged position.
2. On Captain control wheel	

ACTION	RESULT
- place pitch trim control switch in nose up position	Check that pitch trim control wheel rotates in nose up direction.
3. During pitch trim control wheel movement on overhead panel - place **PITCH TRIM 1** lever in **OFF** position.	Check that pitch trim control wheel still rotates in nose up direction.
4. Release pitch trim control switch. Place **PITCH TRIM 2** lever in **OFF** position.	
5. Place pitch trim control wheel in neutral position.	

(c)Close-up
1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

B. TRIM Disconnection Override

(1)Reason for the job

To check electrical **PITCH TRIM** override and disconnection (by manually stopping pitch trim wheel).

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart - Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00,

P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Make certain that hydraulic power is available (GREEN).

NOTE : As an alternative, it is possible to pressurize the hydraulic systems using Green system electric pumps (Ref. 29-21-00, P. Block 301).

- Close overhead panel circuit breakers particularly those associated with AFS and ADS
- Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- Make certain that ADS is operating.
- Make certain that pitch trim control wheel is in neutral position.

(b)Test

NOTE : The test described below concerns side 1. For side 2 which is identical, refer to information between parentheses.

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 (2) lever in ON position.	Lever remains in engaged position.
2. On Captain (First Officer) control wheel - place pitch trim control switch in nose down direction then in nose up direction (without reaching limits)	On center pedestal Pitch trim control wheel rotates in nose down direction then in nose up direction. WHOOLER aural warning sounds during pitch trim control wheel rotation.
3. On center pedestal - manually immobilize pitch trim control wheel during rotation and check that force required to immobilize it, is not excessive.	On overhead panel PITCH TRIM 1 (2) lever returns to OFF position.
4. Rotate pitch trim control wheel manually to neutral position.	

(c)Close-up

1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

C. Override Test

(1)Reason for the job

To check safety disengagement devices in pitch trim engagement (override and disconnection with engage lever held in engaged position).

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Make certain that hydraulic power is available (GREEN).

  NOTE : As an alternative it is possible to pressurize the hydraulic system using Green system electric pumps (Ref. 29-21-00, P. Block 301).
- Close overhead panel circuit breakers particularly those associated with AFS and ADS
- Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- Make certain that ADS is operating.
- Make certain that pitch trim control wheel is in neutral position.

(b)Test

R NOTE : The test described below concerns side 1. For side 2 which is
R identical, refer to information between parentheses.

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 (2) lever in ON position.	Lever remains in engaged position.
2. On Captain control wheel - place pitch trim control switch in nose up position.	Pitch trim control wheel rotates in nose up direction.
3. Manually override pitch trim control wheels, holding PITCH TRIM 1 (2) lever engaged.	After a few seconds, pitch trim control wheel is free to rotate (electric trim declutched).
4. Release PITCH TRIM 1 (2) lever	Lever returns to OFF position.

ACTION	RESULT

5. Place pitch trim control wheel in neutral position.

(c)Close-up
1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

D. PITCH TRIM 1 Disengagement by AP Pitch Torque Limiter Test

(1)Reason for the job

To check PITCH TRIM disengagement when overriding auto-pilot (by pushing firmly control column).

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-33-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Make certain that hydraulic power is available (GREEN, YELLOW).
  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00).
- Close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS.
- Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- Make certain that ADS is operating.

- Align IRS 1, 2 and 3.
- Make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU.
- On center pedestal, adjust ECAM display unit brightness and select FLT CTL page.
- Close circuit breaker 5RK on panel 22VU (line 204/F27).
- Make certain that pitch trim control wheel is in neutral position.
- On glareshield, place FD/FPV selector switches in OFF position.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 lever in ON position - place YAW DAMPER 2 lever in ON position	Levers remain in this position.
2. On flight control unit (FCU) - place AP1 lever in ON position	Lever remains in this position. On FCU : - green CWS comes on. On PFDs : - amber CWS 1 indication appears
- press CWS/CMD pushbutton switch	On this pushbutton switch : - green CWS goes off - green CMD comes on. On PFDs : - white CMD 1 indication appears - amber CWS 1 indication disappears - green V/S and HDG indications appear
NOTE : If AP not in V/S mode, on FCU - select altitude of 5000 ft - pull V/S knob.	
- using V/S selector knob, select positive vertical speed (Vz > 300 ft/mn)	Control column moves in nose up direction. Pitch trim control wheel rotates in nose up direction.
3. Move control column in nose down position	Check that, past load threshold (20 daN approx.) ; On ECAM FLT CTL system page : - ELEV scale reflects control column movement - STAB scale reflects pitch trim control wheel rotation. On FAC/ATS engage unit : - PITCH TRIM 1 lever returns to OFF position. (After some delay).

ACTION	RESULT
	On FCU - AP1 lever returns to OFF position.

4. Release control column and cancel warning.

(c)Close-up
1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

E. Test of PITCH TRIM 2 operation in single channel mode

(1)Reason for the job

To check correct operation of PITCH TRIM 2 when engaged.

(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Make certain that hydraulic power is available (GREEN).

  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps (Ref. 29-21-00, P. Block 301).
- Close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS.
- Close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- Make certain that ADS is operating.

- Make certain that AP is not engaged.
- Make certain that pitch trim control wheel is in neutral position.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 2 lever in ON position	Lever remains in engaged position.
2. On Captain or First officer control wheel - place pitch trim control switch in nose down direction then in nose up direction (without reading limits)	On center pedestal Pitch trim control wheel rotates in nose down direction then in nose up direction. WHOOLER aural warning sounds during pitch trim control wheel rotation.
3. On overhead panel - place PITCH TRIM 2 lever in OFF position.	

(c)Close-up

1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

F. Supply of FAC 1 in Ultimate Emergency

(1)Reason for the job

To check correct operation of PITCH TRIM 1 with standby generator.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 24-24-00, P. Block 501	Emergency AC Generation
- 29-21-00, P. Block 301	Green Auxiliary Power

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED. MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

CAUTION : MAKE CERTAIN THAT PROTECTIVE COVERS ARE REMOVED FROM PROBES.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

- Make certain that electronics racks ventilation is correct.
- Operate the two green hydraulic electric pumps by pressing ELEC PUMPS pushbutton switch on panel 427VU (Ref. 29-21-00, P. Block 301).

NOTE : On FUEL section of panel 430VU, make certain that fuel PUMP pushbutton switches are released (out) (OFF legend on). On panel 427VU, make certain that PTU pushbutton switches are released (out).

- Close overhead panel circuit breakers particularly those associated with AFS NAVIGATION and ADS.
- Make certain that LDG GEAR circuit breakers are closed (aircraft in ground configuration).
- Make certain that pitch trim control wheel is in neutral position.
- Close AFS/FAC1 28VDC/STBY (322CC1, 106VU line A23) circuit breaker.

NOTE : On ground, standby generator stops normally after 2 minutes by automatic release of STBY GEN pushbutton switch (Ref. 24-24-00, P. Block 501).

(b)Test

ACTION	RESULT
1. On place 21VU, - place PITCH TRIM 1 lever in ON position	Lever remains in this position.
2. On panel 424VU, - Press STBY GEN pushbutton switch	On panel 424VU, - OVRD legend on STBY GEN pushbutton switch comes on. - AC EMER ON INV and DC ESS ON BAT warning lights are off. On panel 21VU, - PITCH TRIM 1 engage lever remains engaged.
3. On panel 21VU, - open AFS/FAC1/28VDC/NORM circuit breaker (306 CC1 line 103/G3)	No change.
4. On panel 21VU, - open AFS/FAC1/28VDC/BAT circuit breaker (323 CC1 line 103/G2) - close circuit breaker 323CC1.	No change
5. On panel 106VU - Open AFS/FAC1/28VDC/STBY circuit breaker (322 CC1 line A23). - Close circuit breakers 306CC1 and 322CC1.	No change
6. Disengage PITCH TRIM 1 lever	

ACTION	RESULT
7. On panel 424VU - Press and release STBY GEN pushbutton switch.	On panel 424VU - OVRD legend of STBY GEN pushbutton switch goes off STBY generator stops

(c)Close-up
1 Press and release green ELEC PUMPS pushbutton switch on panel 427VU (Ref. 29-21-00, P. Block 301).
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.

R G. Detailed Inspection of the Autotrim Function
(1)Reason for the job
To check that the electrical pitch trim is not active in autotrim mode.
(2)Equipment and materials

ITEM	DESIGNATION
(1)	Cart-Hydraulic Ground Power Supply (Aircraft - Type Pump)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure
WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN, YELLOW)
NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps and PTUs (Ref. 29-21-00, 29-23-00, P. Block 301).
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that ADS is operating

- align IRS 1, 2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD and ND image brightness on panels 412VU and 413VU
- on Captain and First Officer EFIS control panels, make certain that mode selector switch is in ROSE position
- on overhead panel, make certain that PITCH TRIM 1, YAW DAMPER 2 are engaged
- make certain that slats are extended
- close circuit breaker 5RK (panel 22VU, line 204/F27).
- on glareshield, place FD/FPV selector switches in ON position.

(b)Test

NOTE : The phenomenon given below could be seen :
- with AP1 or AP2 in CMD and P/T1 or P/T2 engaged, the pitch trim wheel could 'creep' slowly,
- however, the pitch trim switches activation with the autopilot in CMD will not modify this phenomenon.

ACTION	RESULT
1. On Flight Control Unit (FCU) : - place AP1 lever in ON position	On Flight Control Unit (FCU) : - lever remains in this position - green CWS comes on. On CAPT and F/O PFDs : - amber CWS1 indication appears.
- press CWS/CMD pushbutton switch	On this pushbutton switch : - green CWS goes off - green CMD comes on. On CAPT and F/O PFDs : - white CMD1 indication appears - amber CWS1 indication disappears - green V/S and HDG indications appear.
R NOTE : If AP not in V/S mode, On FCU R - select altitude of 5000 ft R - pull V/S knob. R R - using V/S selector knob, select positive vertical speed (Vz > 300 ft/mn)	
2. On Captain or First Officer control wheel : - place pitch trim control switch in any position.	On center pedestal : - pitch trim control wheel does not move.
3. On overhead panel, on FAC/ATS engage unit : - place PITCH TRIM 2 lever in ON position	PITCH TRIM 2 lever remains in this position.

ACTION	RESULT
- place **PITCH TRIM 1** lever in **OFF** position.	
4. On Captain or First Officer control wheel : - place pitch trim control switch in any position.	On center pedestal : - pitch trim control wheel does not move.
5. On overhead panel, on **FAC/ATS** engage unit : - place **PITCH TRIM 1** lever in **ON** position - place **PITCH TRIM 2** lever in **OFF** position - place **AP1** lever in off position and cancel warning.	**PITCH TRIM 1** lever remains in this position.
6. On Flight Control Unit (FCU) : - place **AP2** lever in **ON** position	On Flight Control Unit (FCU) : - lever remains in this position - green **CWS** comes on. On **CAPT** and **F/O PFDs** : - amber **CWS 2** indication appears.
- press **CWS/CMD** pushbutton switch.	On this pushbutton switch : - green **CWS** goes off - green **CMD** comes on. On **CAPT** and **F/O PFDs** : - white **CMD2** indication appears. - amber **CWS2** indication disappears. - green **V/S** and **HDG** indications appear.
7. On Captain and First Officer control wheels : - place pitch trim control switch in any position.	Pitch trim control wheel does not move.
8. On overhead panel, on **FAC/ATS** engage unit : - place **PITCH TRIM 2** lever in **ON** position - place **PITCH TRIM 1** lever in **OFF** position.	**PITCH TRIM 2** lever remains in this position.
9. On Captain or First Officer control wheel - place pitch trim control switch in any position.	On center pedestal : Pitch trim control wheel does not move.

ACTION	RESULT
10. On overhead panel, on FAC/ATS engage unit : - place PITCH TRIM 1 lever in ON position. - place PITCH TRIM 2 lever in OFF position. - place AP2 lever in OFF position and cancel warning.	- PITCH TRIM 1 lever remains in this position.
11. Place PITCH TRIM 1 lever in OFF position.	

(c) Close-up
1 Shut down and disconnect hydraulic ground power cart.
2 De-energize the aircraft electrical network (Ref. 24-41-50, P. Block 301).
3 Restore aircraft and system to normal operating condition.

Pitch Trim Control Switch 7CC (8CC) - REMOVAL/INSTALLATION

1. Equipment and Materials

ITEM	DESIGNATION
A.	Cart - Hydraulic Ground Power Supply (Aircraft - Type Pump)
B.	Circuit Breaker Safety Clips
C. MS275-34-22D	Insertion/Extraction Tool
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power

(Ref. Fig. 401)

2. Procedure

A. Job Set-Up

(1)Open, safety and tag the following circuit breakers :

	PANEL	SERVICE	IDENT.	LOCATION
	21VU	AFS/FAC1/28VDC	306CC1	103/G 3
	21VU	AFS/FAC2/28VDC	306CC2	103/G 12
R	132VU	INTERIOR LT-F/O SIDE	52LE	321/P 78
R	132VU	INTERIOR LT-FLOOD CAPT SIDE	4LE	321/P 77

R B. Removal

R (1)Remove chartholder and suspend from Captain's (First Officer's) control wheel to avoid distress to chartholder wires.

R (2)Remove cover plate (1) from Captain's (First Officer's) control wheel.

R (3)Remove screws (2), support ring (3) and cut ties attaching loom to support ring.

R (4)Disconnect electrical wires from electrical connectors 783VCA (4) and 75VCA (5) (782VCA and 74VCA) using insertion/extraction tool (white end).

R (5)Remove screws (7) securing control switch (6) to control wheel.

R (6)Remove control switch (6) with electrical wiring.

R C. Installation (Ref. Fig. 402)

R (1)Slide control switch (6) wiring into control wheel horn.

R (2)Secure control switch (6) to control wheel horn with screws (7).

R (3)Remove and discard clear shrink tube to expose wire pins, being careful not to damage the wire insulation.

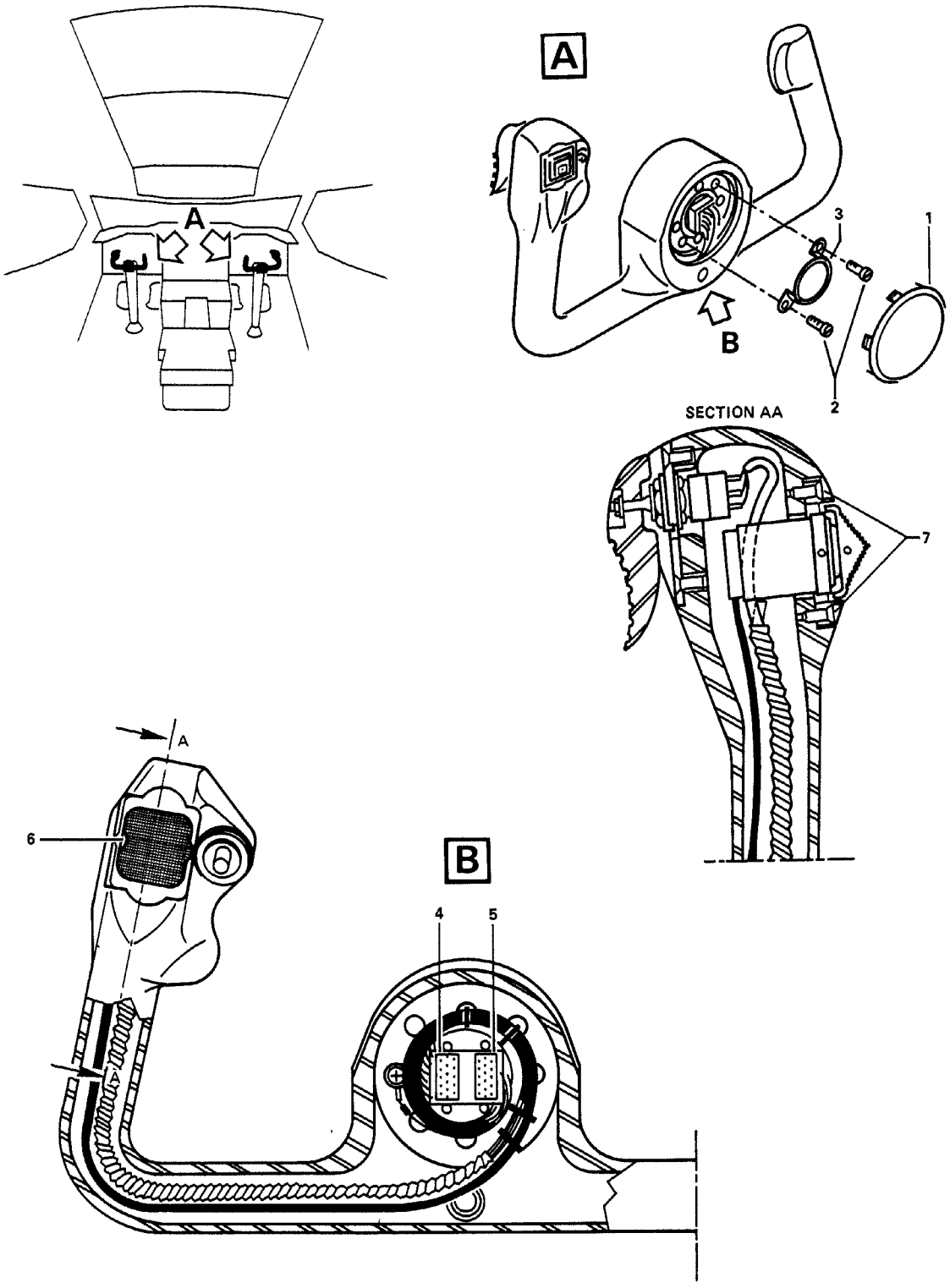
R (4)Connect control switch electrical wires to corresponding connectors (4) and (5) using green end of insertion/extraction tool.

R (5)Attach electrical wiring with cable ties to support ring.

R (6)Install cover plate (1) on Captains's (First Officer's) control wheel.

R (7)Attach chartholder to mount.

R (8)Remove safety clips and tags and close CB's opened in job set-up



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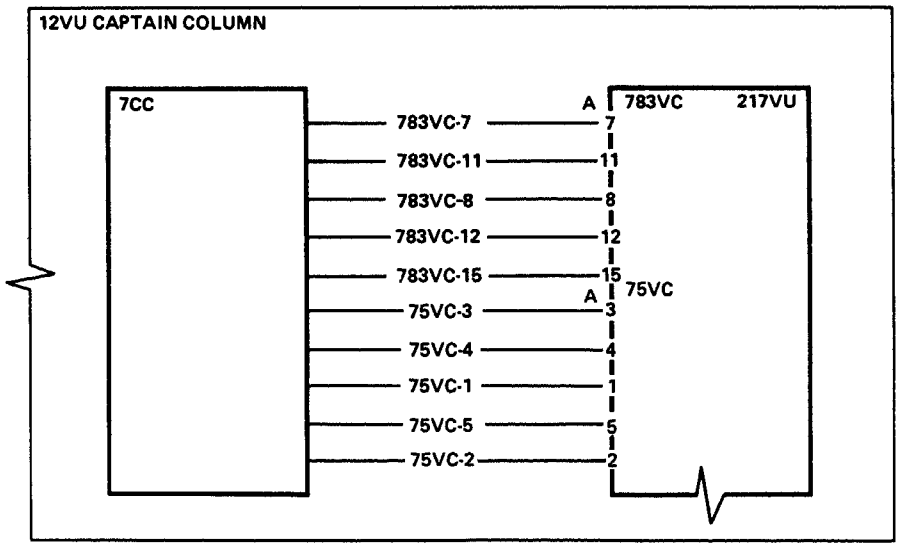
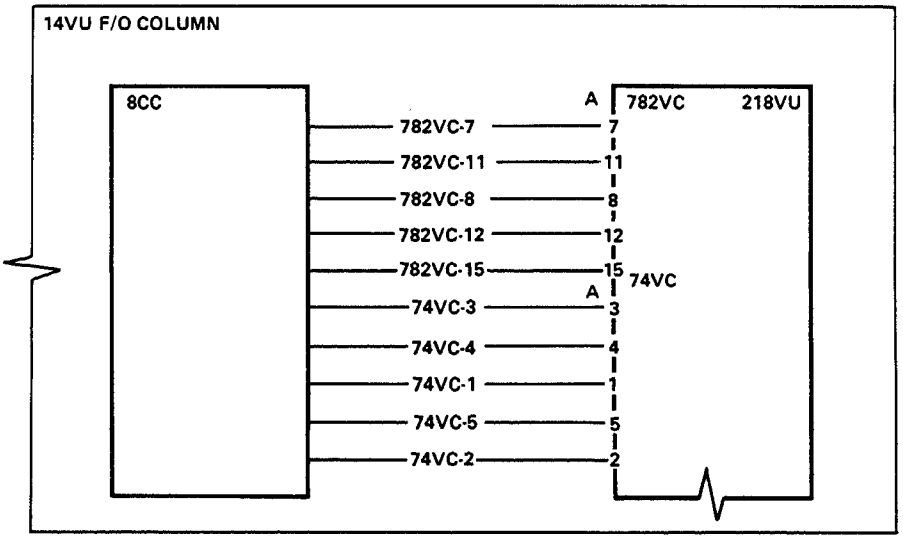
Pitch Trim Control Switch
Figure 401





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Pitch Trim Control Switch - Electrical Wiring
Figure 402

D. Test

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(1)Initial conditions

- correct electrical power supply
- forward services ventilation system in operation
- hydraulic power available (GREEN) (Ref. 29-10-00, P. Block 301).
  NOTE : As an alternative it is possible to pressurize the hydraulic systems using Green system electric pumps (Ref. 29-21-00, P. Block 301).
- overhead panel circuit breakers closed, particularly those associated with AFS and ADS
- aircraft in ground configuration (L/G PROX DET & RELAYS circuit breakers on panel 133VU closed)
- ADS operating
- AP not engaged
- pitch trim control wheel in neutral position.
  NOTE : The following test concerns PITCH TRIM 1.
  For PITCH TRIM 2 test, which is identical, refer to information given in brackets.

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1(2) lever in ON position.	
2. On Captain's control wheel (First Officer's control wheel) - place and hold pitch trim control switch 7CC (8CC) in nose down position.	On center pedestal - pitch trim control wheels rotate to nose down position - whooler aural warning sounds during pitch trim control wheel rotation.
3. On First Officer's control wheel (Captain's control wheel) - place and hold pitch trim control switch 8CC (7CC) in nose up position.	Pitch trim control wheels stop rotating.
4. Release both pitch trim control switches 7CC and 8CC.	
5. On Captain's control wheel (First Officer's control wheel) - place and hold pitch trim control switch 7CC (8CC) in nose up	On center pedestal - pitch trim control wheels rotate to nose up position - whooler aural warning sounds during pitch trim control wheel

ACTION	RESULT
position.	rotation.
6. On First Officer's control wheel (Captain's control wheel) - place and hold pitch trim control switch 8CC (7CC) in nose down position.	Pitch trim control wheels stop rotating.
7. Release both pitch trim control switches 7CC and 8CC.	
8. Return pitch trim control wheels to 0° position.	
9. On overhead panel - place PITCH TRIM 1(2) lever in OFF position.	

E. Close-Up
(1) Depressurize Green auxiliary hydraulic power (Ref. 29-10-00, P. Block 301 and 29-21-00, P. Block 301).
(2) De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
(3) Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

STANDBY ALPHA PROBE (27CC) - REMOVAL/INSTALLATION

1. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 2.6 m. (8 ft. 6 in.)
B.	CIrcuit Breaker Safety Clips
C.	Blanking Caps
D.	O-Ring
R E. Material No. 09-029	Sealants (Ref. 20-31-00)
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 30-31-00, P. Block 501	Probe Ice Protection

(Ref. Fig. 401)

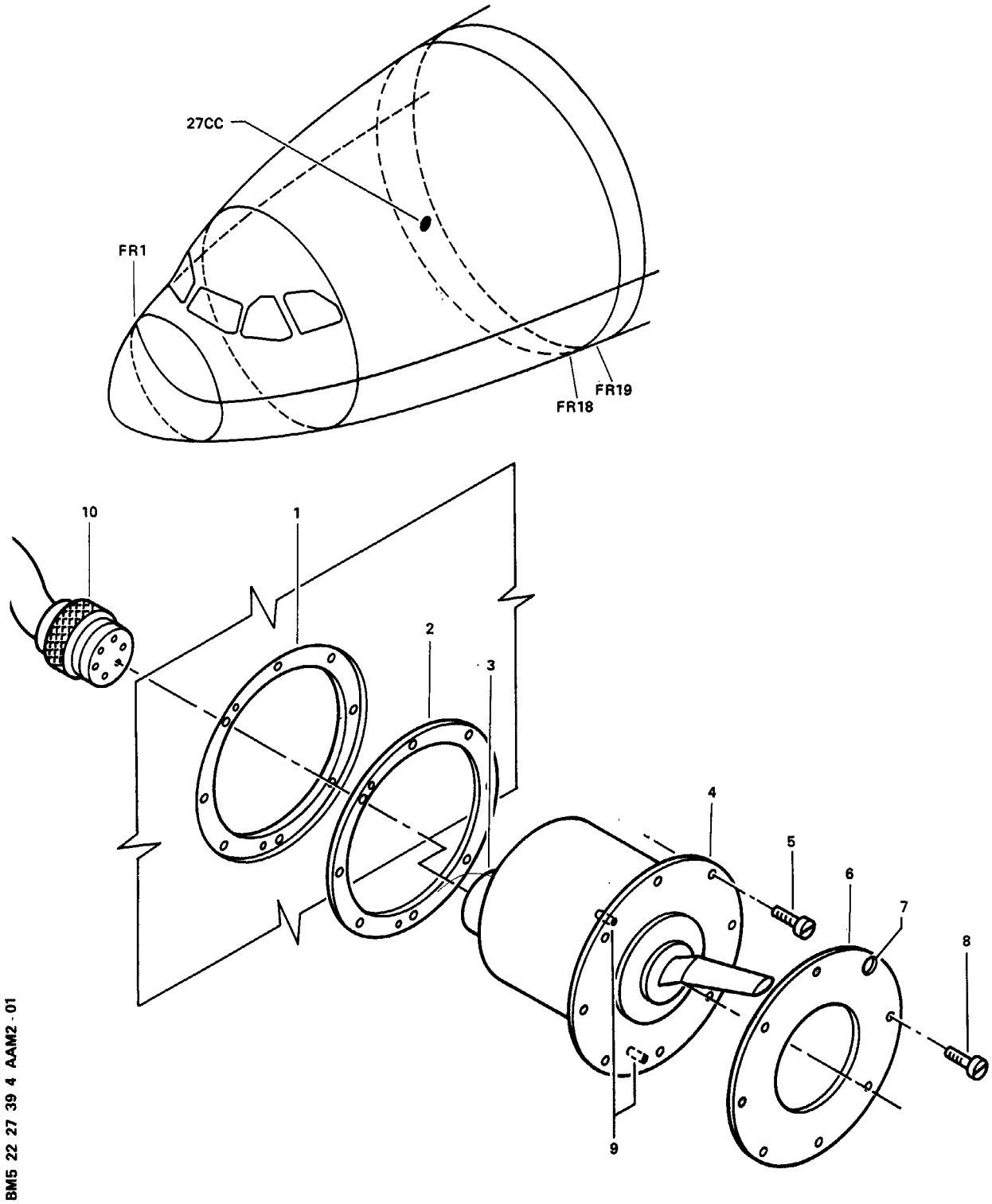
2. Procedure

A. Job Set-Up
(1)Position access platform.
(2)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	ADS/ADC AND ALPHA PROBES TEST/PWR	28FL	105/E07
21VU	AFS/FAC1/26VAC	305CC1	103/G 7
21VU	AFS/FAC2/26VAC	305CC2	103/G 8
132VU	ANTI-ICE/PROBES/STBY ALPHA	66DA	325/K68

B. Removal
(1)Remove the seven attaching screws (8) from blanking plate (6) and remove it.
(2)Remove the attaching screw (5) from probe (4).
(3)Disengage probe (4) until connector (3) is visible and accessible.
(4)Disconnect and blank off connectors (3) and (10).
(5)Remove probe (4).
(6)Remove and discard O-ring (2).

C. Installation
(1)Clean and inspect probe interface blanking plate and/or adjacent area.
(2)Install new O-ring (2) in its recess at end of counterbore (1).
(3)Remove blanking caps from connectors (3) and (10) and carry out a visual check of pins.
(4)Connect electrical connectors.
(5)Install alpha probe (4), correctly positioning by guide pins (9), and tighten screw (5).
(6)Install blanking plate (6), and tighten the seven screws (8).
R (7)Blank the eighth hole (7) off with Sealant (Material No. 09-029).
R (8)Apply a bead of Sealant (Material No. 09-029) around blanking plate (6).
R NOTE : If cure time of sealant will cause a flight delay, the aircraft



Standby Alpha Probe
Figure 401

R

R can operate without the sealant for a maximum period of
R 24 hours from installation of the probe.
R Before application of sealant, it is necessary to make certain
R that surfaces are clean and in the correct condition (no moisture,
R dust, etc.).

(9)Check the bonding contact resistance between the sensor (4) and the fuselage structure (1) (Ref. 20-28-11, P. Block 001).

(10)Remove safety clips and tags and close circuit breakers 66DA, 28FL, 305CC1 and 305CC2.

D. Test

Initial conditions :
- Correct electrical power supply.
- Forward services ventilation system in operation.
- Overhead panel circuit breakers closed, in particular those associated with the ADS.
- Aircraft in ground configuration (LDG GEAR circuit breaker closed).

ACTION	RESULT
1. On ALPHA PROBES section of maintenance panel 470VU - press ALL pushbutton switch for more than 60 seconds	On this pushbutton switch : - TEST legend comes on. - Captain and First Officer stick-shakers are activated Aural warning sounds On CAPTAIN SWITCHING panel 402VU - GPWS fault light comes on. Check that alpha probe in zone 122 moves.
2. Press to release ALL pushbutton switch	- TEST legend goes off. - Captain and First Officer stick-shakers stop vibrating - Aural warning stops.
3. Perform test of ice protection system (Ref. 30-31-00, P. Block 301).	

E. Close-Up

(1)De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

(2)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

(3)Remove access platform.

(4)Restore system and aircraft to normal operating condition.

1. General

The airspeed data are no longer given by an airspeed indicator but displayed on the CRTs (cathode-ray tube) managed by the EFIS (electronic flight instrument system).
The FAC generates several data signals dealing with flight envelope protections and expressed in speed parameters. These data are computed in the monitoring channel only (computation is not duplicated).
Airspeed information is displayed on the PFD (primary flight display) on a speed scale on the left part of the image. Each item of information is always displayed at the same place on the image and using the same symbol.
The PFD uses the following colors :
- white for scale display
- red for flight envelope limits (Vss, V max)
- green for some active values (maneuvering speed, V3, V4)
- amber for information requiring particular attention from the pilot (V ms).

In the event of a FAC information failure all the speed scale displays are cleared to avoid any confusion and a red SPD LIMITS failure warning message appears. This message flashes during a few seconds then remains visible until the system is switched over to another source (FAC2).

2. Airspeed Information Display

A. General

The airspeed is indicated by a white scale which moves in front of the fixed computed airspeed symbol.
R The display range is 80 kts with a stop at 30 kts (half scale).
R All the data delivered by the FAC are displayed on this scale
R (Ref. Fig. 001).
Airspeed information shall be displayed on the EFIS instruments only if they are valid and depending on the flight phase requiring this type of
R information.

B. Information Validation (Ref. Fig. 002)

The various airspeed parameters transmitted through EFIS line are monitored by the FAC under certain conditions depending on the computation which is performed.
The monitoring result is given at the level of the status matrix associated with each parameter :
- failure warning : in the event of internal computation failure or in the event of a failure of the detector taking part in the computation (SFCC, ADC)
  NCD = on ground and in flight in some cases (see table).

At the EFIS level the processing logic is as follows :
- on NCD concerning one parameter, this parameter is not displayed on the speed scale
- on F/W concerning any parameter delivered by the FAC, all the FAC information signals are cleared and the SPD LIMITS message appears at the bottom of the speed scale.

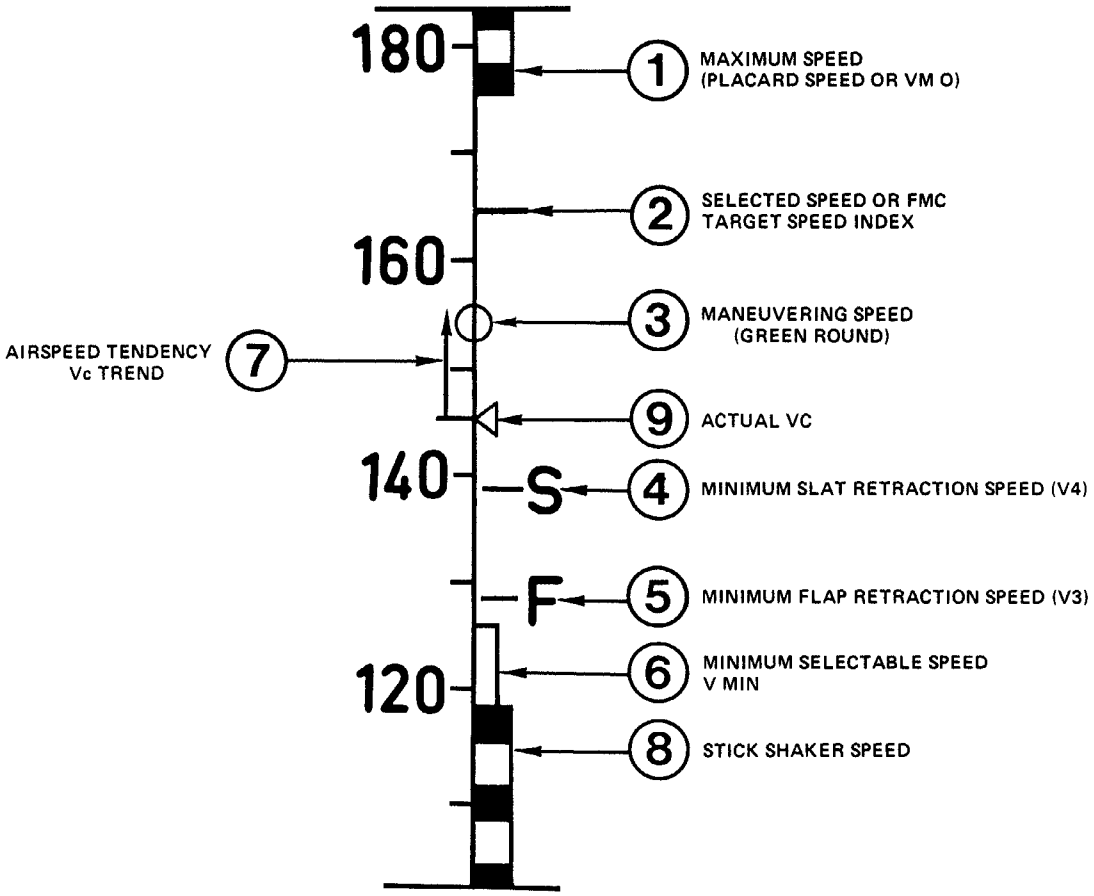
C. Symbol Display Principle

The airspeed parameters are displayed during the flight phase when they are required and cleared in the event of computation failure or impossibility.

Then all the following parameters using angle-of-attack value cannot be computed on the ground :

- Vss
- V min
- maneuvering airspeed
- V3
- V4

All these parameters are cleared on the ground.



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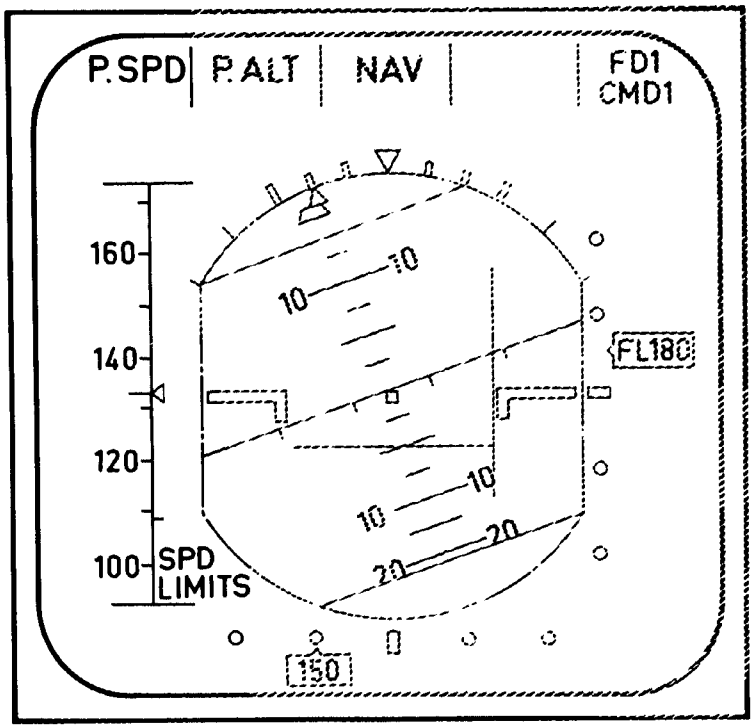
PFD - Data Delivered by the FAC
Figure 001



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FAC Failure
Figure 002

ON EFIS BUS AND FMC 12 K
TABLE 1

	PARAMETER LABEL	STATUS MATRIX	NORMAL OPERATION	NO COMPUTED DATA	FAILURE WARNING
R	VMS 245		5 s after lift off	On ground	FAC failure Peripheral failure - OWN ADC (alpha, Zp, Vc) - OWN IRS (AZ1) - OWN SFCC
R	VM 265		S/F sel pos = 1 in flight	On ground S/F sel pos ≠ 1 or own ADC failure (Vc, Zp)	Fac failure
	Vss 256		5 s after lift off	- On ground - If peripheral failure . Own ADC (Zp, alpha, Vc) . Own SFCC . Own IRS (AZ1)	FAC failure
R	V3 263		5 s after lift off if S/F sel pos = 3 or 4	- ditto or if S/F sel pos = 1, 2, 5	FAC failure
R	V4 264		5 s after lift off if S/F sel pos = 2	- ditto Vss or if S/F sel pos = 1, 3, 4, 5	FAC failure
	Vc TREND 262		Always	- If peripheral failure Own ADC (Vc)	FAC failure
	VmO --- Vfe Vle ___	VMAX 207	Clean configuration and landing gear retracted Slats or flaps extended Clean configuration and landing gear extended	In flight if peripheral failure : - Own ADC (VmO) - Own SFCC	FAC failure and on ground if peripheral failures - Own ADC (VmO) - Own SFCC

D. Speed Parameters - Definition

(1)Airspeed tendency (ΔVc) (Ref. Fig. 001)

The airspeed tendency is indicated by a yellow pointer initiating in the computed airspeed.
This information is the filtered derivative of the computed airspeed (Vc).
- Direction of the pointer indicates the aircraft airspeed tendency.
- Amplitude represents the airspeed value which will be attained in 10 s if the acceleration remains constant.

(a)Display

As soon as amplitude is greater than a dissymmetrical threshold :
- 2 kt when the vector increases
- 1 kt when it decreases.

This hysteresis avoids flashing around the threshold.

(b)Inhibition

Conditions : - in-flight NCD or speed failure : airspeed tendency disappears
- FAC failure : a red line is displayed across the computed airspeed symbol and SPD LIMITS message is visible.

(2)Limit airspeeds

The FAC generates limit airspeed data (V min, Vss, V max) within the framework of the flight envelope protection function. These indications appear on the PFD and disappear in the event of computation failure.
The SPD LIMITS message is then displayed at the RH bottom of the scale.

(a)Minimum selectable airspeed (V min)

V min is the minimum selectable airspeed for a given configuration.
It enables to determine a minimum value giving a certain margin with respect to low speed stall and cruising speed buffeting.
The following table shows V min significance.

		VMS
	GROUND CONFIGURATION	1.2 Vs
Z < 25,000 ft.	FLIGHT CONFIGURATION	1.2 Vs not actuating F/S selector 1.3 Vs actuating F/S selector
Z > 25,000 ft. in clean configuration	0.3 g/buffeting	

The minimum selectable airspeed is represented by an amber strip against the airspeed scale (Ref. Fig. 001).
This item of information is inhibited :
- beyond the speed scale limits
- in the event of NCD or F/W code on the parameter
- at takeoff and up to 5 s after shock absorber extension (and then instantaneous display).

(b)Stick shaker speed (Vss) (Ref. Fig. 001)

This is the speed at which the stick shaker is activated.

R The Flight Warning Computers (FWC) generate the Stall Warning
R according to the slat configuration and angle of attack.

Vss is represented by a red and black checkered tape.

This symbol is inhibited :

- outside speed scale limits
- at takeoff up to 5 s after closure of landing gear shock absorber relay contact
- upon NCD or F/W code of parameter considered.

(c)Maximum allowable airspeed (V max) (Ref. Fig. 001)

V max is the maximum allowable airspeed for a given configuration. It enables to determine a maximum value with respect to the normal flight envelope. It has various significations :

1 Slats = 0 and landing gear retracted,
V max = VMO

VMO (maximum operating limit speed) is generated by the ADS and takes account of MMO which is converted into speed as a function of altitude.

2 Slats = 0 and landing gear extended,
V max = VLE

VLE means maximum landing gear extended speed (270 kt).

3 Slats ≠ 0
V max = VFE

VFE means maximum flap and slat extended speed.

V max is represented by a red and black checkered tape.

This symbol is inhibited :

- outside speed scale limits
- in the event of parameter NCD or F/W code
- at takeoff up to 5 s after closure of landing gear shock absorber relay contact.

(3)Maneuvering speeds (Ref. Fig. 003)

Some parameters generated by the FAC and displayed on the PFD represent maneuvering speed data.

These data deal with normal and limit speeds for flap and slat extension or retraction.

The following data are displayed :

- V3 : minimum flap retraction speed
- V4 : minimum slat retraction speed
- VM : maneuvering speed.

These are represented by green symbols which disappear when the values are outside the speed scale limits (zones not displayed).

In the event of a failure, the SPD LIMITS message appears at the right part of the scale.

(a)V3

This is the MINIMUM flap retraction speed. It corresponds to 1.25 Vs of slats 15°/flaps 0° configuration.

It is represented by a green -F symbol which moves along the speed scale.

The flaps can be retracted when the computed airspeed (circle) is greater than V3, that is when -F symbol lies below Vc symbol.

-F symbol is not displayed when :

- the aircraft is on the ground
- the flaps are retracted
- the flaps are fully extended
- the parameter is assigned an NCD or F/W code.

Computation is made from the flap and slat control lever position and display is made according to true position.

In S/F sel pos = 4 position, V3 represents the value memorized at previous notch.

(b)V4

This is the MINIMUM slat retraction speed. It corresponds to 1.25 Vs of clean configuration.

It is represented by a green -S symbol which moves along the speed scale.

The slats can be retracted when the computed airspeed is greater than V4, that is when -S symbol lies below Vc symbol (circle).

-S symbol is not displayed when :

- the aircraft is on the ground
- the flaps are extended
- if the parameter is assigned an NCD or F/W code.

(c)V3 or V4 utilization

The minimum flap and slat retraction speed is in fact the speed to be reached before slat or flap retraction.

It is displayed :

- at takeoff (F)
- during landing phase to overcome untimely retraction of control surfaces in a go around maneuver.

Utilization of V3 and V4 is shown in figure (Ref. Fig. 003).

(d)Maneuvering speeds (Vm) (Ref. Fig. 001)

R This is the Drift Down Speed which is the maximum speed with one engine
R running up to 20,000 ft., its value depends on the aircraft weight

R only.
Above 20,000 ft. previous computation will be increased by 2 kts/1000 ft. up to M = 0.8.
On PFD, Vm is represented by a green circle as soon as aircraft is in clean configuration in flight.

(e) Control surface jamming - airspeed data

1 General

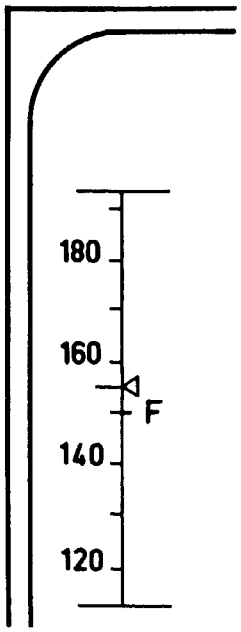
Airspeed data, especially Vms, become important in the event of control surface (flaps and slats) jamming : a corrective action must then be taken and the displayed information must be in line with the procedure.
To this end, information is computed from true position data of control surfaces. In addition specific precautions are to be taken in the event of jamming.

2 Information

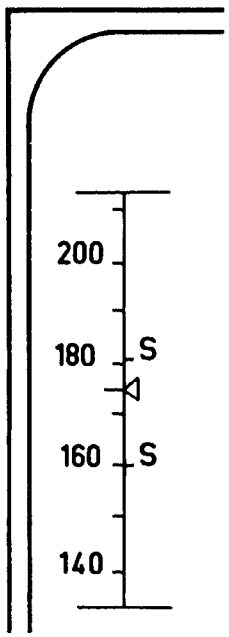
It is assumed that simultaneous jamming of flaps and slats is unlikely to happen.
In consequence :
- if the slats are jammed, the flaps can be extended on all the positions
- if the flaps are jammed, the slats, although they are not jammed, are not actuated.

The computation ensures the following :
- in the event of slat jamming only (S/F sel pos 2) (flaps not extended) V min computation is performed by taking the true position of the slats into account.
- in the event of flap jamming only, Vms computation is always made by taking the true position of the flaps into account.
- in the event of slat jamming with a flap position other than zero. Vms computation is made as follows :
  Vms = V Ref + KV
  KV is a function of the true position of the slats varying in a linear way from 0 to 19.5 kt.

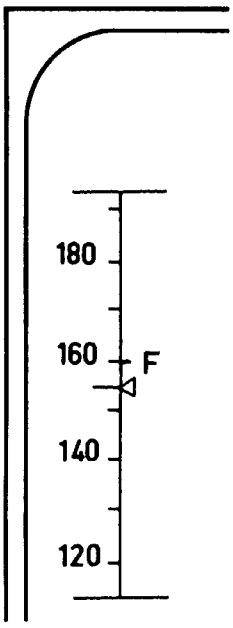
This law is based on the jamming detection processed by the FAC. This detection consists in comparing the flap and slat control lever position with the true position of the flaps and slats within 15 s approximately. In the event of jamming, V3 and V4 are displayed. (Ref. Fig. 003)



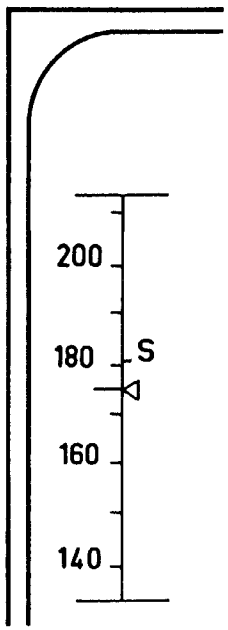
VC = 155kt
V3 = 150kt
FLAP RETRACTION AUTHORIZATION
VC > V3



VC = 175kt
V4 = 160kt
SLAT RETRACTION AUTHORIZATION
VC > V4



VC < V3
FLAP RETRACTION
INTERDICTION



VC < V4
SLAT RETRACTION
INTERDICTION

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V3 and V4 - Display and Function
Figure 003

3. Alpha-Floor and Windshear Protection

Alpha-floor and windshear protection is provided by the FACs which send a signal of THR mode engagement to the TCC which ensures thrust demand up to THR limit if these modes are previously ensured.
The angle-of-attack values which cause AT engagement in THR mode depend on :
- aircraft configuration
- presence of windshear (S/F sel pos ≥ 2).

SLATS/FLAPS SELECTOR POSITION	1	2,3,4	5
ANGLE OF ATTACK VALUE	8.5°	14.5°	11.5°

In windshear conditions the principle is to reduce the detection threshold according to the detected windshear in order to get the possibility of performing a go around maneuver sooner. Windshear detection inhibits alpha-floor function during 30 s at takeoff.
The Logic Associated with Alpha-Floor Protection is inhibited below 100ft. (R/A data) to avoid untimely activation during final landing phase.
This logic can however be activated and tested on the ground by the alpha probe test (1+2+3).
NOTE : Engagement of the TCC in THR mode is not possible unless it has been armed.

(Ref. Fig. 004)
(Ref. Fig. 005)

4. Windshear Detection and Warning

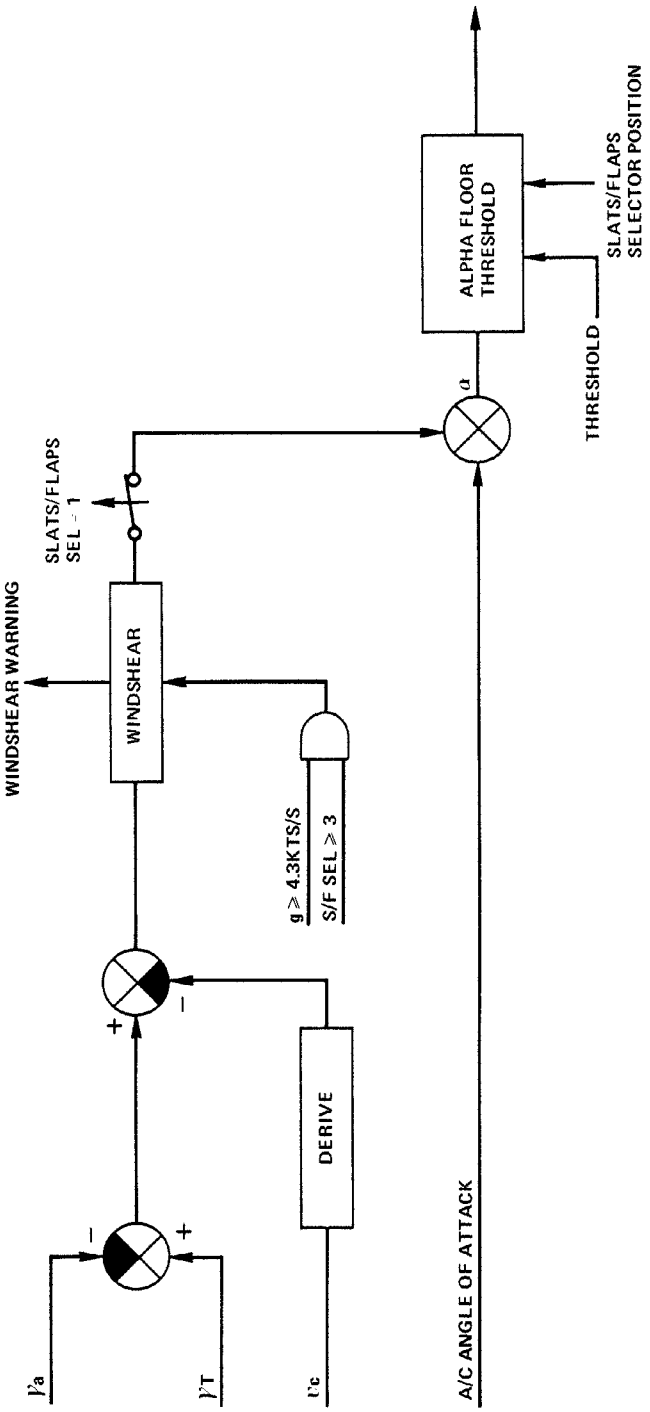
A. General

The windshear is a sudden change in wind direction and/or speed over a relatively short distance in the atmosphere and this may have an effect on aircraft performance during take off and landing phase.
In windshear conditions, the principle is to reduce the detection threshold according to the detected windshear in order to get the possibility of performing a go-around maneuver sooner.
The guidance law in windshear conditions is implemented in the FCC'S.

B. Warning

The windshear warning is only active at take-off and at landing below 1000 ft. During the landing phase the warning is inhibited at 50 ft.
The crew is informed of a windshear by-activation of the following warnings :
- red WINDSHEAR legend displayed on the PFD's at least 15 s.
- WINDSHEAR three times announcement generated by the FWC.

In case of warning inhibition in both FAC's, the message WINDSHEAR ALERT NOT AVAILABLE appears on the L ECAM display unit during TO or landing as



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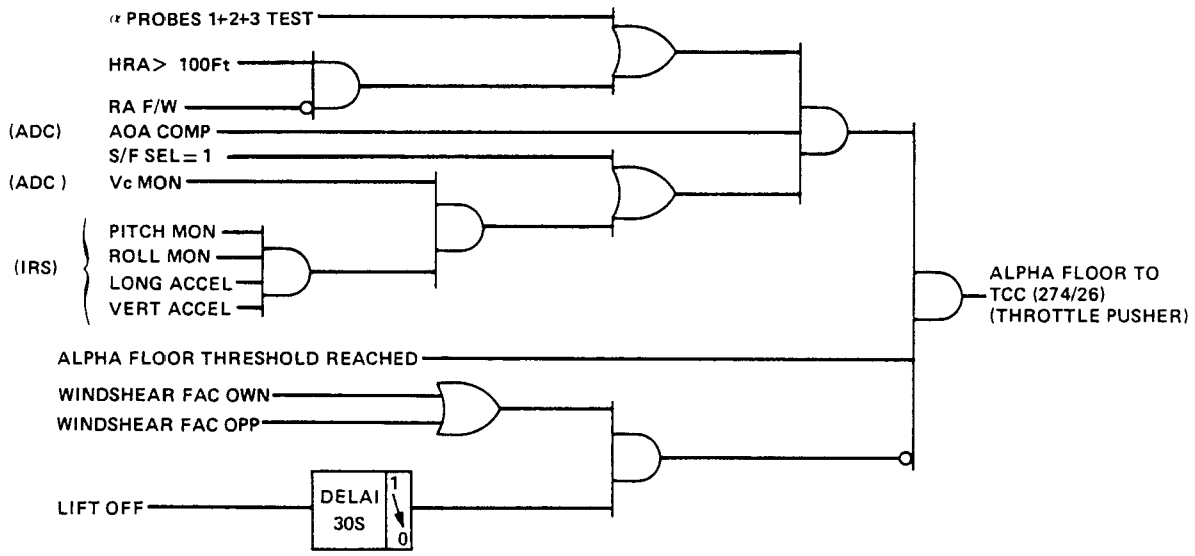
Alpha-Floor Detection Principle
Figure 004



BM5 22 28 00 0 AZL2 00

R

Alpha-Floor Activation Logic
Figure 005

soon as the slats are extended.
(Ref. Fig. 006)
(Ref. Fig. 007)
(Ref. Fig. 008)

C. Specific Monitoring
In order to reduce the rate of spurious warning activations, the windshear warning is inhibited in the FAC when a failure is detected by the following internal monitorings:
- monitoring of the status matrices of the sensors used over the whole flight envelope through the existing logic structures
- specific monitoring by means of a validity test of the following parameters :
  . Vground-Vtas as a function of the wind speed
  . VZBI/Vground-Gamma a as a function of the vertical wind slope.
  Such validity tests (computed value compared with theoretical value) allow the parameter validity to be checked.
- if the warning is not available, this condition will be dealt with and announced by bit 26 of EFIS bus label 274
- any failure which may affect the windshear function must not be the cause of the degradation of all the other functions.

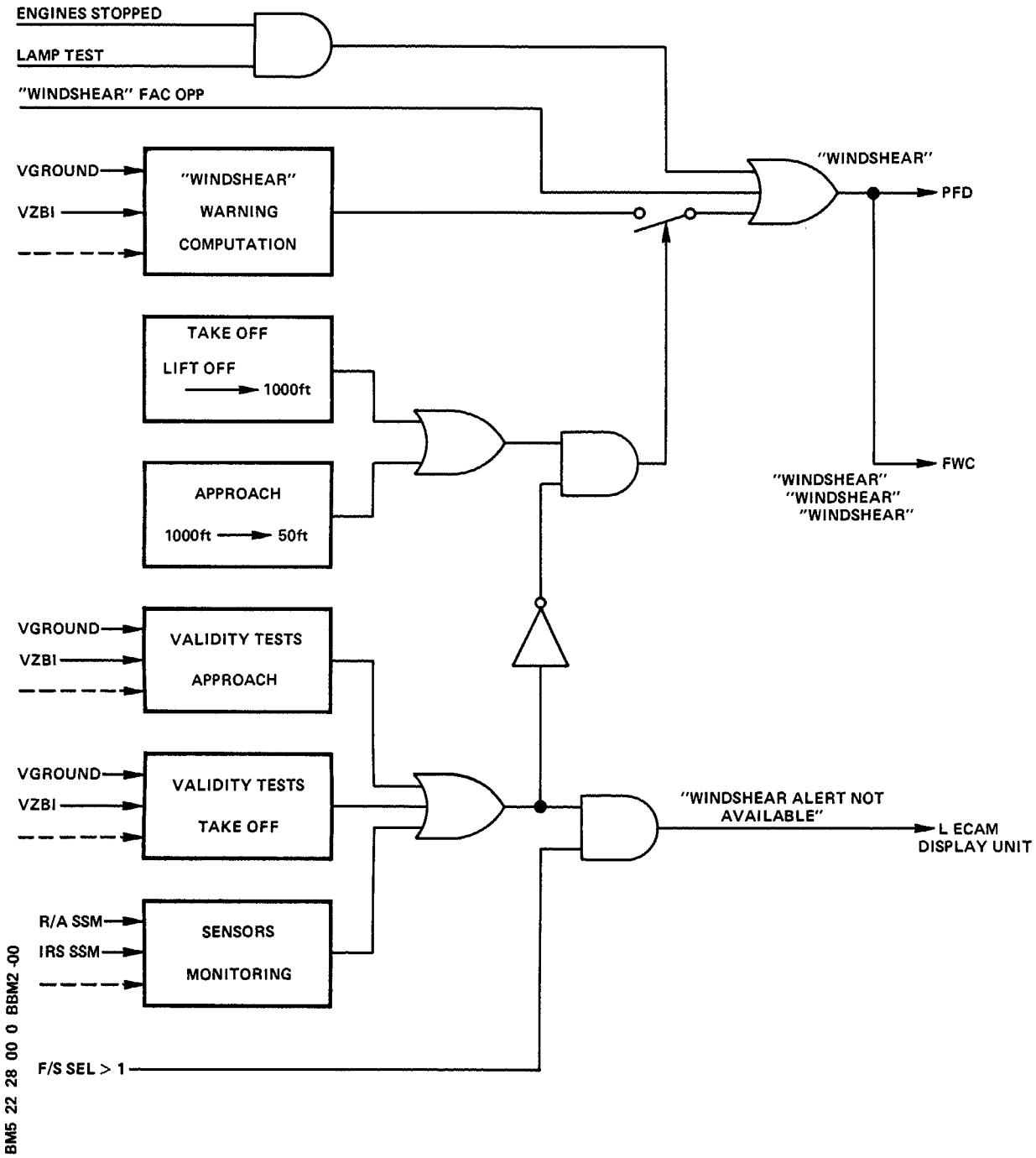
(Ref. Fig. 009)

D. Ground Test of Windshear Function
This test serves to check if the system is able to generate (aurally and visually) the following warnings :

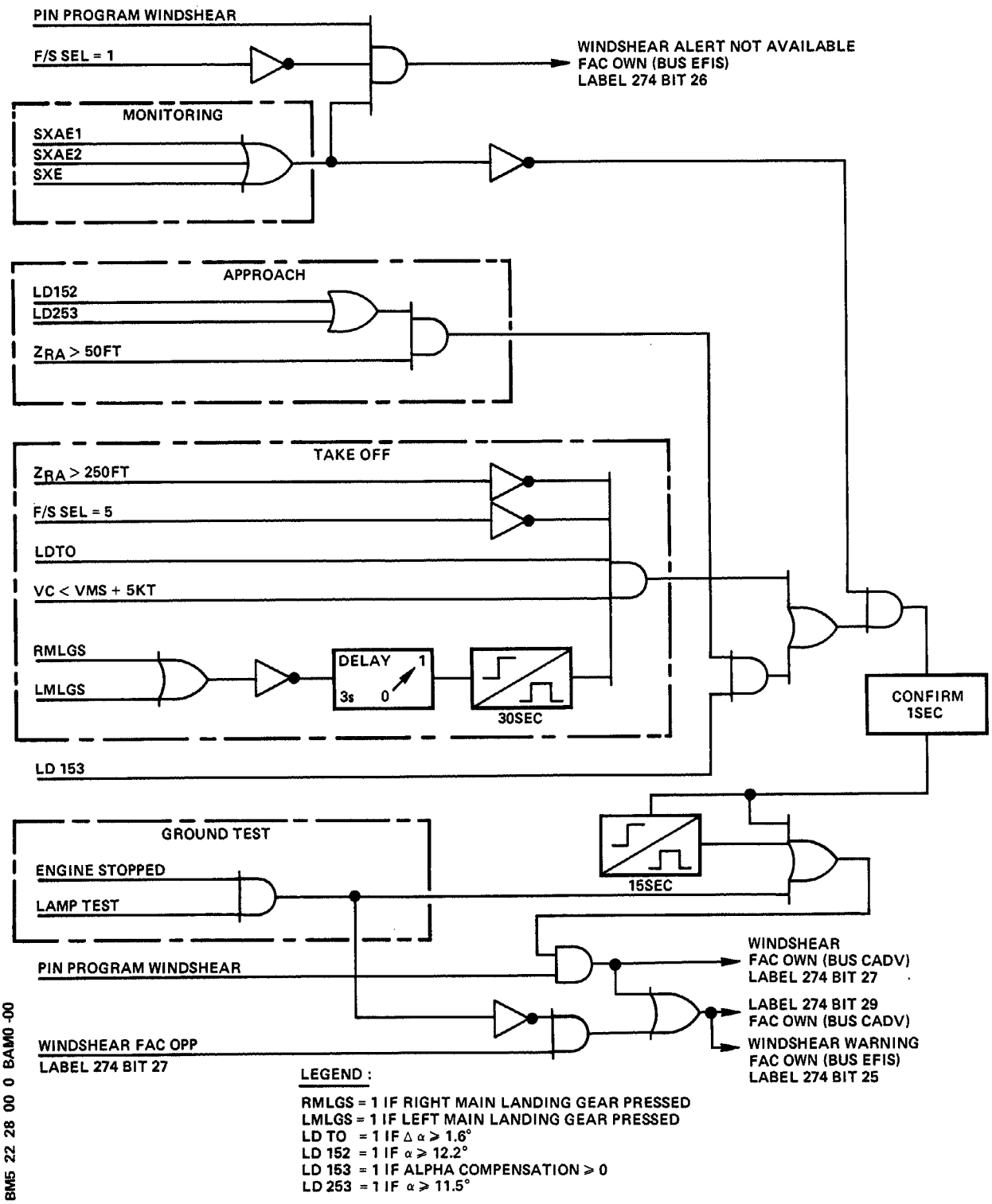
R - WINDSHEAR ALERT NOT AVAILABLE on L ECAM display unit
R - WINDSHEAR on PFD
R - WINDSHEAR WINDSHEAR WINDSHEAR aural warning.

To activate the test, place the engines in shut-down configuration and, on the overhead panel (436VU), put the TEST/BRT/DIM switch in the TEST position.
During the test, the windshear warning from the opposite FAC is inhibited.

WINDSHEAR WARNING SYNOPTIC DIAGRAM



Windshear Warning - Block Diagram
Figure 006



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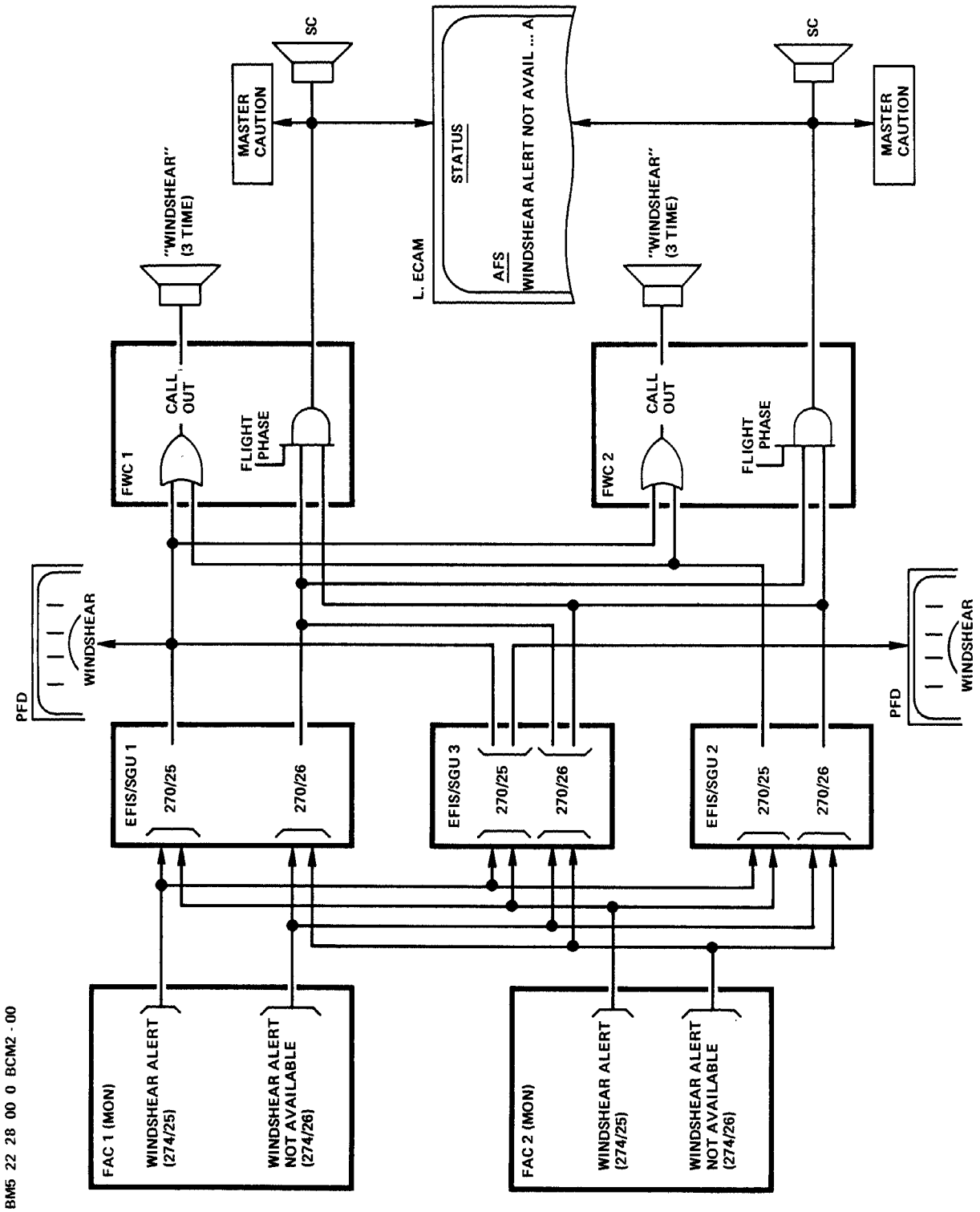
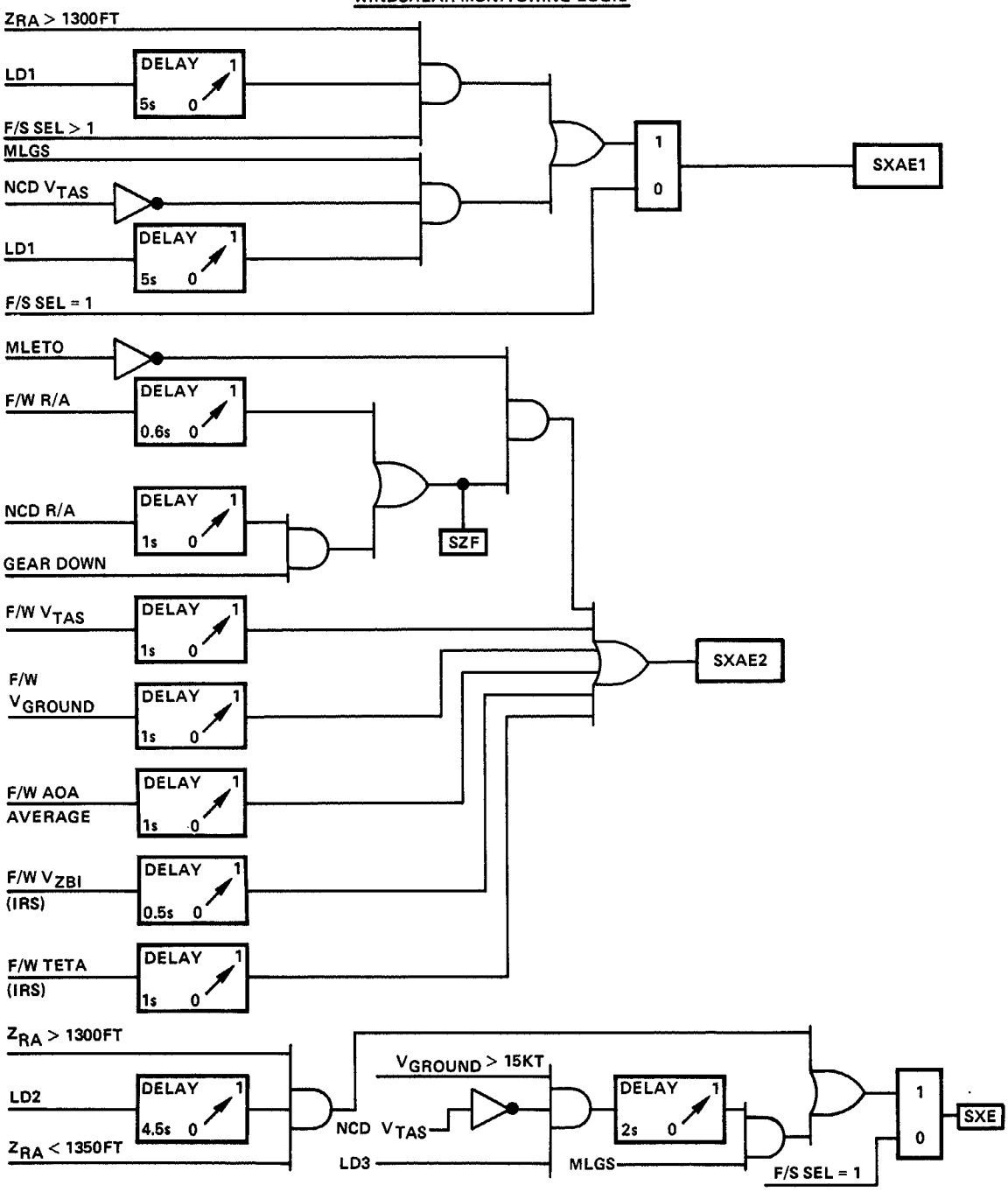
Windshear Detection Logic
Figure 007

R



Windshear Warning-Block Diagram
Figure 008



LEGEND : LD1 = 1 IF | $V_{TAS} - V_{GROUND}$ | > 100 KTS
LD2 = 1 IF | $V_{ZBI} / (KT * V_{GROUND}) * 57.3 - \gamma$ | > 5°
LD3 = 1 IF | $V_{ZBI} / (KT * V_{GROUND}) * 57.3 - \gamma$ | > 2°

BM5 22 28 00 0 BDM2 - 00

Windshear Monitoring Logic
Figure 009

1. Operational Test

A. ALPHA FLOOR Function (Throttle Pusher)
Carry out an autothrottle operational test (Ref. 22-30-00, P. Block 501 paragraph A).

B. Speed Limits Computation
(1)Reason for the job
To check Vmax display on PFD for a given configuration.
(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure
WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
(a)Job set-up
- energize the aircraft electrical network (ref. 24-41-00, P. Block 301).
- make certain that electronics racks ventilation is correct.
- make certain that aircraft is ground configuration.
- close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS.
- make certain that flaps and slats are retracted.
- make certain that ADS1 and 2 are in test condition (on panel 47OVU press ADS SYS1 and ADS SYS2 pushbutton switches)
- align IRS 1,2 and 3 (perform ALIGN procedure until red HDG flags on NDs disappear ; present position entering is not necessary).
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU.
- make certain that ADC INST pushbutton is switched off on L and R instrument panels.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 lever in ON position.	Lever remains in this position. On PFDs - Vmax is displayed on airspeed scale (represented by a red and black checkered tape).
2. De-energize AFS components by placing PITCH TRIM 1 lever in OFF	

ACTION	RESULT
position.	
(c)Close-up 1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301). 2 Restore aircraft and system to normal operating condition.	

1. FAC General Monitoring

A. General

Correct running of the FAC program is checked by the watchdog.
This hardware circuit enables to validate the pitch trim and yaw damper engage logic as well as ARINC bus transmission.

B. Watchdog Description (Ref. Fig. 001)

The watchdog is a monostable set periodically to right state by the real-time monitor.
In the event of absence of re-activation the monostable is latched to wrong state. This fault information is used in :
- the command and monitoring hard-wired logic of the pitch trim and yaw damper systems
- the transmission validation of 12K ARINC data bus
- the associated microprocessor check (program disabling).

The maximum delay of non-reactivation has been set to twice the long cycle with a margin of 10% approximately. Delay is then of 330 ms.
The monitoring functions are activated at the end of this delay.
The watchdog is also reactivated and unlatched by operating the PITCH TRIM and YAW DAMPER engage levers if those had previously returned to OFF position and also at each power rise.
In the event of a detection of a watchdog at wrong state (command or monitoring), a permanent reset is made on the microprocessor and on all the booleans transmitted outside and forced to zero.

C. Real-Time Monitor Description

The real-time monitor manages the sequencing and execution of the various tasks required by the program execution.
It also coordinates interruption acquisitions. It is composed of a certain number of software modules (CLOCK, SUPER CLOCK, TASKCALL).
The real-time monitor (RTM) generates two types of monitoring signals.

(1)Boolean BRLTIME

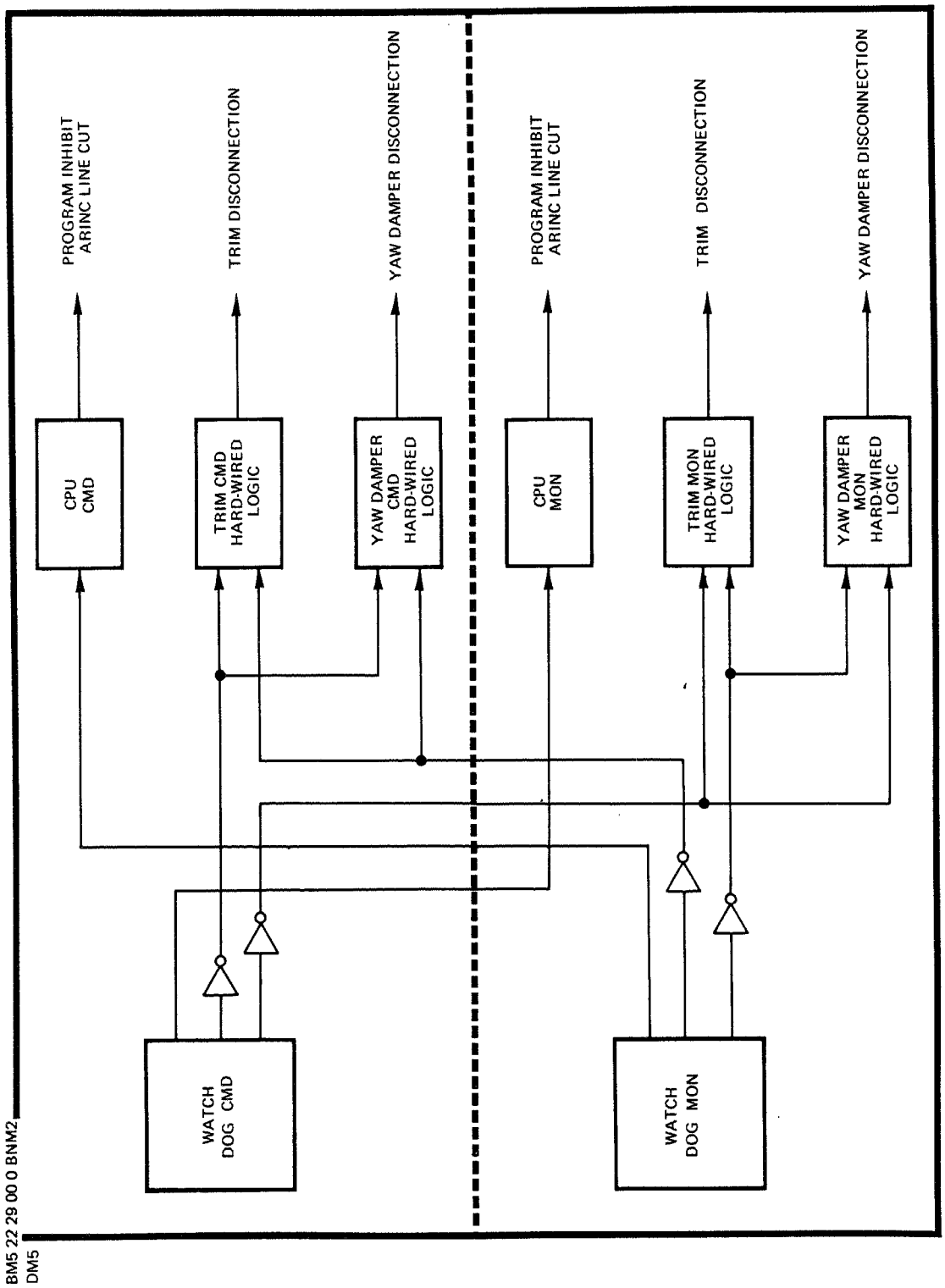
Boolean BRLTIME represents a discrepancy in the real-time sequencing of an execution. In fact this boolean represents monitoring of the interruptions which may happen in the program as well as any possible resulting disturbance.
This boolean is used in the common LOCFAC logic and causes :
- FAC loss (pitch trim and yaw damper)
- ARINC transmission cutoff.

Another indirect consequence is the watchdog non activation as the module which is specialized in this task (TASKCALL) is not called in this case.

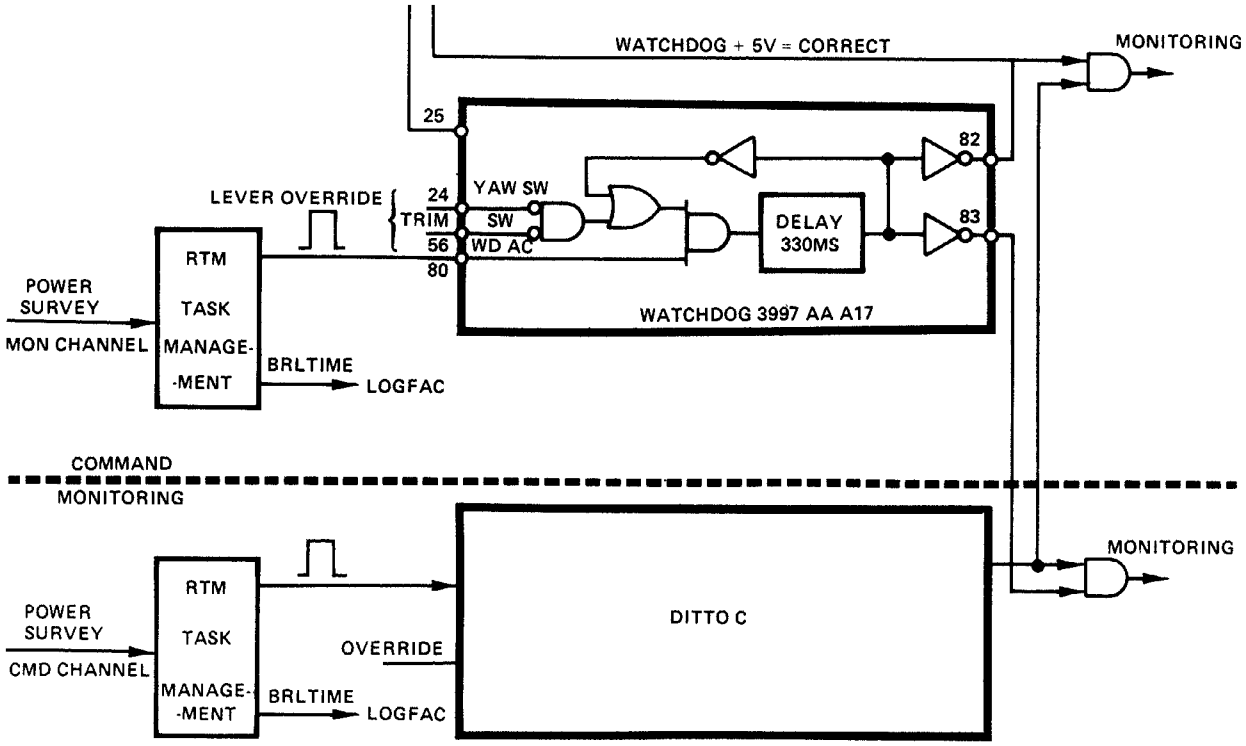
(2)Watchdog activation (Ref. Fig. 002)

The watchdog is activated by the TASKCALL module when the task execution is in conformity with the expected program depending on the running cycle.
The watchdog is reset at the end of the running task.
The activation signal is a pulse. The status of the internal power supply

monitoring also conditions the status of the real-time monitor monitoring functions.



Watchdog Data - Schematic (Hard-Wired Logic)
Figure 001



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Watchdog Activation Principle
Figure 002

2. FAC Self-Test

A. General

The FAC safety devices (comparators, monitoring circuitry...) undergo tests at regular intervals in order to reduce the time of risk concerning the system detection efficiency.

B. Activation Conditions

These safety tests are activated automatically at each FAC general energization on the ground (28 V power rise in aircraft on the ground).
Voltage is held while tests are performed.
Result is memorized and conditions system engagement in flight and on the ground.
Specific precautions are taken to prevent any test activation in flight :
- hardware inhibition ("nose gear pressed" signal not present) of memories containing the resident test program
- in-flight test request inhibition.

The safety test results are safeguarded in the memory even upon 28 V short power cutoffs and even in the event of battery discharge. In the event of an in-flight program failure detected by the watchdog on one channel only (command or monitoring), the test result is forced to right state if system reset is possible. If the two channels are failed, the engagement of trim and yaw damper is only authorized if the safeguard test results are OK.

C. Safety Test Principle (Ref. Fig. 003, 004)

The test principle consists in performing, through generation of stimuli, the test of :
- all the hardware devices taking part in FAC safety
- ROM memory check sums, which is the only check concerning software devices
- writing/reading of RAM memory and check of a precise address.

These devices are :
- the watchdog
- the voters, comparators and delays of the yaw damper channel
- the hardware circuits taking part in the yaw damper disconnect logic (hard-wired logic)
- the comparators and hardware delays of the pitch trim channel
- the hardware circuits taking part in the pitch trim disconnect logic (hard-wired logic).

These tests are performed in the command and monitoring channels. Each comparator-type element is checked by positive and negative stimuli. A crosstalk between command and monitoring processors enables test running.
During the tests, the computer is isolated from the outside world through demodulator hardware inhibition.
Thresholds and delays are checked at ±20% of theoretical value. The hard-wired disconnect logics are generally checked, input by input, through their action on the system switch over.
The FAC software devices are :
- IRS comparison
- oscillation detector
- angle-of-attack monitoring

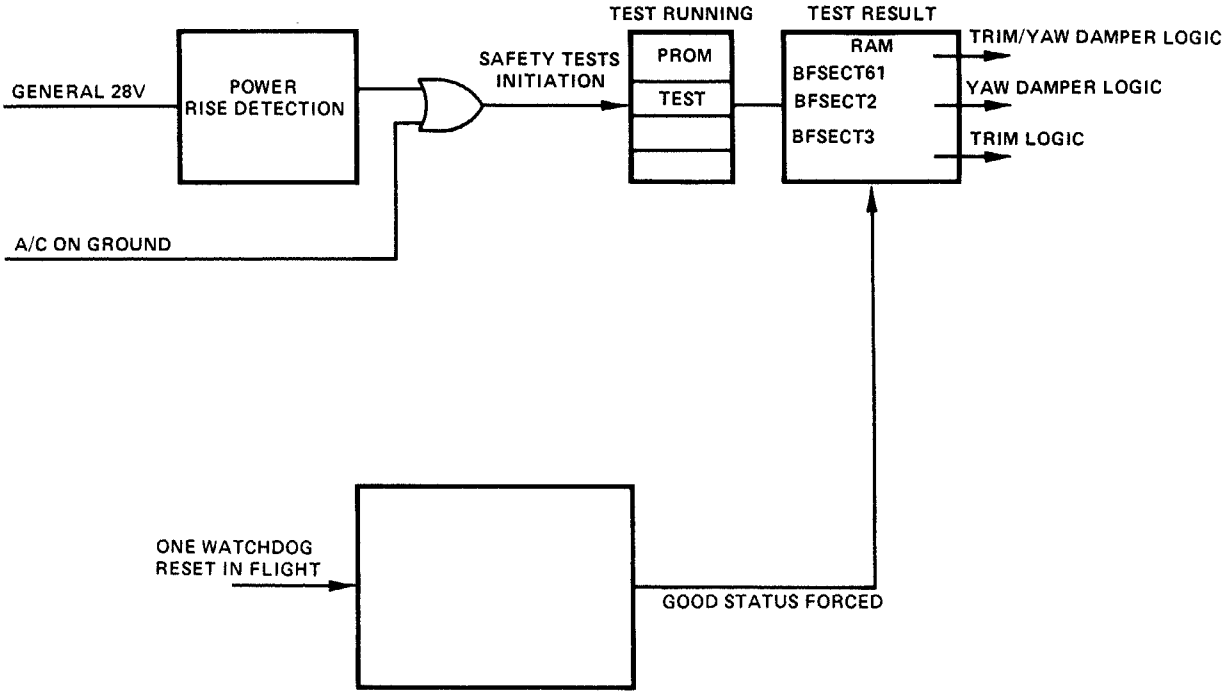
- ARINC line monitoring
- amplitude and speed limitation
- power supply comparison
- real-time monitor monitoring
- trim monitoring (C4, C15, C12 and BTTRATE)
- safeguard upon short cutoffs through comparator inhibition.

All these software devices are checked :
- through check sum of all the read only memories (PROM)
- through all the random access memories.

The result of the safety tests is indicated by three booleans as shown in the following table :
- right state = FF
- wrong state = 00

	BFSECTS 1	BFSECTS 2	BFSECTS 3
Failure in numerical core common part	WRONG	WRONG	WRONG
Failure in yaw damper part	RIGHT	WRONG	RIGHT
Failure in pitch trim part	RIGHT	RIGHT	WRONG

As soon as a failure is detected in one of the three considered configurations, the safety test is stopped.
Booleans BFSECT1, 2 and 3 are taken into account by the FAC engage logic as follows :
BFSECTS 1 : FAC logic
BFSECTS 2 : yaw damper logic
BFSECTS 3 : pitch trim logic.
Complete test lasts for 20 s approximately. Any attempt to engage the system must not be made during the test as the program could be stopped.



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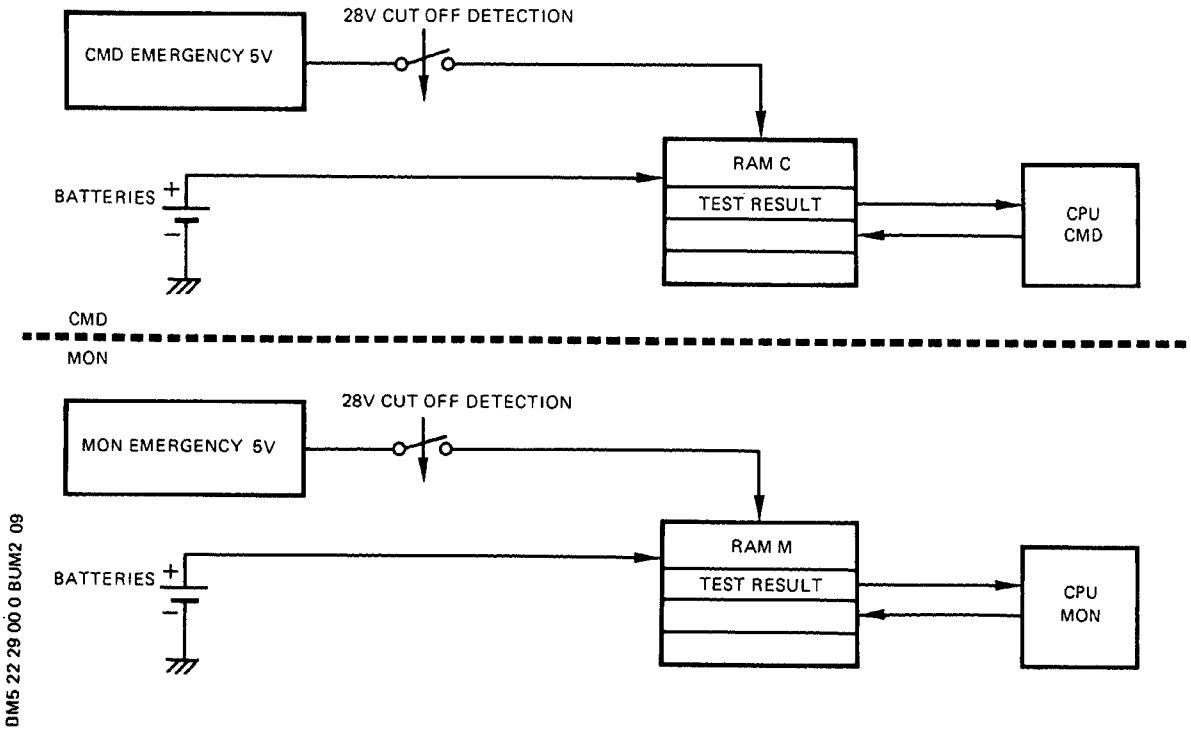
FAC Safety Tests-Schematic
Figure 003



Safety Test Safeguard
Figure 004

3. FAC Maintenance

A. General

FAC in-line maintenance is based on a resident program enabling isolation of faults which can affect :
- the computer itself
- the peripheral systems
- the power loops
- the engage unit
- the standby alpha probe.

The result of this fault isolation is displayed on the maintenance test panel (MTP). The fault data are described in detail in 22-40.

B. Fault Isolation Operation and Activation

Any disconnection or refusal to engage of the pitch trim or yaw damper system causes generation of an interruption command in the command and monitoring channels which is used to :
- take a snapshot of the system engage logic at the time of disconnection. The result is stored in a specialized memory accessible through a special tool (maintenance information printer)
- initiate a disconnection analysis as soon as possible (specialized background task) from the safeguarded unit fault data (snapshot).

Disconnection is analysed in an independent way by each processor and the result established in the monitoring channel is sent to the command channel.
The command channel makes a comparison between the results and initiates any additional demand.
The final diagnosis is sent to the MTP in coded form (ARINC format 12K bus).
This can be :
- a voluntary disconnection
- an abnormal but identified disconnection.
- a disconnection not identified.

In the last case, complementary tests on the ground can be required. If a doubt persists, a faulty FAC message is given instead. The disconnection analysis takes into account all the pitch trim and yaw damper logics and monitoring functions.
In the event of ambiguity it considers the failure probabilities in order to give the most probable causes. As far as the standby alpha probe and ADC angle-of-attack data are concerned, the in-flight isolation function is permanent and displayed on the MTP.

C. Precautions

The in-flight test function must not interfere with normal operation of the FAC.
In consequence, the following precautions are taken :
- during the flight, a "nose gear pressed" wired signal prohibits access of RPROMs where the test program is located.
- software precautions are taken at the level of the FAC and MTP.

Consequently the in-flight test program cannot interrupt the system program as it is only executed in background task.

Table 2 : Input Parameter Processing

PARAMETER	NON VALIDITY	FLIGHT CONDITIONS	CONSEQUENCES
Lateral accelera-tion	F/W and NCD	AP CMD in TO/GA	Yaw damper loss
Control wheel deflection	F/W and NCD	Manual or CWS	Yaw damper loss, if only one EFCU lost
Vc	F/W and NCD not taken into account	Flaps retracted Flaps at 8°	FAC total loss Gains fixed Yaw damper or Trim not lost
TAS	F/W and NCD	Throughout flight envelope except flaps 8°	Yaw damper loss except on ground and flaps ≥ 8
ADC angle-of-attack	F/W and NCD not taken into account	Incidence trim	Trim 1 and 2 loss if ADC1 and 2 alpha faulty
Standby alpha probe angle-of-attack	CORDIC validity (sin2 alpha + cos2 alpha = 1)	Throughout flight envelope	Standby alpha probe eliminated from the vote
Bank angle	F/W and NCD	Throughout flight envelope	Yaw damper loss
Mach	F/W and NCD not taken into account	Flaps retracted Flaps at 8°	FAC total loss Gains fixed Yaw damper or Trim not lost
Landing gear	- L and R MLG dissymmetry - validity 28 V loss	Throughout flight envelope	Pitch trim loss or 2 SFCC
SFCC	Flag and NCD	Throughout flight envelope	Loss of one SFCC : reconfiguration AFS not lost

1. Operational Test

A. Landing Gear Relay Test
(1)Reason for the Job
To check FAC monitoring of landing gear data.
(2)Equipment and Materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure
WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED
(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- close overhead panel circuit breakers particularly those associated with AFS, ADS and NAVIGATION.
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that ADS is operating.

(b)Test

ACTION	RESULT
1. On panel 22VU, - open circuit breaker 5RK (line 204/F27).	
2. On overhead panel, - place PITCH TRIM 1 lever in ON position.	Lever remains engaged in this position.
3. On panel 133VU, - open L/G PROX DET & RELAYS/ SYS1/RELAYS & RETRACT/CTL circuit breaker 2GB (line 332/U51). - close circuit breaker 2GB.	PITCH TRIM 1 lever returns to OFF position.
4. On overhead panel, - place PITCH TRIM 1 lever in ON position.	Lever remains engaged in this position.
5. On panel 133VU, - Open L/G PROX DET & RELAYS/SYS1/ FLT/GND circuit breaker 119GB (line 332/U52).	PITCH TRIM 1 lever returns to OFF position.

ACTION	RESULT
- close circuit breaker 119GB.	
6. On overhead panel, - place PITCH TRIM 2 lever in ON position.	Lever remains engaged in this position.
7. On panel 133VU, - open L/G PROX DET & RELAYS/ SYS 2/FLT/GND circuit breaker 1GB (line 331/V52). - close circuit breaker 1GB.	PITCH TRIM 2 lever returns to OFF position.
8. On panel 22VU, - close circuit breaker 5RK (line 204/F27).	

(c)Close-up

1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

2 Restore aircraft and system to normal operating condition.

AUTOTHROTTLE - DESCRIPTION AND OPERATION

1. Introduction

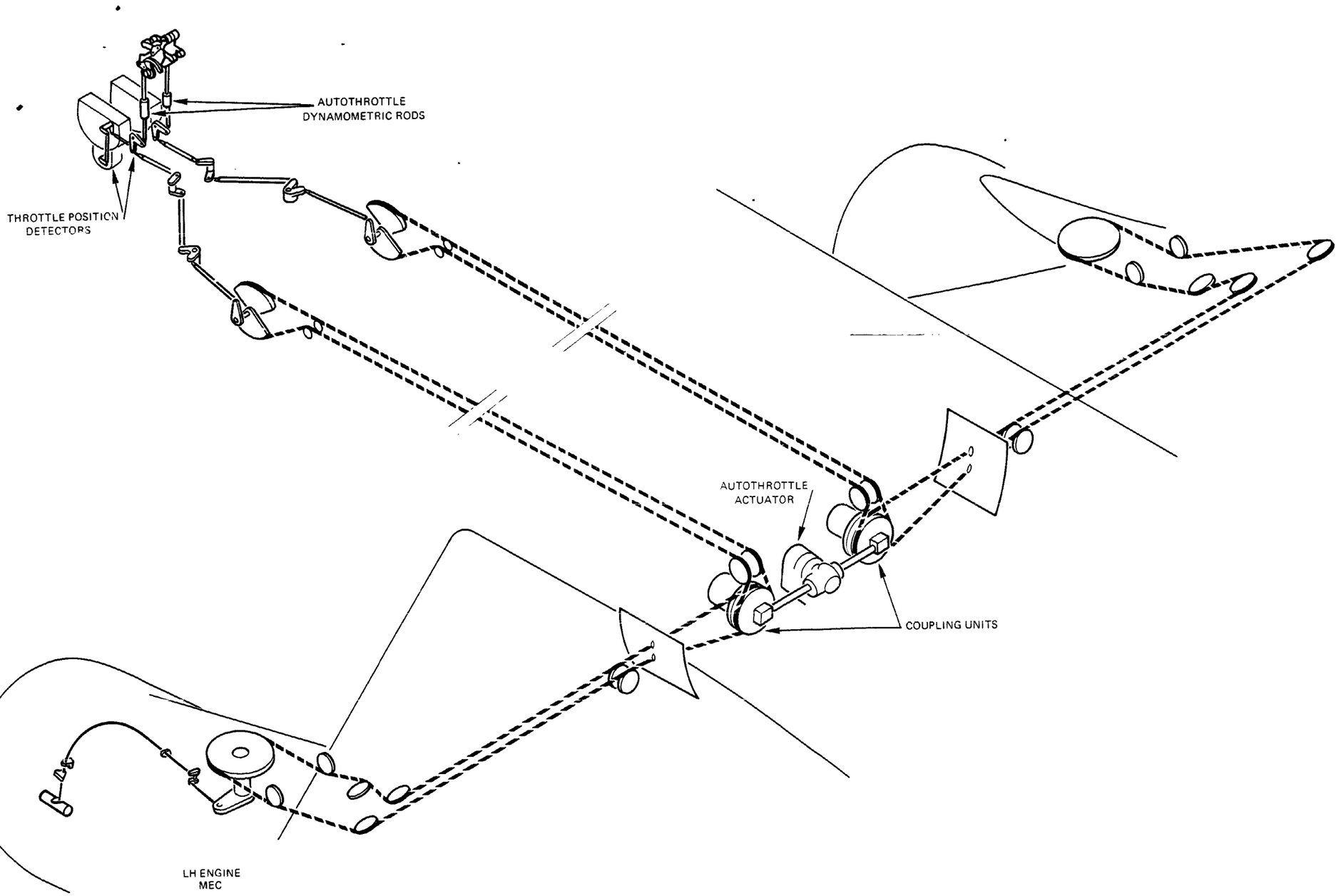
The Thrust Control Computer System (TCCS) is a portion of the larger Automatic Flight System (AFS) whose characteristics are described in System Description Note 22-00.
The functions of the Thrust Control Computer System are :

A. Permanent Computation of the N1 Limit Value Corresponding to the Flight Phase Selected on the Thrust Rating Panel (Thrust Modes) Function of Ambient Conditions.

B. Autothrottle
(1) Acquisition and hold of the selected speed or Mach number through automatic selection by the FCC or FMC.
(2) Acquisition and hold of the limit thrust by :
(a) Actuating the TO/GA levers (Takeoff, Go Around phases)
(b) Automatic engagement (alpha floor).
(c) Engagement upon FCC request, when LVL/CH climb is initiated.
(3) Acquisition and hold of rating (TARGET) computed by the FMC in PROFILE mode.
(4) Decrease of thrust upon FCC request when LVL/CH descent is initiated upon FMC request in descent or during flare out.
(5) Underspeed, overspeed and THR limit overshoot protection in SPD/MACH mode.

2. Component Location
(Ref. Fig. 001)
(Ref. Fig. 002)
(Ref. Fig. 003)
(Ref. Fig. 004)

FIN	FUNCTIONAL DESIGNATION	PANEL	ZONE	ACCESS DOOR	ATA REF.
3CJ1	COUPLING UNIT-AT, L		141	131SW	22-33-28
3CJ2	COUPLING UNIT-AT, R		142	132SW	22-33-29
5CJ	P/BSW-INSTINCTIVE DISCONNECT, CAPT	9VU	211		
6CJ	P/BSW-INSTINCTIVE DISCONNECT, F/O	9VU	211		
12CJ	ACTUATOR-AUTOTHROTTLE		141	131RW	22-33-22
32CJ	ROD-AT DYNAMOMETRIC, L	9VU	211	121BL	76-11-21
34CJ	ROD-AT DYNAMOMETRIC, R	9VU	211	121BL	76-11-21
301CJ1	TCC-1	81VU	121	121BL	22-36-34
301CJ2	TCC-2	81VU	121	121BL	22-36-34
303CJ	TRP	4VU	211		22-35-12
313CJ	DET-THROTTLE POSITION, L	9VU	211	121BL	22-33-16
314CJ	DET-THROTTLE POSITION, R	9VU	211	121BL	22-33-16
27CA	LEVER-TO/GA, L	9VU	211		
28CA	LEVER-TO/GA, R	9VU	211		
34FN1	PFD-CAPT	3VU	211		34-72-22
34FN2	PFD-F/O	5VU	212		34-72-22



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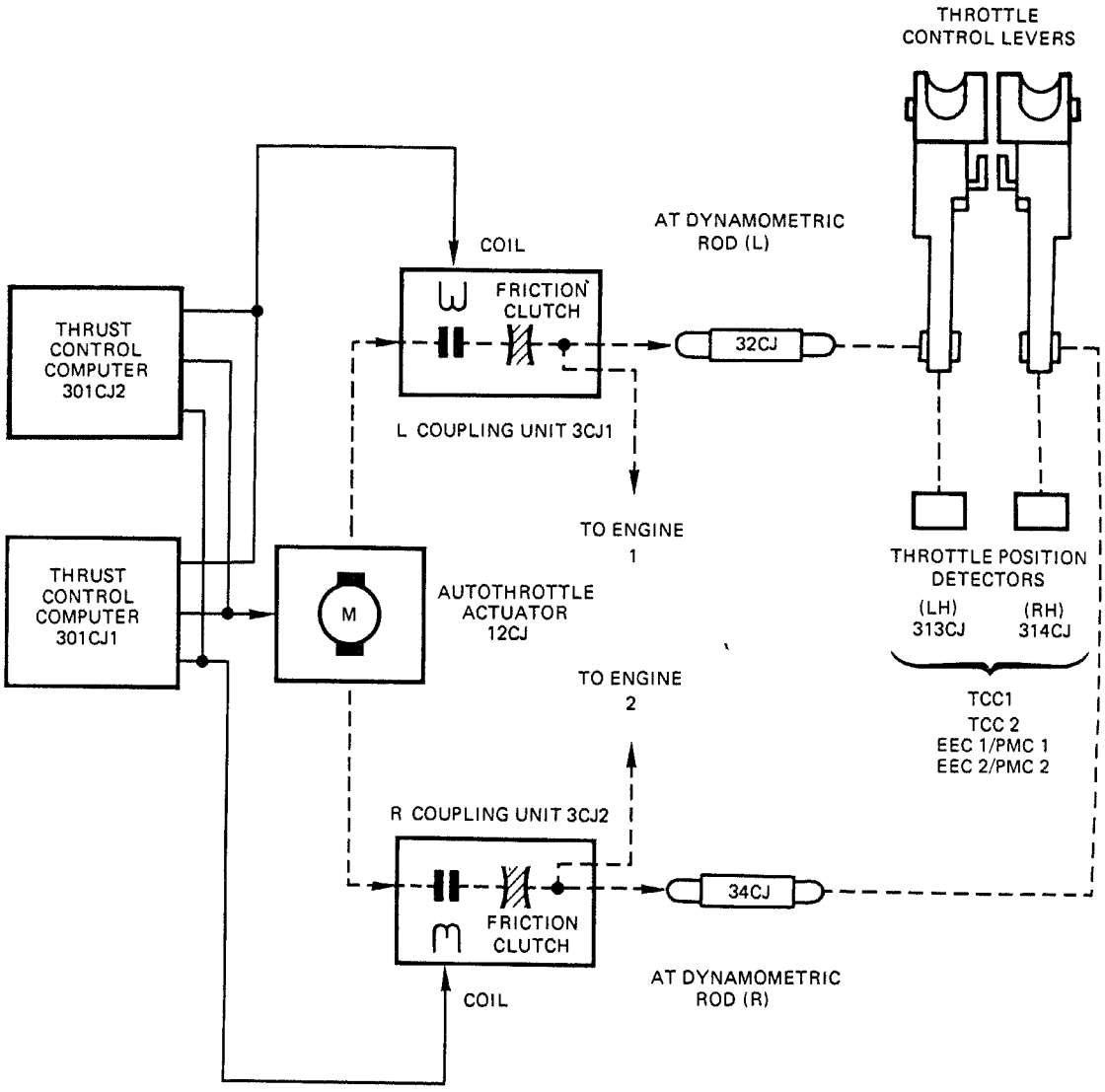
TCCS - Control Channel Layout
Figure 001



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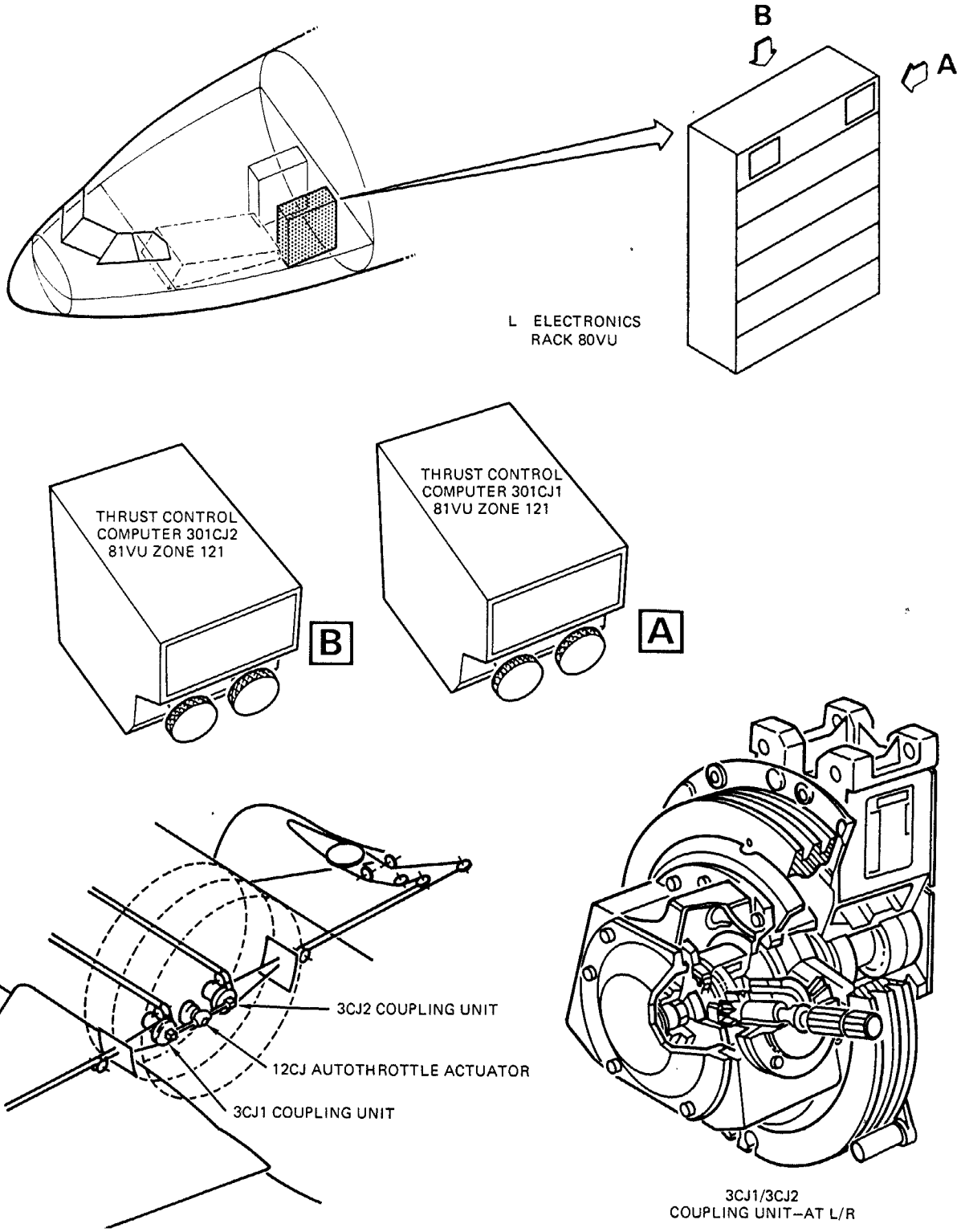
DM5

TCCS - General Arrangement
Figure 002



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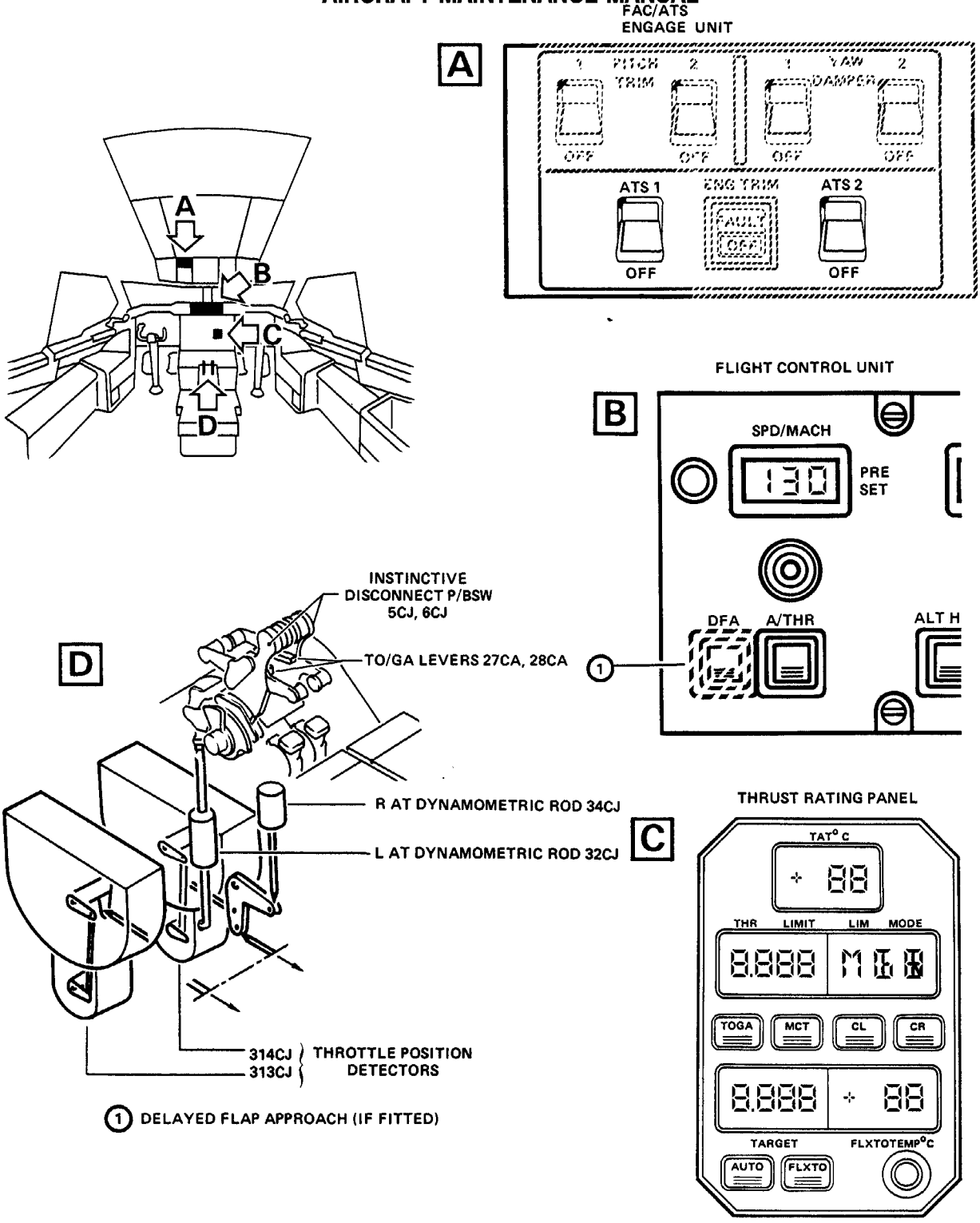
DM5

TCCS - Component Location
Figure 003



TCCS - Controls and Indicating in Flight Compartment
Figure 004

3. Thrust Control Computer System Functions

A. Computation and Display of N1 Limit Value

Computation of the N1 Limit Value enables :

(1) Reduction of crew workload related to engine management, by removing the need to consult curves dependent on a great number of parameters ; moreover, the shorter the flight, the more frequent the consultation.

(2) Reduction of the risk of errors, achieved by monitoring computations.

(3) Reduction of the response time following a change in an external parameter (temperature, altitude, Mach) and highly accurate correction of the effects of air bleed and wing, nacelle and engine anti-icing conditions very accurately.

(4) Availability of an engine limit assisting the internal engine control.
This system enables the crew to perform a flexible takeoff by selecting a fictitious total air temperature higher than the actual temperature on the TRP.
Selection of this temperature is made in accordance with takeoff conditions (takeoff weight lower than maximum takeoff weight, runway length...).
Such a procedure results in an increase in engine component life, which is directly related to high-power operation.
Takeoff power may be selected between the maximum and minimum limits authorized by the engine manufacturer (FLX TO).

N1 limit value is obtained from the following data :

- altitude
- Mach number
- computed airspeed
- total air temperature
- total pressure and impact pressure
- antiice and air bleed conditions.

These computations are made for the six following flight modes :

- takeoff (TO)
- flexible takeoff (FLX TO)
- go around (GA)
- maximum continuous (single engine operation) (MCT)
- maximum cruise (CR)
- climb (CL).

Computation of the N1 limit value is detailed in 22-36.

In AUTO mode, management of the modes is made automatically by the flight management computer (FMC).

Display of the N1 limit value :

The N1 limit value is sent to the TRP and engine N1 indicators thus allowing comparison on these indicators between actual engine rating and maximum engine rating.

B. Automatic Thrust Control

The following modes are available if the ATS has been armed and if the A/THR mode has been selected on the FCU.

(1)SPEED/MACH control

In SPEED (or MACH) hold mode the system enables slaving of the aircraft speed (or Mach), transmitted by the ADC, to the reference value set on the FCU or in PROFILE mode to the target speed computed by the FMC (cyan index displayed on PFD speed scale).

SPEED or MACH mode is selected by pressing SPD/MACH pushbutton switch located on the left-hand side of SPD/MACH display.

The speed (Mach) reference value can be modified by rotating the selector knob located below the display.

The SPD/MACH mode is engaged mainly in response to FCC or FMC signals.

(2)THRUST control

In THRUST mode the system enables slaving of the engine rating to the N1 limit value computed by the TCC or to the target value given by the FMC using information from the TRP (mode selection), the engines (N1 command/actual), air conditioning system and ADC parameters (TAT, Mach, altitude, Vc, total pressure, impact pressure)

- Takeoff thrust is set separately on each engine by means of clutches. After TO/GA lever activation, both throttle control levers are driven together at a constant rate of + 3 °/s TLA until alignment of each N1 command on N1 limit is achieved (with a delay of 200 ms).
  When this condition is fulfilled on the leading engine, the corresponding clutch is deactivated.
  When this condition is fulfilled on the trailing engine, the 2nd clutch is deactivated, the actuator is stopped, and cyan THR replaces green THR on PFD.
  It is to be noted that the declutch logic is latched until landing gear is retracted or mode is changed on TRP.
- In flight, after reclutch, the system makes the difference between the N1 limit value and the higher N1 command. The throttle control levers are driven together to align the higher N1 command on the N1 limit.
  When the PROFILE mode is engaged and active (P.THR on PFD) or when the ENG TRIM pushbutton switch is selected OFF, both N1 actual values are used instead of N1 command.
  The THRUST mode is engaged in response to FCC or FMC signals.

(3)RETARD mode

The RETARD mode becomes active when the FCC delivers a descent command signal in the level change phase. In this mode the pilot can stop the throttle control levers before they reach the idle position by applying a load on either of them.

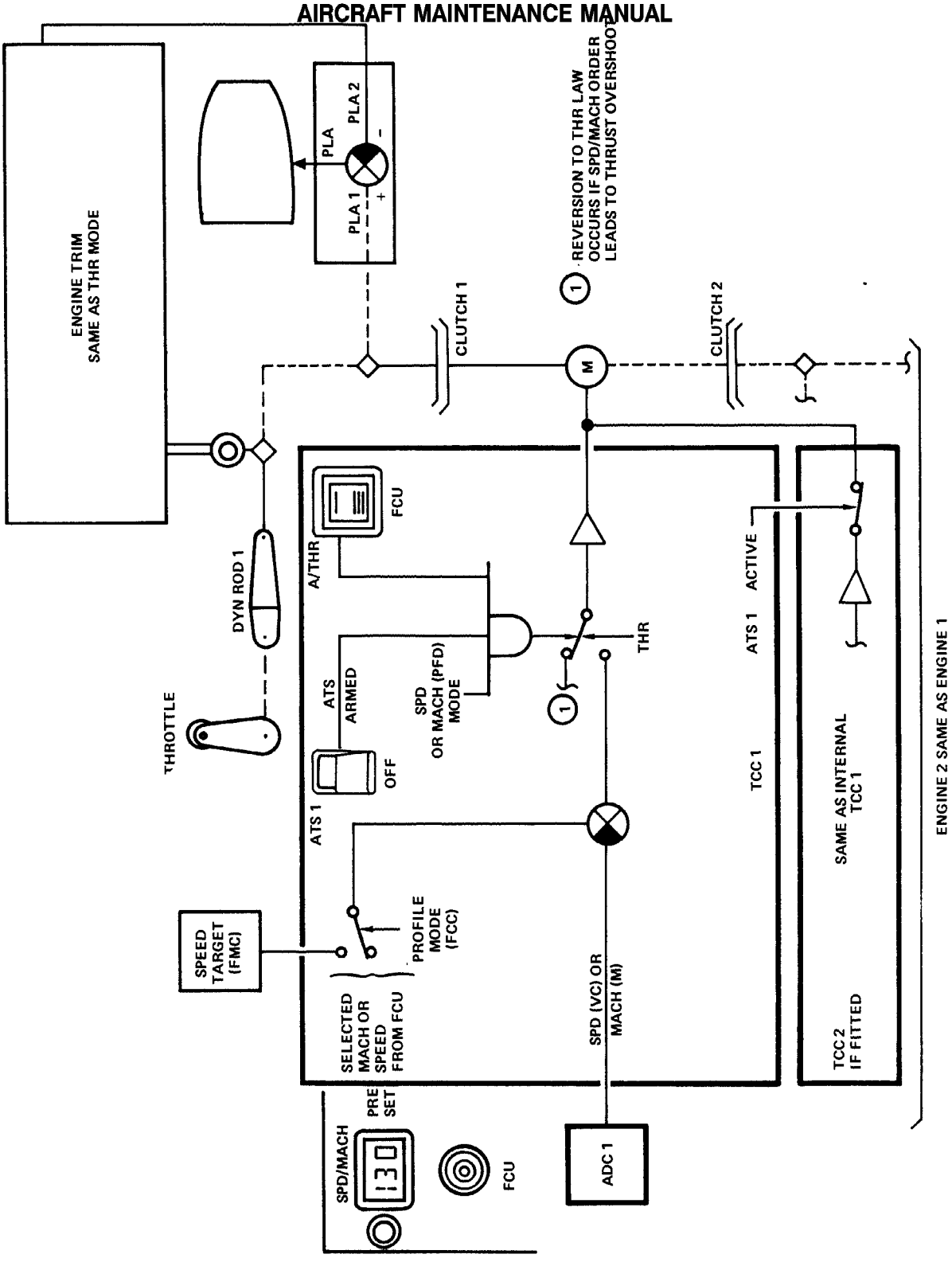
This mode is also active during flare out as soon as altitude reaches 30 ft.

The throttle control levers move to the idle position.

(Ref. Fig. 005)

(Ref. Fig. 006)

(Ref. Fig. 007)



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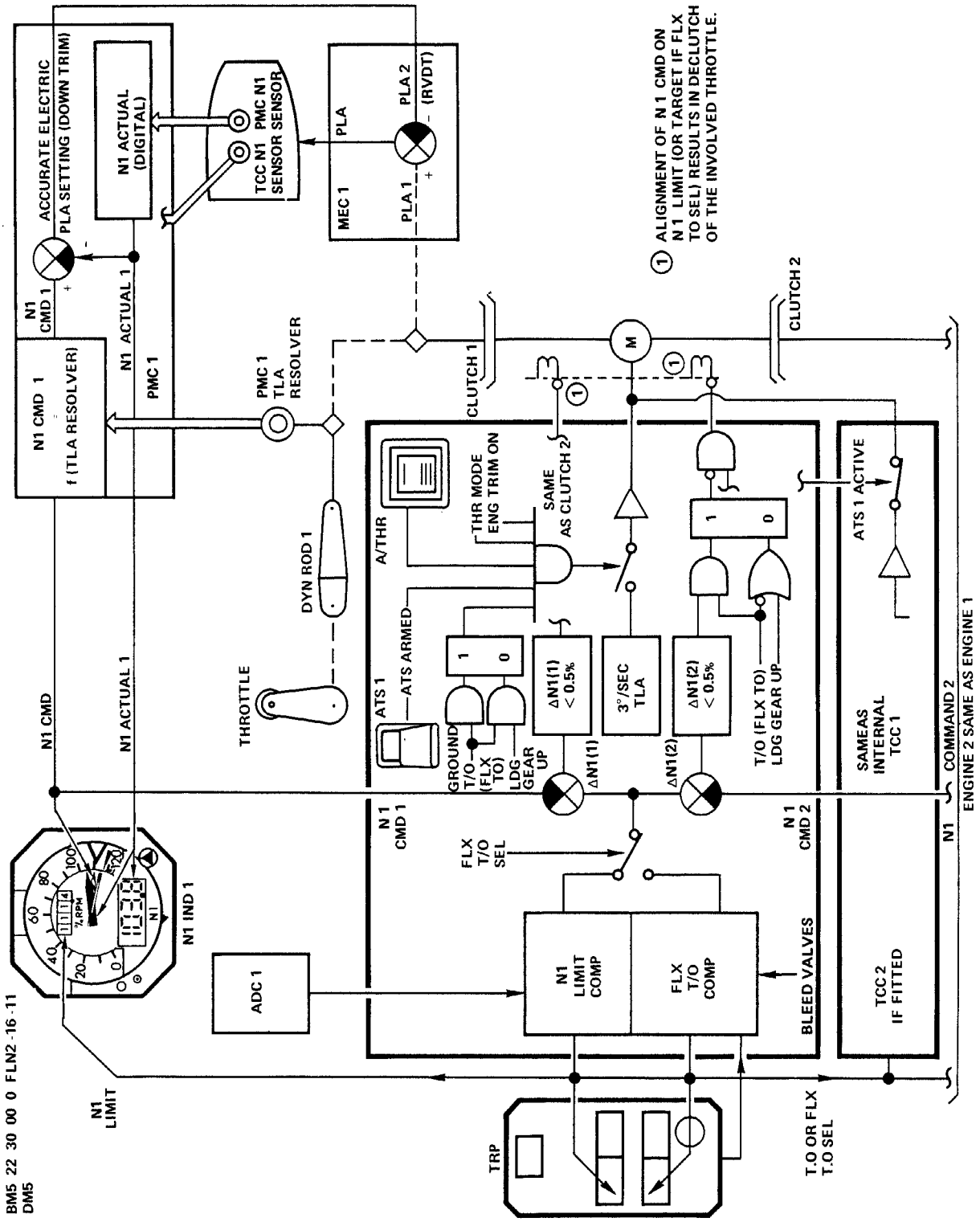
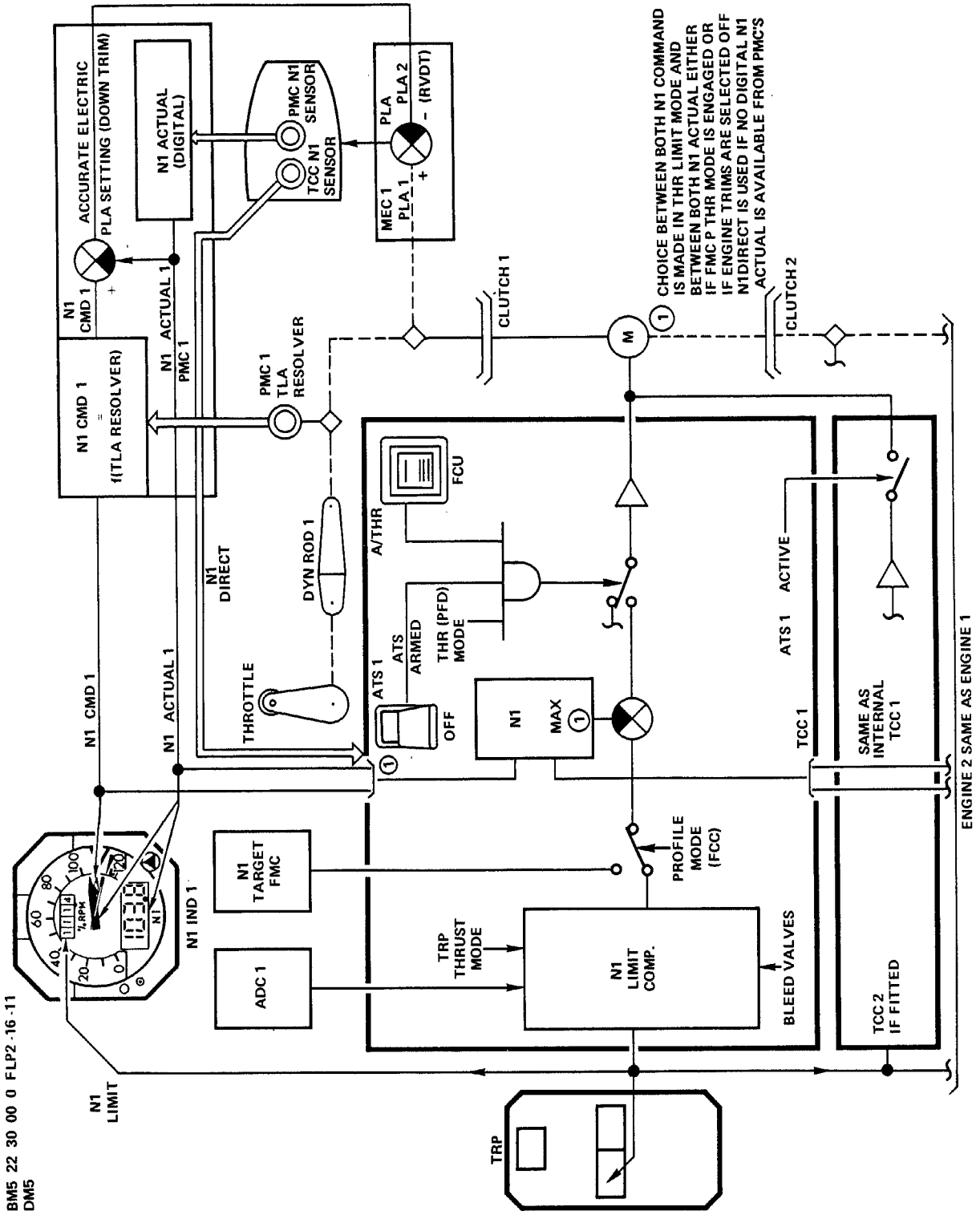
DM5

SPEED/MACH Mode
Figure 005



Takeoff Thrust Setting
Figure 006



Cruise Thrust Mode
Figure 007

C. Function and Operation of Power Management Control

(1) Function

The power management control (PMC) operates continuously during flight either in automatic control or in manual flight.

The PMC which is linked to the main engine control (MEC) provides fuel control : the main control which is mechanical is modified by the PMC. with a restricted authority.

The function of this electronic trim is to simplify the engine control and provide protection against overspeed.

(2) Operation

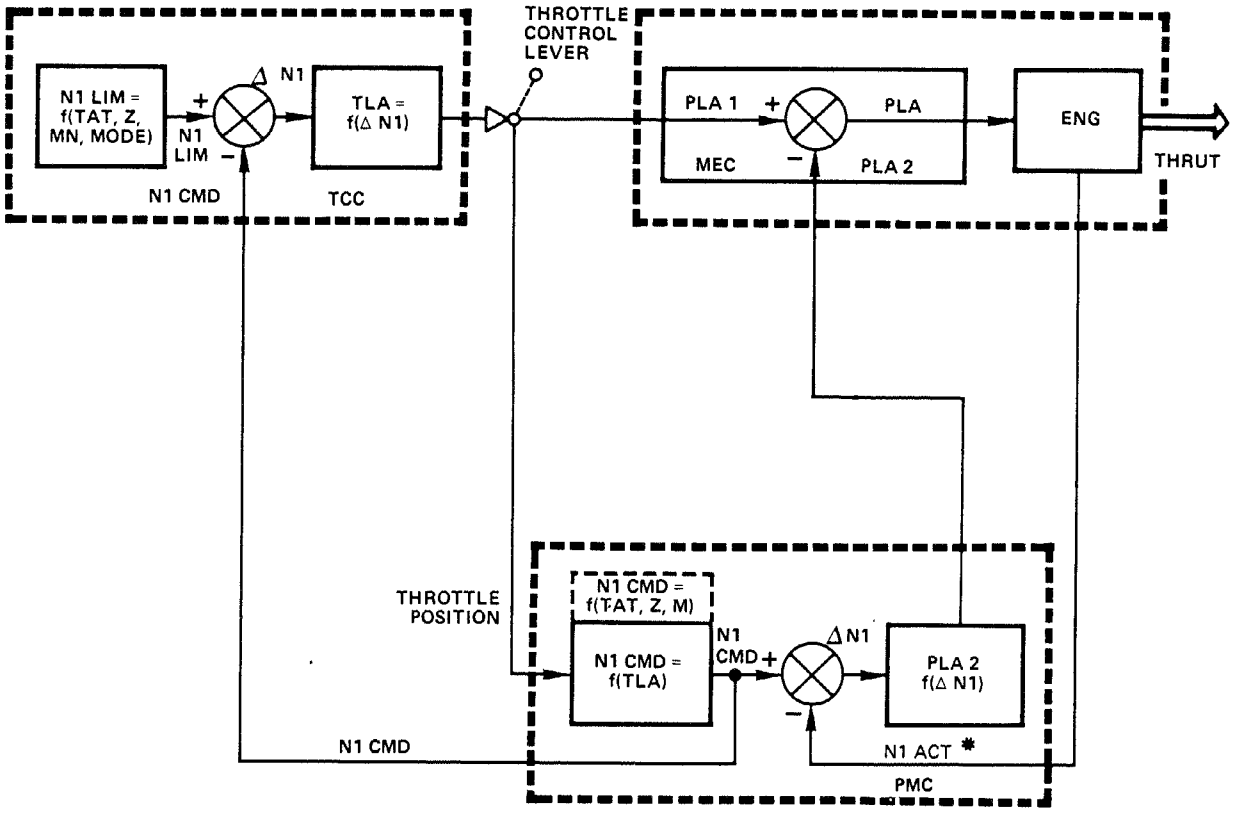
After a mode selection, the TCC generates a lever displacement signal. This signal is based on the difference computed by the TCC between N1 limit and the commanded N1 signal received from the PMCs, this signal being based on original lever position.

The PMC provides :

(a) Linearity between the throttle control lever position and the N1 command.

(b) Slaving of the engine rating (N1 actual) to the N1 command through the main engine control (MEC).

(Ref. Fig. 008)



TLA = THROTTLE LEVER ANGLE

PLA = POWER LEVER ANGLE

BM5 22 30 00 0 FNM2-07-09
DM5

TCC - PMC Operational Block Diagram
Figure 008

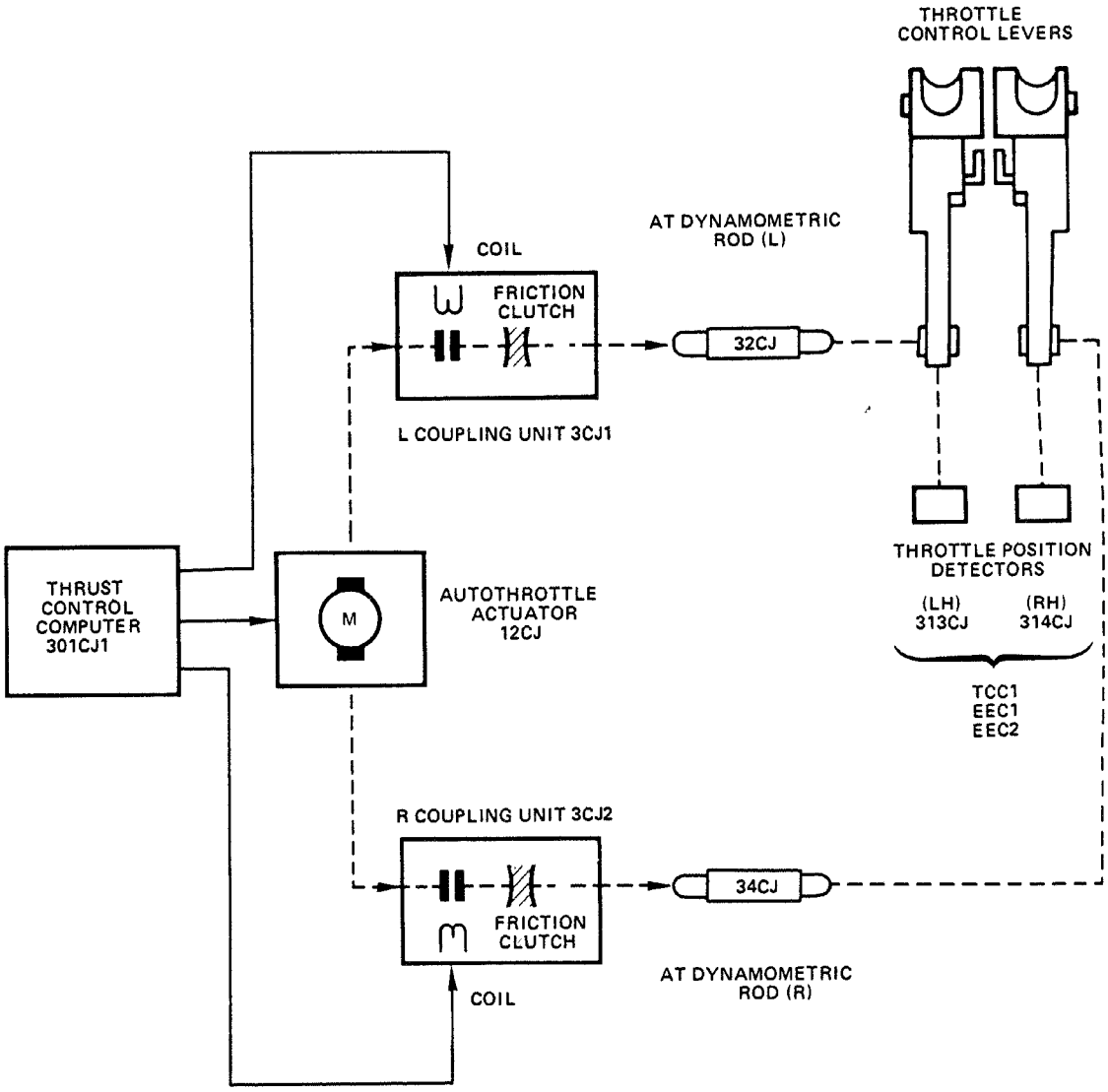
AUTOTHROTTLE - DESCRIPTION AND OPERATION

1. Introduction

The Thrust Control Computer System (TCCS) is a portion of the larger Automatic Flight System (AFS).
The functions of the Thrust Control Computer System are :

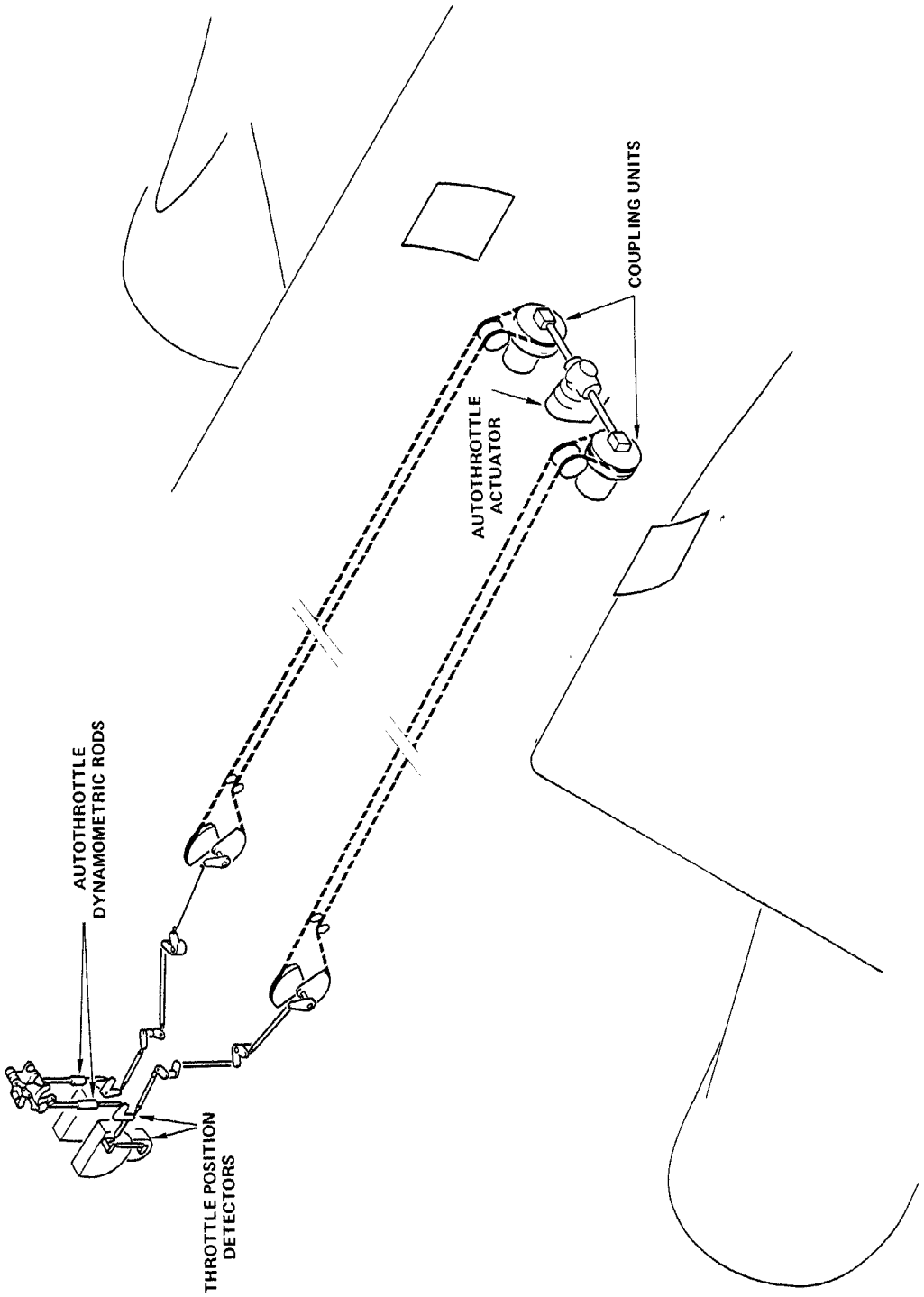
A. Permanent Computation of the EPR Limit Value Corresponding to the Flight Phase Selected on the Thrust Rating Panel (Thrust Modes) Function of Ambient Conditions
This value is given in digital form in the THR LIMIT display and in some cases in the TARGET display on the Thrust Rating Panel (TRP) and repeated by the bugs on the EPR indicators.

B. Autothrottle
(1)Acquisition and hold of the selected speed or Mach number through automatic selection by the FCC or FMC.
(2)Acquisition and hold of the limit thrust by :
(a)Actuating the TO/GA levers (Takeoff, Go Around).
(b)Automatic engagement (alpha floor).
(c)Engagement upon FCC request, when LVL/CH climb is initiated.
(3)Acquisition and hold of rating (TARGET) computed by the FMC in PROFILE mode.
(4)Decrease of thrust upon FCC (or FMC) order when LVL/CH descent is initiated or during flare out.
(5)Underspeed, overspeed and THR limit overshoot protection in SPD/MACH mode.
(Ref. Fig. 001)
(Ref. Fig. 002)



BM5 22 30 00 0 CAM2 21

TCC - General Arrangement
Figure 001



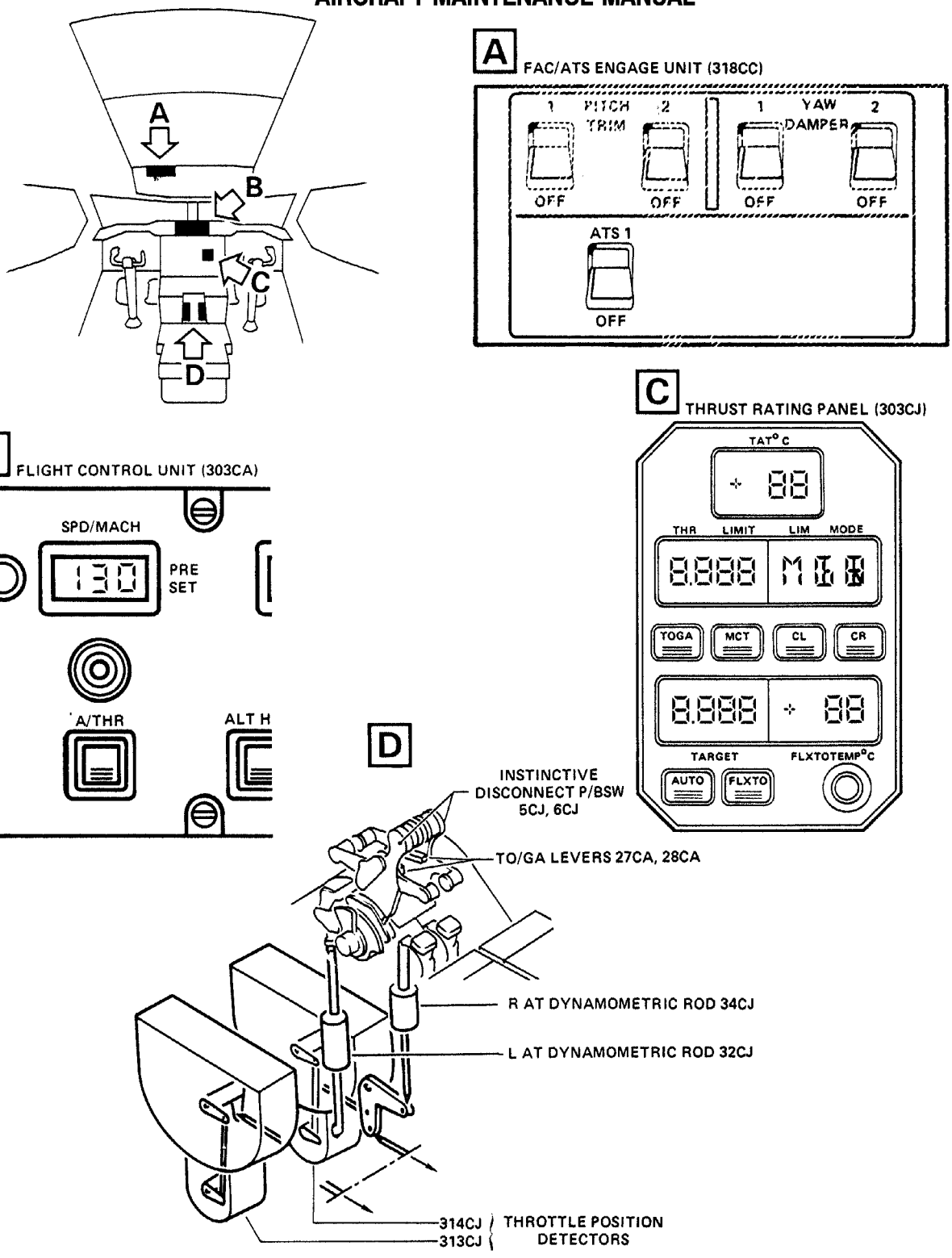
BM5 22 30 00 0 CCM2 21

TCC - Control Channel Layout
Figure 002

2. Component Location

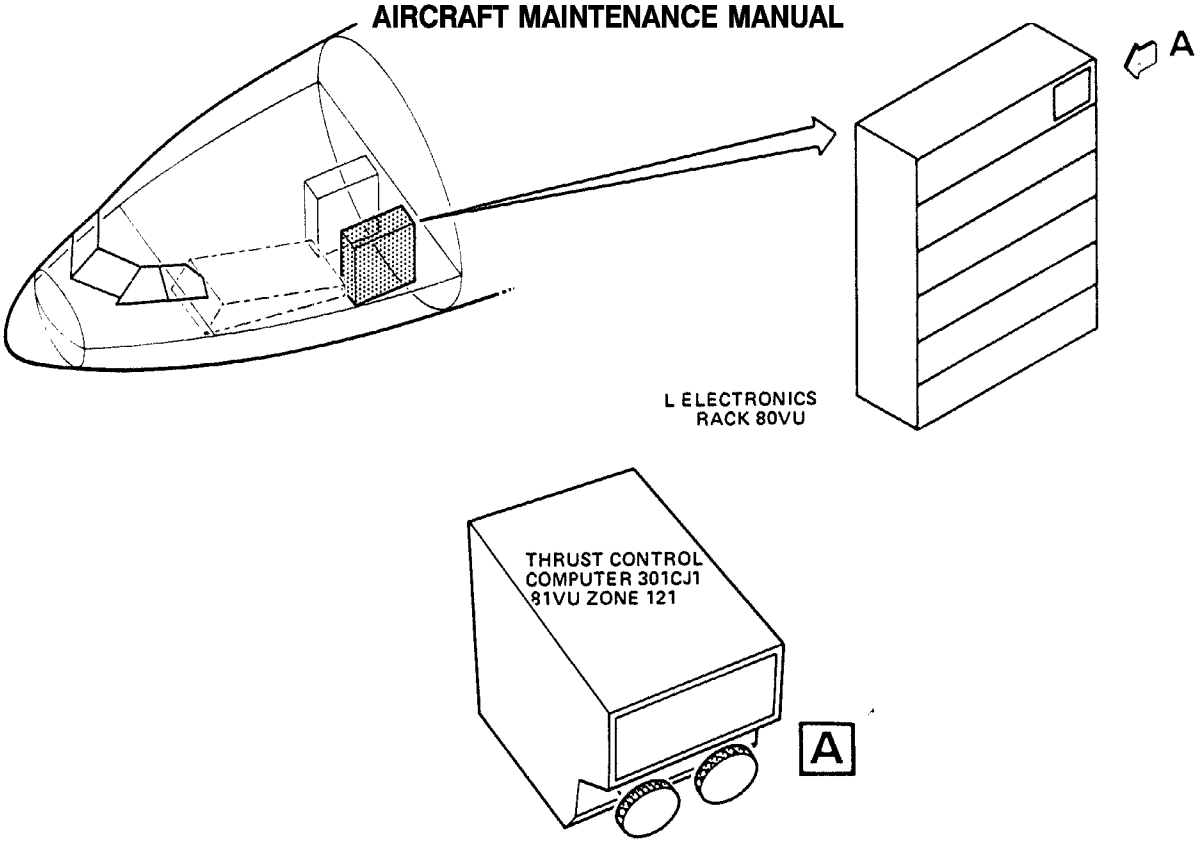
(Ref. Fig. 003)
(Ref. Fig. 004)

FIN	FUNCTIONAL DESIGNATION	PANEL	ZONE	ACCESS DOOR	ATA REF.
3CJ1	COUPLING UNIT-AT, L		141	131SW	22-33-28
3CJ2	COUPLING UNIT-AT, R		142	132SW	22-33-29
5CJ	P/BSW-INSTINCTIVE DISCONNECT, CAPT	9VU	211		
6CJ	P/BSW-INSTINCTIVE DISCONNECT, F/O	9VU	211		
12CJ	ACTUATOR-AUTOTHROTTLE		141	131RW	22-33-22
32CJ	ROD-AT DYNAMOMETRIC, L	9VU	211	121BL	76-11-21
34CJ	ROD-AT DYNAMOMETRIC, R	9VU	211	121BL	76-11-21
301CJ1	TCC-1	81VU	121	121BL	22-36-34
303CJ	TRP	4VU	211		22-35-12
313CJ	DET-THROTTLE POSITION, L	9VU	211	121BL	22-33-16
314CJ	DET-THROTTLE POSITION, R	9VU	211	121BL	22-33-16
27CA	LEVER-TO/GA, L	9VU	211		
28CA	LEVER-TO/GA, R	9VU	211		
34FN1	PFD-CAPT	3VU	211		34-72-22
34FN2	PFD-F/O	5VU	212		34-72-22



BM5 22 30 00 0 CFW2 00

TCC - Control and Indicating in Flight Compartment
Figure 003



BM5 22 30 00 0 CDM2 21

TCC - Component Location
Figure 004

3. Thrust Control Computer System Functions

A. Computation and Display of EPR Limit Value

Computation of the EPR Limit Value enables :

(1)Reduction of crew workload related to engine management, by removing the need to consult curves dependent on a great number of parameters ; moreover, the shorter the flight, the more frequent the consultation.

(2)Reduction of the risk of errors, achieved by monitoring computations.

(3)Reduction of the response time following a change in an external parameter (temperature, altitude, Mach) and highly accurate correction of the effects of air bleed and wing, nacelle and engine anti-icing conditions very accurately.

(4)Availability of an engine limit assisting the internal engine control.
This system enables the crew to perform a flexible takeoff by selecting a fictitious total air temperature higher than the actual temperature on the TRP.
Selection of this temperature is made in accordance with takeoff conditions (takeoff weight lower than maximum takeoff weight, runway length...).
Such a procedure results in an increase in engine component life, which is directly related to high-power operation.
Takeoff power may be selected between the maximum and minimum limits authorized by the engine manufacturer (FLX TO).

EPR limit value is obtained from the following data :
- altitude
- Mach number
- computed airspeed
- total air temperature
- total pressure
- antiice and air bleed conditions.

These computations are made for the six following flight modes :
- takeoff (TO)
- flexible takeoff (FLX TO)
- go around (GA)
- maximum continuous (single engine operation) (MCT)
- maximum cruise (CR)
- climb (CL).

Computation of the EPR limit value is detailed in 22-36.

In AUTO mode, management of the modes is made automatically by the flight management computer (FMC).

Display of the EPR limit value :

The EPR limit value is sent to the TRP and engine EPR indicators thus allowing comparison on these indicators between actual engine rating and maximum engine rating.

B. Automatic Thrust Control

The following modes are available if the ATS has been armed and if the A/THR mode has been selected on the FCU.

(1) SPEED/MACH control

In SPEED (or MACH) hold mode the system enables slaving of the aircraft speed (or Mach), transmitted by the ADC, to the reference value set on the FCU or in PROFILE mode to the target speed computed by the FMC (cyan index displayed on PFD speed scale).

SPEED or MACH mode is selected by pressing SPD/MACH pushbutton switch located on the left-hand side of SPD/MACH display.

The speed (Mach) reference value can be modified by rotating the selector knob located below the display.

The SPD/MACH mode is engaged mainly in response to FCC or FMC signals.

(Ref. Fig. 005)

(2) Thrust control

In THRUST mode the system enables slaving of the engine rating to the EPR limit value computed by the TCC or to the target value given by the FMC using information from the TRP (mode selection), the engines (EPR command/actual), air conditioning system and ADC parameters (TAT, Mach, altitude, Vc, total pressure).

- Takeoff thrust is set separately on each engine by means of clutches. After TO/GA lever activation both throttle control levers are driven together at a constant rate of + 3 °/s TLA until alignment of each EPR command on EPR limit is achieved (with a delay of 200 ms).
  When this condition is fulfilled on the leading engine, the corresponding clutch is deactivated.
  When this condition is fulfilled on the trailing engine, the 2nd clutch is deactivated, the actuator is stopped, and cyan THR replaces green THR on PFD.
  It is to be noted that the declutch logic is latched until landing gear is retracted or mode is changed on TRP.
- In flight after reclutch, the system makes the difference between the EPR limit value and the higher EPR command. The throttle control levers are driven together to align the higher EPR command on the EPR limit. When the PROFILE mode is engaged and active (P.THR on PFD).
- The THRUST mode is engaged in response to FCC or FMC signals.

(3) RETARD mode

The RETARD mode becomes active when the FCC delivers a descent command signal in the level change phase. In this mode the pilot can stop the throttle control levers before they reach the idle position by applying a load on either of them.

This mode is also active during flare out as soon as altitude reaches 30 ft.

The throttle control levers move to the idle position.

(Ref. Fig. 006)

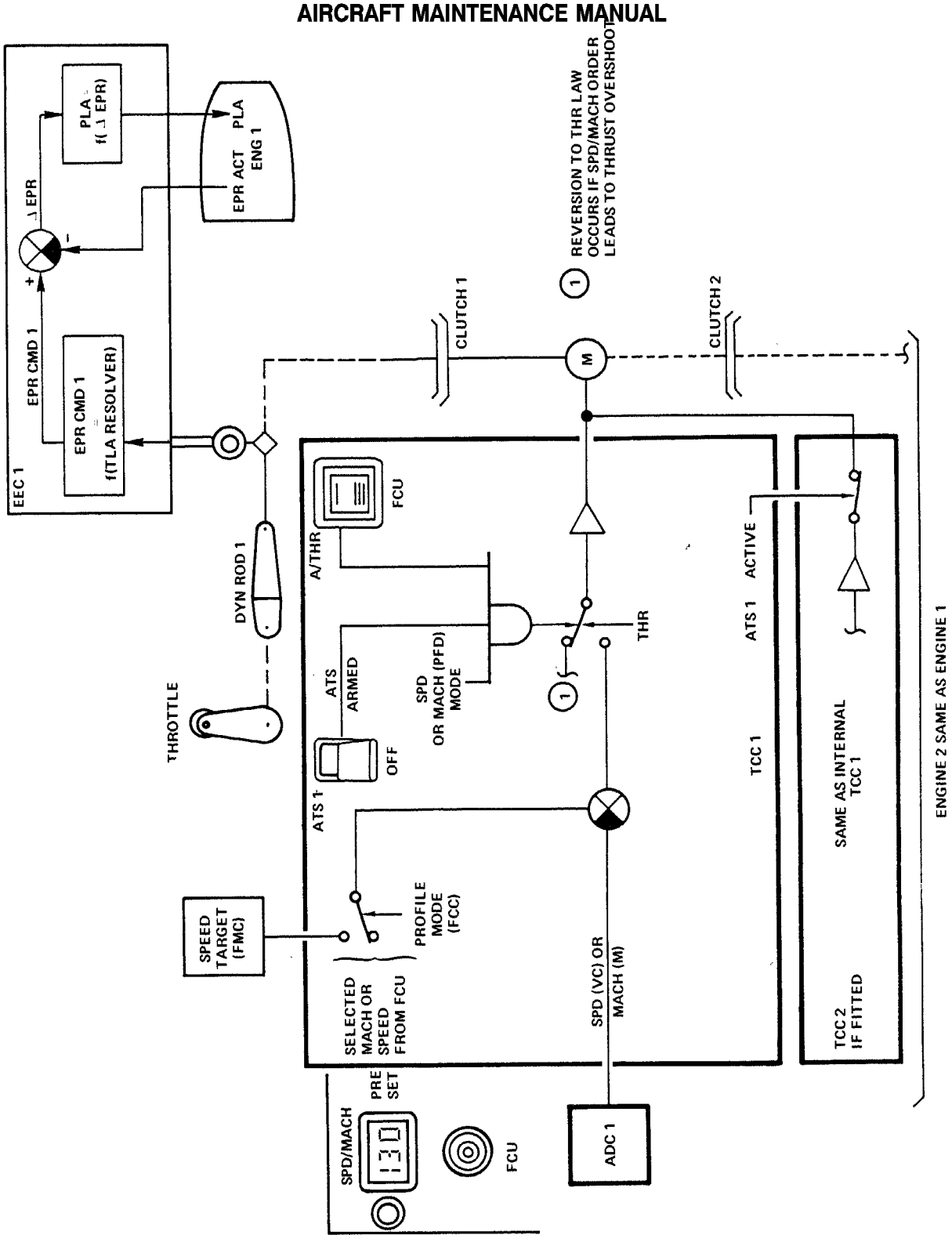
(Ref. Fig. 007)

C. Function and Operation of Electronic Engine Control

(1) Function

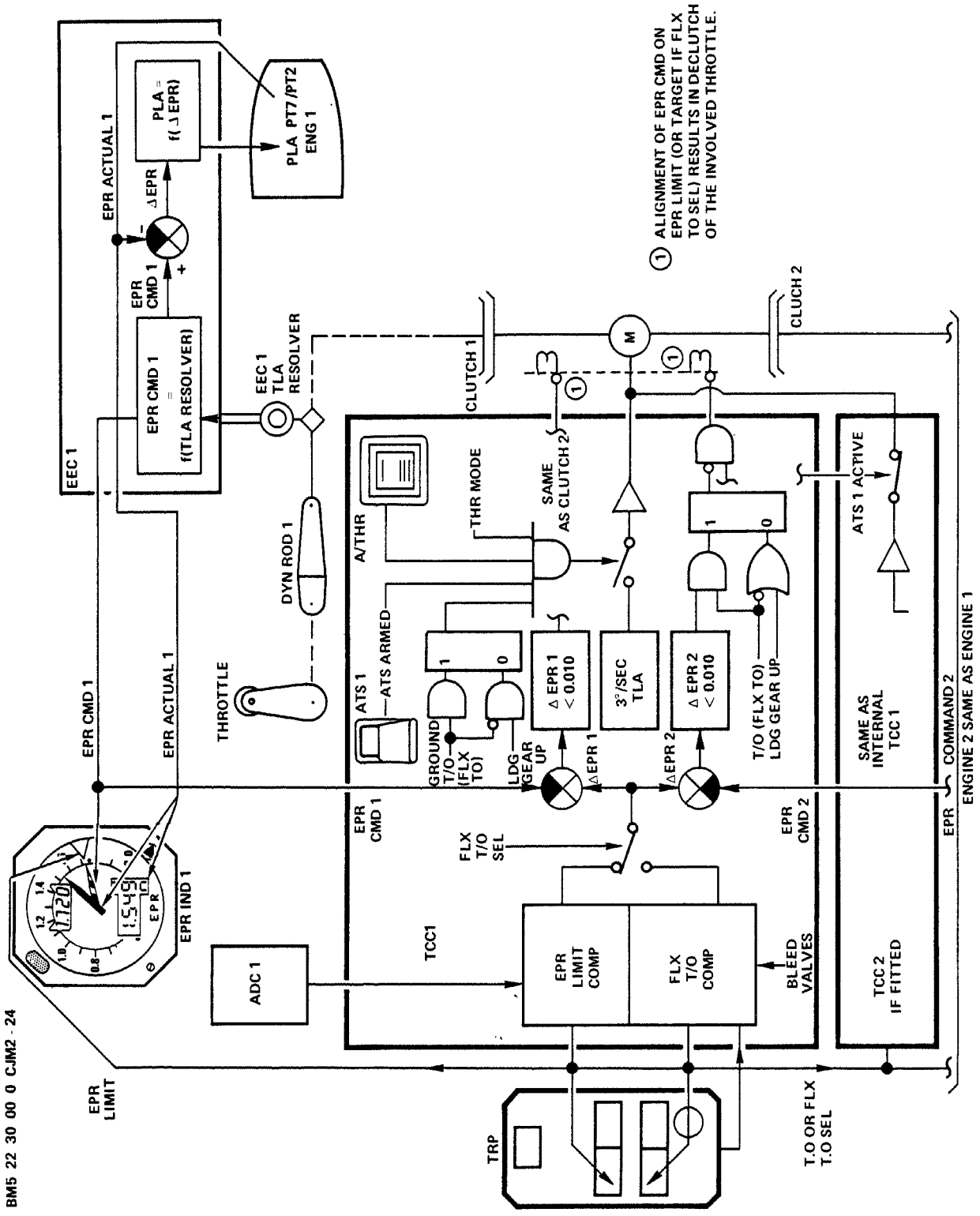
The electronic engine control (EEC) operates continuously during flight either in automatic control or in manual flight.

The EEC provides fuel control with a full authority.

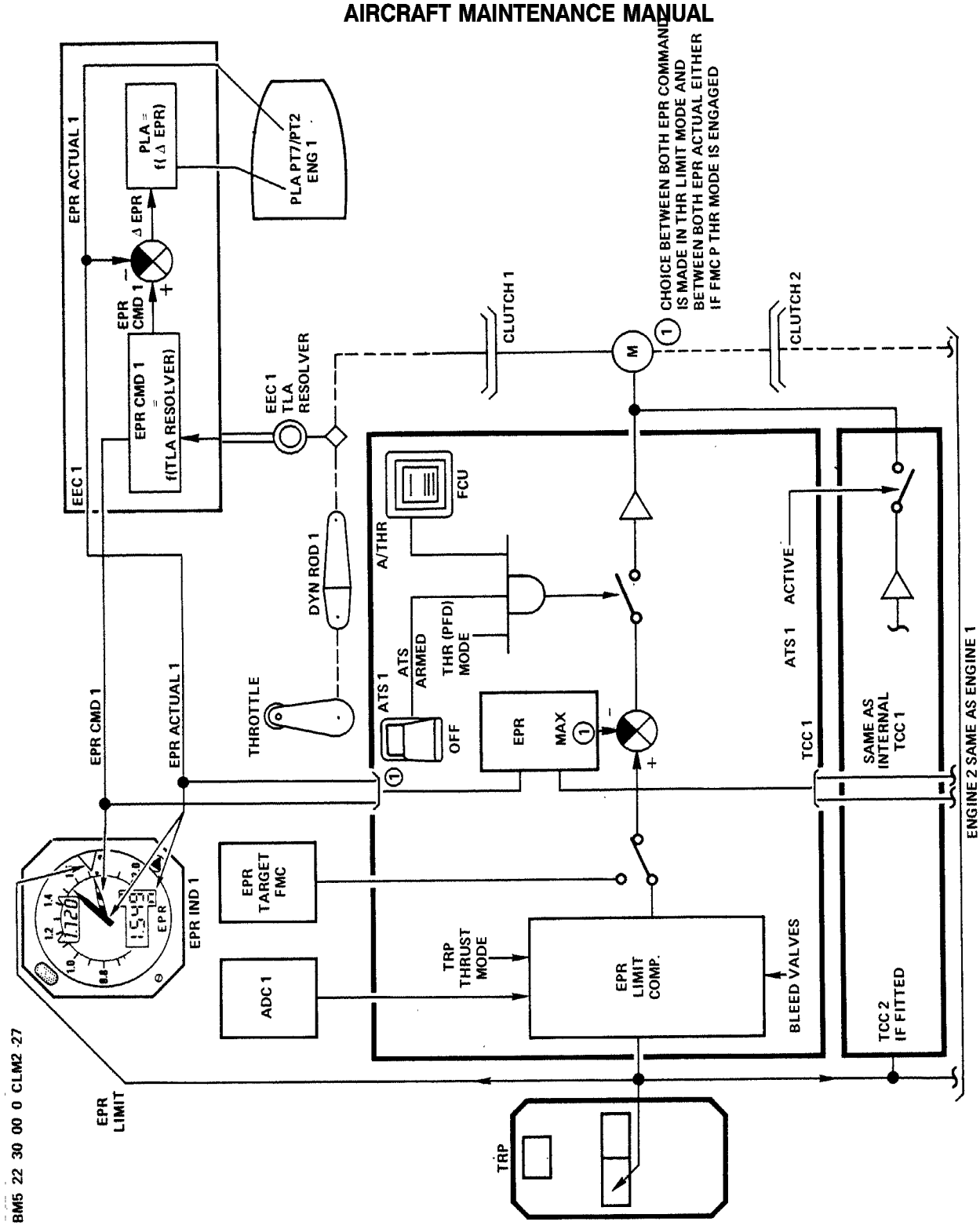


BM5 22 30 00 0 CHM2 21

SPEED/MACH Mode
Figure 005



Takeoff Thrust Setting
Figure 006



Cruise Thrust Mode
Figure 007

The function of this electronic is to simplify the engine control and provide protection against overspeed.

(2)Operation

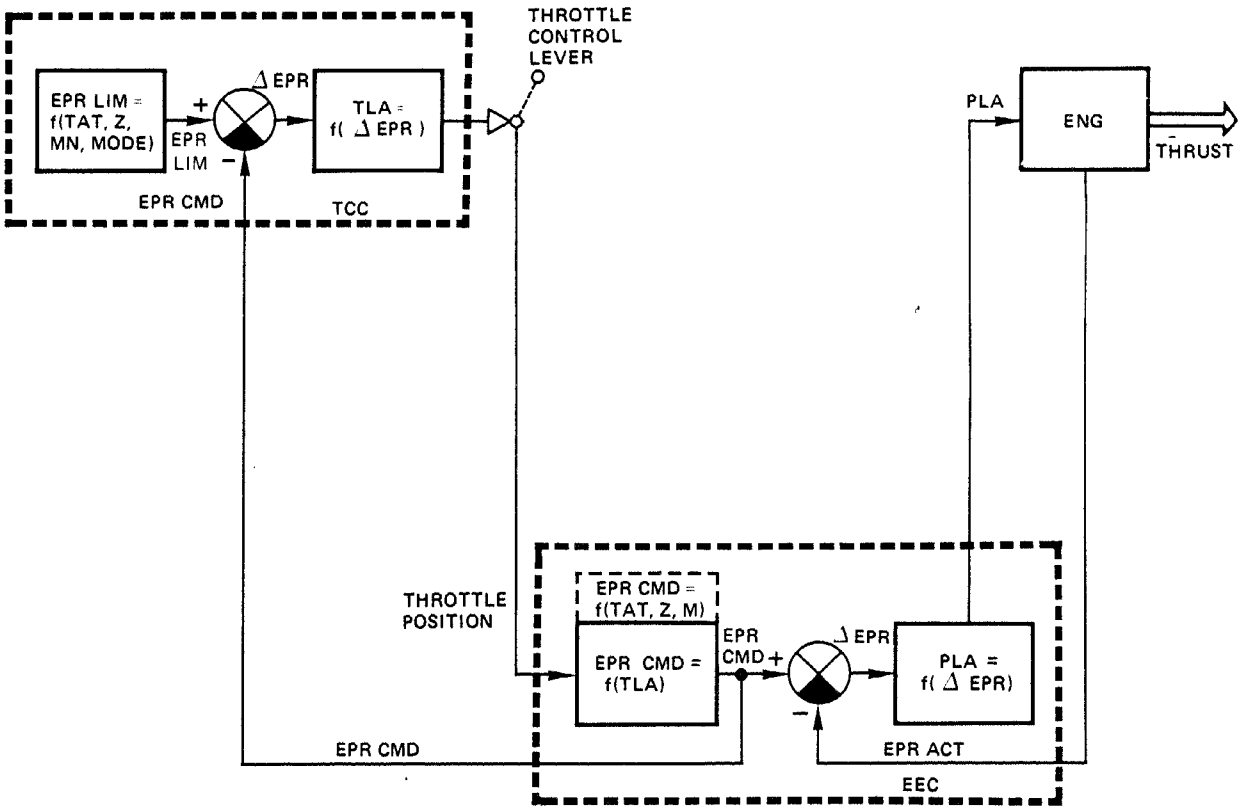
After a mode selection, the TCC generates a lever displacement signal. This signal is based on the difference computed by the TCC between EPR limit and the commanded EPR signal received from the EECs, this signal being based on original lever position.

The EEC provides :

(a)Linearity between the throttle control lever position and the EPR command.

(b)Slaving of the engine rating (EPR actual) to the EPR command.

(Ref. Fig. 008)



BM5 22 30 00 0 CNM2 21

TLA = THROTTLE LEVER ANGLE

PLA = POWER LEVER ANGLE

TCC - EEC Operational Block Diagram
Figure 008

1. Operational Test

A. Alpha Floor Function

(1)Reason for the job

To check alpha floor protection for correct operation by means of the alpha probe test.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS,MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN)
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION AND ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that flaps and slats are retracted
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that ADS is operating
- make certain that pitch trim control wheel is in neutral position
- make certain that PITCH TRIM 1 lever is in ON position

(b)Test

ACTION	RESULT
1. On center pedestal - place throttle control levers in 15° position.	
2. On overhead panel - place ATS arming lever in	Lever remains in this position. On primary flight display

ACTION	RESULT
ON position.	- amber **MAN THR** comes on.
3. On **ALPHA PROBES** section of maintenance panel 47OVU - press **ALL** pushbutton switch.	On this pushbutton switch, - **TEST** legend comes on. On primary flight display, - **MAN THR** goes off - green **THR L** comes on steady. On center pedestal, - throttle control levers move to forward throttle position and stop at full position. Stick shaker is activated. Pitch Trim control wheel rotates in nose down direction due to **ALPHA TRIM** activation.
- press **ALL** pushbutton switch again.	On this pushbutton switch - **TEST** legend goes off. On **PFD** - green **THR L** flashes. Pitch trim control wheel returns to 0° position.
4. Press **AT** instinctive disconnect pushbutton switch.	On **PFD** - green **THR L** goes off - amber **MAN THR** comes on.
5. On overhead panel - place **PITCH TRIM2** lever in **ON** position - place **PITCH TRIM1** lever in **OFF** position.	- Lever remains in this position.
6. On center pedestal - bring back the throttle control levers to 15° position.	
7. Repeat actions described in steps 3 and 4.	Same as steps 3 and 4.
8. Place **ATS** arming lever in **OFF** position.	
9. De-energize **AFS** components by placing **PITCH TRIM2** lever in **OFF** position.	

(c)Close-up
1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
2 Restore aircraft and system to normal operating condition.
3 Make certain that throttle control levers are in idle position.

B. Actuator Clutches and TO/GA Levers Operation
(1)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(2)Procedure
WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN)
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION AND ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that EFIS is operating and adjust PFD image hightness on panels 412VU and 413VU
- make certain that ADS is operating
- make certain that PITCH TRIM 1 lever is in ON position

(b)Test
NOTE 1 : Test described below concerns left throttle control lever, left coupling unit and left dynamometric rod. For right side, which is identical, refer to information between parentheses.
NOTE 2 : To check L (R) TO/GA lever after removal/installation, test must be performed twice :
- first with circuit breaker 306CA2 open (panel 21VU, AFS/AP2-FD2/FCC2/28VDC, line 102/H6)
- a second time with circuit breaker 306CA1 open (panel 21VU, AFS/AP1-FD1/FCC1/28VDC, line 102/H3).

ACTION	RESULT
1. On center pedestal - place throttle control levers in 0° position.	
2. On overhead panel - place ATS arming lever in ON position.	Lever remains in this position.

ACTION	RESULT
3. On center pedestal - press TO/GA lever on L or R throttle control lever	Check that - on center pedestal : both throttle levers start to move.
4. During throttle displacement apply progressively a force on L (R) throttle control lever to retain it	Check that passing load threshold, L (R) throttle control lever moves freely.
5. Release L (R) throttle control lever	Check that L (R) throttle control lever moves again to full throttle.
6. Press left AT instinctive disconnect pushbutton switch and place both throttle control levers to 0° position.	Check that throttle control levers stop.
7. On center pedestal - press TO/GA lever on L (R) throttle control lever	Check that throttle control levers start to move again.
8. Press right AT instinctive disconnect pushbutton switch	Check that throttle control levers stop.
9. Place ATS arming lever in OFF position.	
10. De-energize AFS components by placing PITCH TRIM 1 lever in OFF position.	

(c)Close-up
1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
2 Restore aircraft and system to normal operating condition.
3 Make certain that throttle control levers are in idle position.

C. EPR Limit Computation
(1)Reason for the job
In TAKE OFF mode, for one TAT, read EPR limit computed by TCC and displayed on ENG EPR indicators.
(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(3)Procedure
WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT

ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up
- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN)
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION AND ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- on EPR indicator, make certain that EPR limit index is in automatic mode (knob pressed and upper counter blank).
- make certain that ADS is operating.

(b)Test

ACTION	RESULT
1. On overhead panel - place PITCH TRIM 1 lever in ON position	Lever remains in this position. On thrust rating panel - TO/GA pushbutton switch comes on - LIM MODE display shows T.O. - TAT display shows local TAT - THR LIMIT display shows a value compatible with static pressure and TAT. On ENG EPR indicators - EPR limit index is set to the same THR limit value as that displayed on TRP
2. De-energize AFS components by placing PITCH TRIM 1 lever in OFF position.	

(c)Close-up

1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

2 Restore aircraft and system to normal operating condition.

D. TAT Display

(1)Reason for the job

To check automatic switching of TAT value on thrust rating panel from ADC1 in case of TCC failure.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN)
- close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION AND ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- make certain that ADS is operating.
- remove slip on cover from first officer TAT probe, zone 122, STA981.

(b)Test

ACTION	RESULT
1. On panel 22VU, - open circuit breaker 5RK (line 204/F27)	On thrust rating panel - TAT value appears.
2. On thrust rating panel, - note TAT value	This is the first reading.
3. On panel 132VU, - open F/O PROBES SUPPLY circuit breakers F/O PITOT (line 322/N67), R STAT (line 322/N69) and L STAT (line 323/M66).	
4. On panel 22VU, - open circuit breaker 9FL1 (line 208/B19).	
5. On panel 133VU, - open circuit breaker 1GB (line 331/V52).	
6. On PROBE HEAT section of panel 433VU, - press F/O pushbutton switch	On this pushbutton switch - white OFF legend goes off On thrust rating panel - TAT value increases.
- press F/O pushbutton switch again when TAT value exceeds the other TAT value (first reading) by 10°	

ACTION	RESULT
7. On panel 133VU, - close circuit breaker 1GB (line 331/V52).	
8. On thrust rating panel, - note new TAT NOTE : These values are not fixed as the probe ambient temperature decreases.	- This is the second reading.
9. On panel 21VU, - open circuit breaker 306CJ1 (line 102/H8)	On thrust rating panel - TAT display shows---
10.On panel 22VU, - close circuit breaker 9FL1 (line 208/B19)	On thrust rating panel - TAT display shows the first reading
11.On panel 21VU, - close circuit breaker 306CJ1 (line 102/H8).	No effect.

(c)Close-up

1 On panel 132VU, close F/O PROBES SUPPLY circuit breakers previously open.

2 On panel 22VU, close circuit breaker 5RK (line 204 F/27).

3 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

4 Restore aircraft and system to normal operating condition.

E. ATS OFF Warning

(1)Reason for the job

To check ATS OFF warning for correct operation.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)

- make certain that electronics racks ventilation is correct
- close overhead panel circuit breakers particularly those associated with **AFS, NAVIGATION** and **ADS.**
- close **L/G PROX DET & RELAYS** circuit breakers on panel **133VU** (aircraft in ground configuration).
- make certain that **EFIS** is operating and adjust **PFD** image brightness on panels **412VU** and **413VU**
- make certain that **ADS** is operating
- place **PITCH TRIM 1** lever in **ON** position
- make certain that flaps and slats are retracted.
- make certain that pitch trim control wheel is in neutral position
- align **IRS1, 2** and **3** (Ref. 34-25-00, P. Block 501) (Perform **ALIGN** procedure until red **HDG** flags on **ND**s disappear ; present position entering is not necessary)
- make certain that **ATS** lever is in **OFF** position.
- perform recall action on the **ECAM** control panel.

(b)Test

ACTION	RESULT
1. On overhead panel - place **ATS** arming lever in **ON** position.	Lever remains in this position. On **PFD**s, - amber **MAN THR** comes on.
- place **ATS** arming lever in **OFF** position.	On **PFD**s, - amber **MAN THR** goes off. On each Captain and First Officer main instrument panel, - amber **MASTER CAUTION** light comes on. On left **ECAM** display unit, - amber **ATS OFF** message appears. In Captain and First Officer loudspeakers, - single chime sounds.
2. On **ECAM** control panel : - press **CLR** pushbutton switch as many times as necessary to call-up **MEMO** page.	All warnings are cancelled.

(c)Close-up

- De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Restore aircraft and system to normal operating condition.

1. Operational Test

A. Alpha Floor Function

(1)Reason for the job

To check alpha floor protection for correct operation by means of the alpha probe test.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- make certain that hydraulic power is available (GREEN)

NOTE : As an alternative it is possible to pressurize the hydraulic system using Green system electric pumps (Ref. 29-21-00, P. Block 301).

- close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that flaps and slats are retracted
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that ADS is operating
- make certain that pitch trim control wheel is in neutral position
- make certain that PITCH TRIM 1 lever is in ON position.
- make certain that ENG TRIM pushbutton switch on FAC/ATS engage unit is in OFF position.

R (b)Test

R ACTION	RESULT
R 1. On center pedestal:	
R - place throttle control levers in	

ACTION	RESULT
15° position.	
2. On overhead panel: - place ATS1 arming lever in ON position.	Lever remains in this position. On Primary Flight Display (PFD): - amber MAN THR message appears.
3. On ALPHA PROBES section of maintenance panel 470VU: - press ALL pushbutton switch	On this pushbutton switch: - TEST legend comes on. On PFD: - MAN THR message disappears - green THR L indication comes on steady. On center pedestal: - throttle control levers move to forward throttle position and stop at 33Δ. Stick shaker is activated. Pitch trim control wheel rotates in nose down direction due to ALPHA TRIM activation.
- press ALL pushbutton switch again	On this pushbutton switch: - TEST legend goes off. On PFD: - green THR L indication flashes. Pitch trim control wheel returns to 0° position.
4. Press AT instinctive disconnect pushbutton switch	On PFD: - green THR L indication goes off. - amber MAN THR message appears.
5. On center pedestal: - place throttle control levers in 15° position.	
6. On overhead panel: - place ATS1 arming lever in OFF position - place ATS2 arming level in ON position	On PFD: - MAN THR message disappears. Lever remains in this position. On PFD: - amber MAN THR message appears.
7. Repeat actions described in step 3 and then in step 4	Same as steps 3 and 4.
8. On overhead panel: - place PITCH TRIM2 lever in ON position	Lever remains in this position.

ACTION	RESULT
- place PITCH TRIM1 lever in OFF position	
9. Repeat actions described in step 3 and then in step 4	Same as steps 3 and 4.
10. On overhead panel: - place ATS2 arming lever in OFF position - place ATS1 arming lever in ON position	On PFD: - MAN THR message disappears. Lever remains in this position. On PFD: - amber MAN THR message appears.
11. Repeat actions described in step 3 and then in step 4	Same as steps 3 and 4.
12. Place ATS1 arming lever in OFF position.	
13. De-energize AFS components by placing PITCH TRIM2 lever in OFF position.	

(c)Close-up

1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

2 Restore aircraft and system to normal operating condition.

3 Make certain that throttle control levers are in idle position.

B. Actuator Clutch and TO/GA Lever Operation

(1)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(2)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)

- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that ADS is operating
- place PITCH TRIM 1 lever in ON position

(b)Test

NOTE 1 : The test described below concerns ATS1. For ATS2 which is identical, refer to information in parentheses.

NOTE 2 : The test described below concerns left throttle control lever, left coupling unit and left dynamometric rod. For right side, which is identical, refer to information in parentheses.

NOTE 3 : To check LH (RH) TO/GA lever after removal/installation, test must be performed twice :
- first with circuit breaker 306CA2 open (panel 21VU, AFS/AP2-FD2/FCC2/28VDC, line 102/H6)
- a second time with circuit breaker 306CA1 open (panel 21VU, AFS/AP1-FD1/FCC1/28VDC, line 102/H3).

ACTION	RESULT
1. On center pedestal: - place throttle control levers in 0° position.	
2. On overhead panel: - place ATS1 (2) arming lever in ON position.	Lever remains in this position.
3. On center pedestal: - press TO/GA lever on LH or RH throttle control lever	Check that: - on center pedestal : both throttle levers start to move.
4. During throttle displacement apply progressively a force on LH (RH) throttle control lever to retain it	Check that passing load threshold, LH (RH) throttle control lever moves freely
5. Release LH (RH) throttle control lever	Check that LH (RH) throttle control lever moves again to full throttle.
6. Press left AT instinctive disconnect pushbutton switch and place both throttle control levers to 0° position.	Check that throttle control levers stop.
7. On center pedestal: - press TO/GA lever on LH (RH) throttle control lever	Check that throttle control levers start to move again.
8. Press right AT instinctive disconnect pushbutton switch	Check that throttle control levers stop.
9. Place ATS1 (2) arming lever in OFF position.	

ACTION	RESULT
10. De-energize AFS components by placing PITCH TRIM1 lever in OFF position. (c) Close-up 1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301). 2 Restore aircraft and system to normal operating condition. 3 Make certain that throttle control levers are in idle position.	

C. N1 Limit Computation

(1) Reason for the job

In TAKE OFF mode, for one TAT, read N1 limit computed by TCC1 then by TCC2 and displayed on ENG N1 indicators.

(2) Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(3) Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a) Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct
- close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS
- make certain that ADS is operating.
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- on N1 indicator make certain that N1 limit index is in automatic mode (knob pressed and upper counter blank).

(b) Test

ACTION	RESULT
1. On overhead panel: - place PITCH TRIM 1 lever in ON position	Lever remains in this position
- place ATS1 arming lever in ON position	Lever remains in this position. On thrust rating panel: - TO/GA pushbutton switch comes on - LIM MODE display shows TO - TAT display shows local TAT (note this TAT value)

ACTION	RESULT
	- THR LIMIT display shows a value compatible with static pressure and TAT (note this THR limit value). On ENG N1 indicators: - N1 limit index is set to the same THR limit value as that displayed on TRP
- place ATS1 arming lever in OFF position - place ATS2 arming lever in ON position	Lever remains in this position. On thrust rating panel: - TAT display shows same TAT value On ENG N1 indicators: - N1 limit index is set to the same THR limit value as that displayed on TRP
- place ATS2 arming lever in OFF position.	

2. De-energize AFS components by placing PITCH TRIM 1 lever in OFF position.

(c)Close-up

1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

2 Restore aircraft and system to normal operating condition.

D. TAT Display

(1)Reason for the job

To check automatic switching of TAT value on thrust rating panel from ADC1 in case of TCC failure.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301)
- make certain that electronics racks ventilation is correct

- close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that ADS is operating
- remove slip-on cover from First Officer TAT probe, zone 122, STA981.

(b)Test

ACTION	RESULT
1. On panel 21 VU: - Open circuit breaker 306CJ2 (line 102/H12).	
2. On panel 22 VU: - Open circuit breaker 5RK (line 204/F27).	On thrust rating panel: - TAT value appears.
3. On thrust rating panel: - note TAT value	This is the first reading
4. On panel 132VU: - open F/O PROBES SUPPLY circuit breakers F/O PITOT (line 322/N67), R STAT (line 322/N69) and L STAT (line 323/M66).	
5. On panel 22VU: - open circuit breaker 9FL1 (line 208/B19).	
6. On panel 133VU: - open circuit breaker 1GB (line 331/V52).	
7. On PROBE HEAT section of panel 433VU: - press F/O pushbutton switch	On this pushbutton switch - White OFF Legend goes off On thrust rating panel - TAT value increases
- press F/O pushbutton switch again when TAT value exceeds the other TAT value (first reading) by 10°	
8. On panel 133VU: - close circuit breaker 1GB (line 331/V52)	
9. On thrust rating panel:	

ACTION	RESULT
- note new TAT	This is the second reading
NOTE : These values are not fixed as the probe ambient temperature decreases.	
10.On panel 21VU: - open circuit breaker 306CJ1 (line 102/H8).	On thrust rating panel: - TAT display shows ---.
11.On panel 22VU: - close circuit breaker 9FL1 (line 208/B19).	On thrust rating panel: - TAT display shows the first reading.
12.On panel 21VU: - close circuit breaker 301CJ1 (line 102/H8).	No effect.

(c)Close-up

1 On panel 132VU, line 322 and 323 close F/O PROBES SUPPLY circuit breakers.

2 On panels 21VU and 22VU, close circuit breakers 306CJ2 (line 102/H12) and 5RK (line 204/F27).

3 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

4 Restore aircraft and system to normal operating condition.

E. ATS OFF Warning

(1)Reason for the job

To check ATS OFF warning for correct operation.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- make certain that electronics racks ventilation is correct
- close overhead panel circuit breakers particularly those associated with AFS, NAVIGATION and ADS.

- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration)
- make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU
- make certain that ADS is operating
- place PITCH TRIM 1 lever in ON position
- make certain that flaps and slats are retracted
- make certain that pitch trim control wheel is in neutral position
- align IRS 1, 2 and 3 (Ref. 34-25-00, P. Block 501) (Perform ALIGN procedure until red HDG flags on ND's disappear). Present position entering is not necessary.
- make certain that ATS 1 and ATS 2 levers are in OFF position.
- perform recall action on the ECAM control panel.

(b)Test

NOTE : The test described below concerns ATS 1. For ATS 2 which is identical, refer to information in parentheses.

ACTION	RESULT
1. On overhead panel:	
- place ATS 1 (2) arming lever in ON position.	Lever remains in this position. On PFDs: - amber MAN THR message appears. On left ECAM Display Unit: - ATS 1 (2) OFF ------ RESET message appears with ATS 1 (2) OFF in amber and ------ RESET in cyan.
- place ATS 1 (2) arming lever in OFF position.	On PFDs: - amber MAN THR message disappears. On each CAPT and F/O main instrument panel: - amber MASTER CAUTION light comes on. On left ECAM Display Unit: - ATS 1 (2) OFF message appears. On Captain's and First Officer's loudspeakers: - a single chime sounds.
2. On ECAM control panel: - press CLR pushbutton switch as many times as necessary to call up MEMO page.	All warnings are cancelled.

(c)Close-up

- De energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Restore aircraft and systems to normal operating condition.

1. Aircraft Electrical Power Supplies

A. 115 V/400 Hz Power Supply

TCC1 is supplied with 115 V/400 Hz through circuit breaker 308CJ1.
TCC2 (if fitted) is supplied with 115 V/400 Hz through circuit breaker 308CJ2.

In each TCC this voltage is applied to a center tap transformer whose output signal at each winding is 20 V.
Both voltages are rectified and filtered so as to obtain the ±24 V supplying the actuator.
(Ref. Fig. 001)

B. 26 V/400 Hz Power Supply

The TCC(s), dynamometric rods and throttle position detectors are supplied with 26 V/400 Hz through circuit breaker 305CJ.

(1) Dynamometric rods 32CJ and 34CJ

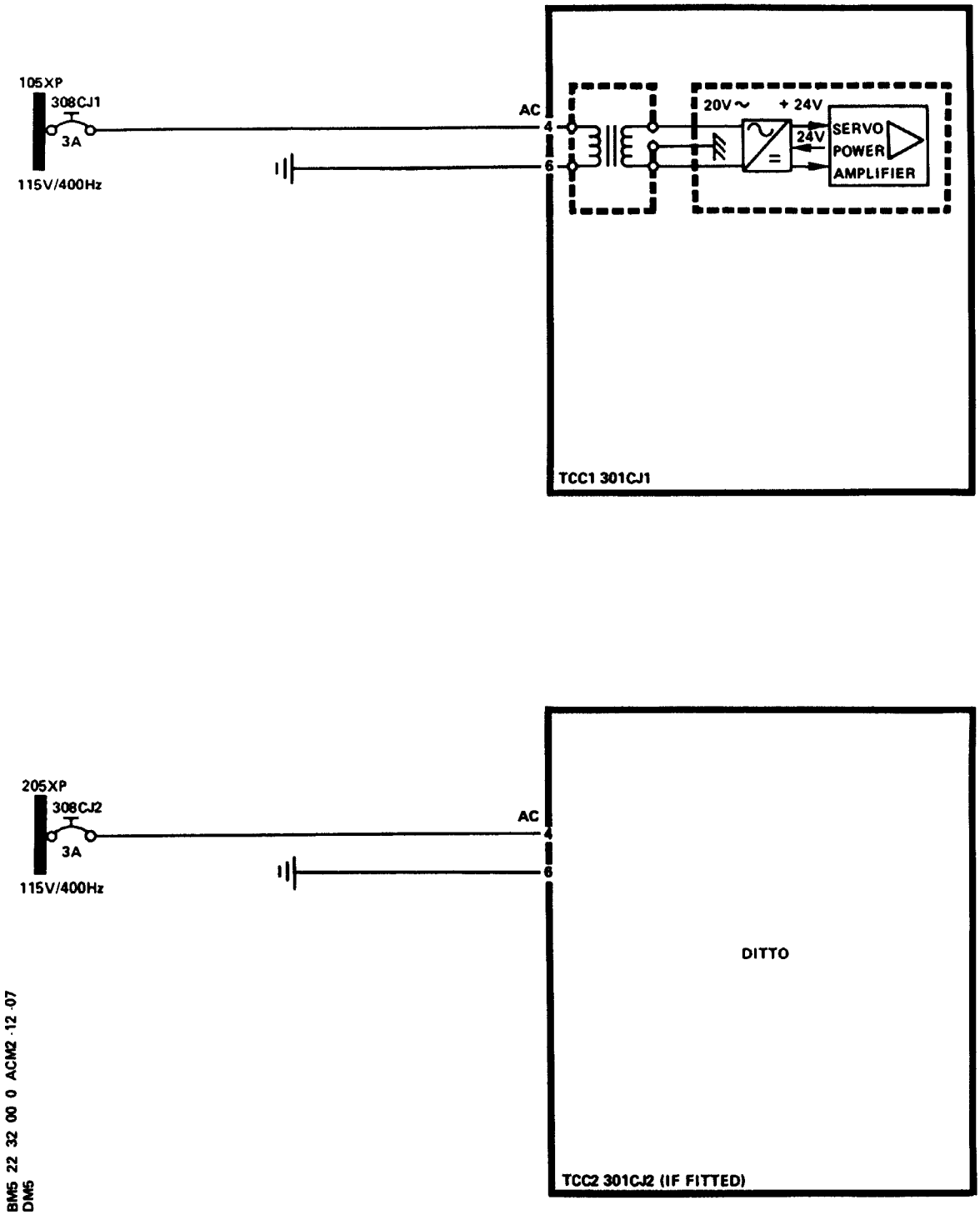
For each dynamometric rod, the 26VAC power supply is used to energize the primary sensor detecting loads applied by the pilot to the control linkage. (For further details concerning operation, see 22-33-1-E).

(2) Thrust control computer (TCC)

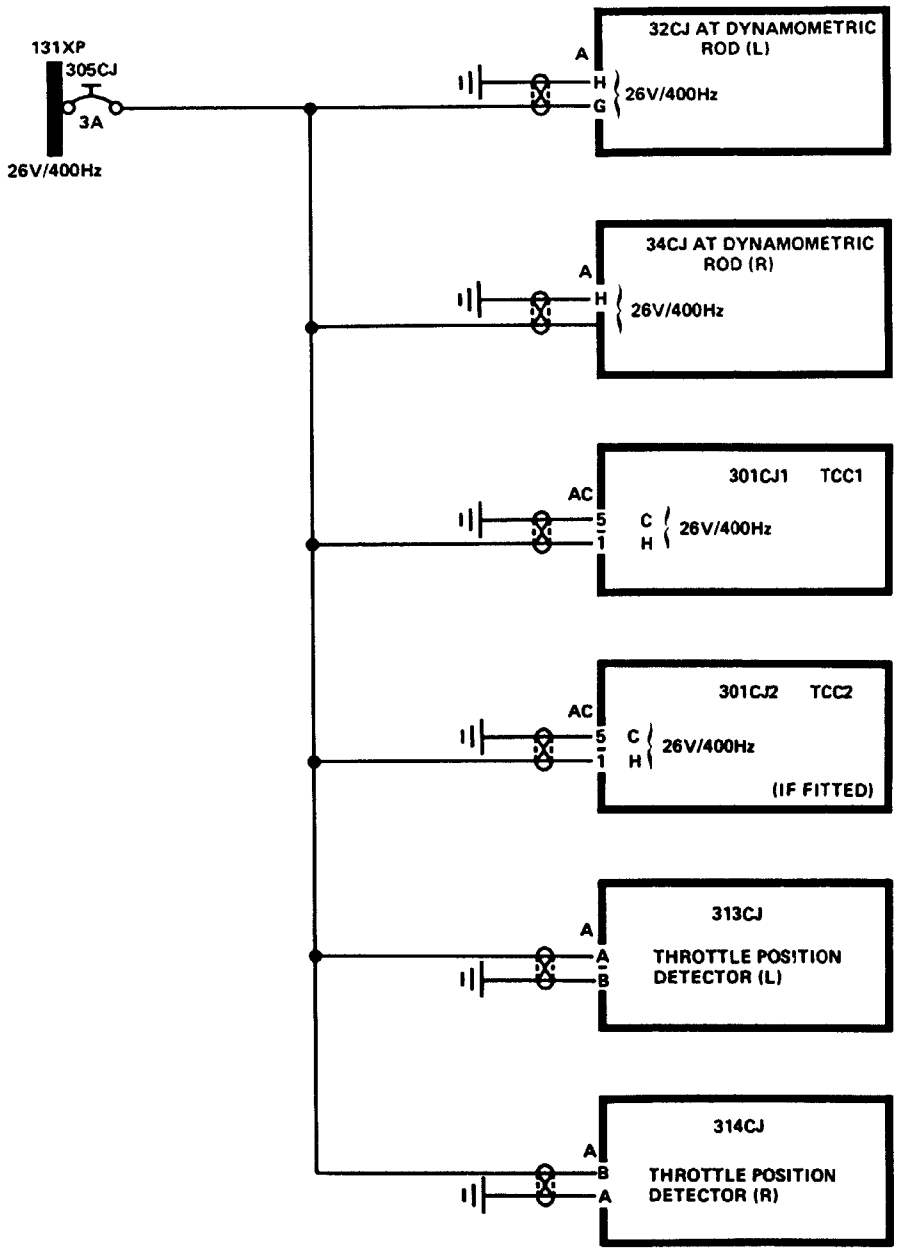
The 26 V/400 Hz power supply is adapted, rectified and filtered and the resultant signal is transmitted to the output and input conversion boards. The signal is compared to a 26 V reference signal. Should the difference between both signals exceed 5.2 V the TCC(s) is(are) disarmed The system is identical on the monitoring channel and acts on the arming condition of the monitoring channel.

(3) Throttle position detectors

The 26 V/400 Hz power supply is used to supply the throttle position detectors driven by the throttle control levers.
(Ref. Fig. 002)
(Ref. Fig. 003)

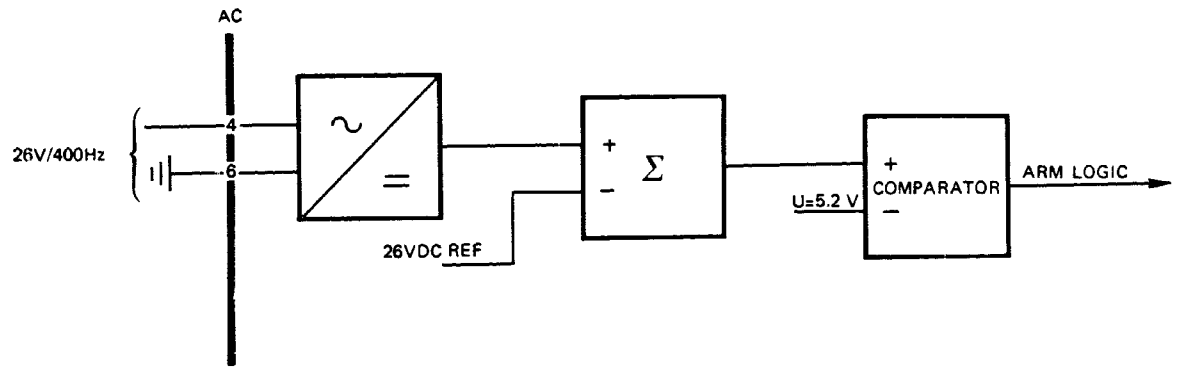


115 V/400Hz Electrical Power Supply
Figure 001



BM5 22 32 00 0 AEM2 12 07
DM5

26 V/400Hz Electrical Power Supply
Figure 002



BM5 22 32 00 0 AGM2

26 V/400 Hz - Monitoring
Figure 003

2. TCC Internal Power Supply Units and Power Cut Off Management

A. 28VDC Power Supply

TCC1 is supplied with 28VDC through circuit breaker 306CJ1.
TRP is supplied with 28VDC through circuit breaker 36CJ.
TCC2
(if fitted) is supplied with 28VDC through circuit breaker 306CJ2.
After filtering, the 28V power supply is sent to the following :
- ±15V converters (command and monitoring)
- +5V converters (command and monitoring)
- TO/GA levers 27CA and 28CA
- AT instinctive disconnect pushbutton switches 5CJ and 6CJ.
- connector wiring for engine reference encoding
- primary power supply boards (command and monitoring).

The CPU, memory, logic and ARINC interface boards located in the command channel are supplied by the converters located in the command channel.
Distribution is the same for the monitoring channel.
The voltages delivered by the converters are monitored on each command and monitoring primary power supply board. Regulations of +5 and +15 V are performed on each board and are used as a reference by comparators which monitor the 5 V power. If the converter 5 V power is outside a predetermined range (4.4 V - 5.6 V) a signal is fed to the detection logic which ensures power supply cutoff management and sends the various command signals to the memory and program boards. A relay cuts the board +28 V power supply when power supply cutoff on the ground is detected and engines are shut down.
(Ref. Fig. 004)

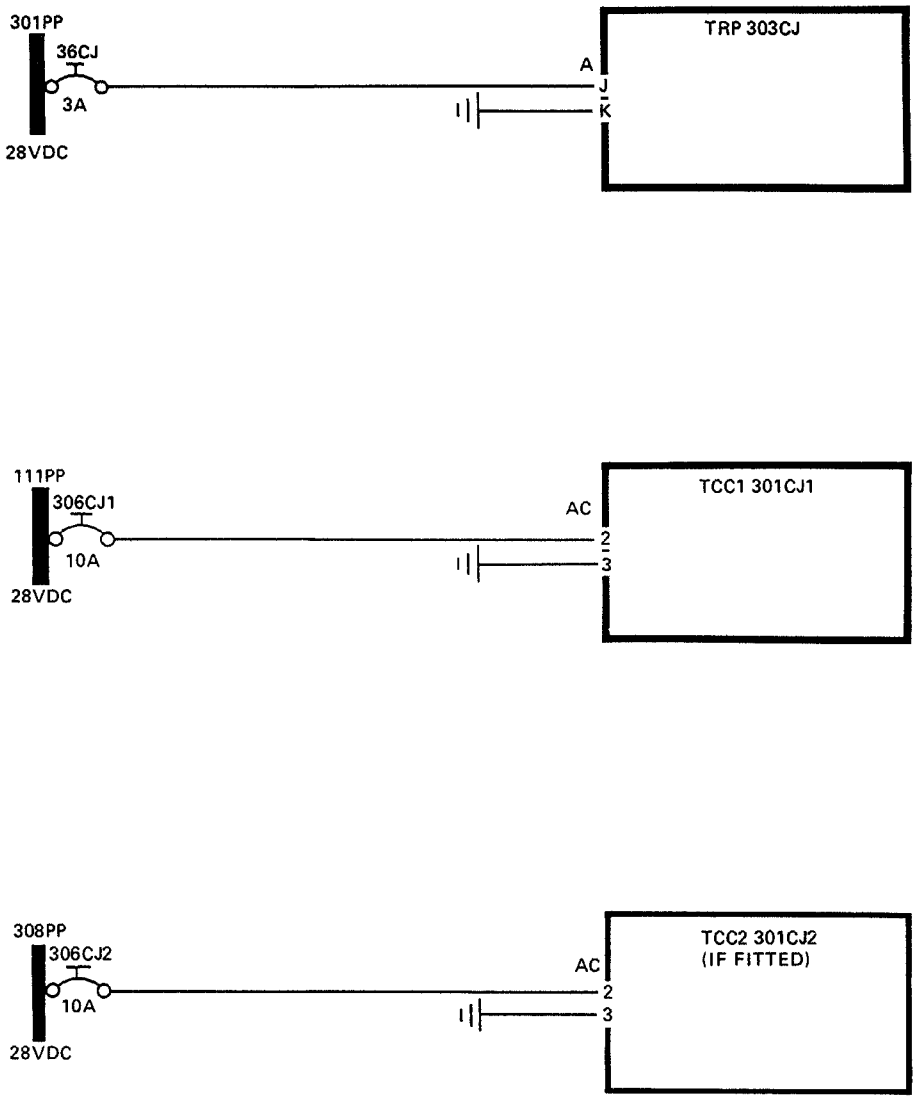
B. Power Supply Cut Off on the Ground (Ref. 22-34)

Each TCC is provided with an L/R ENGINE STOP signal which gives an indication of the engine running status.
An AND circuit is made through the low oil pressure switch of each engine.
When the AND circuit is checked, relay 164CA2 (engines stopped) sends a 28 V signal to the ground power cut off logic.
When both engines are operating contacts of relay 164CA2 are open and the TCC is faced with an open circuit.

C. Power Supply Automatic Cut Off (Ref. Fig. 005)

Internal power supply cut off is possible only if the three following conditions are fulfilled :
- aircraft on ground
- engines shut down
- no test required : lamp test
  power rise test
  test by the MTP

Power supply is cut off 12.5 s after the last one of the following conditions is fulfilled :
- aircraft on ground
- engines shut down
- end of lamp test
- end of power rise tests



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DM5

TCCS - 28VDC Electrical Power Supply
Figure 004

- end of tests by the MTP (GROUND SCAN, AFS/LAND test)
- PITCH TRIM disengagements.

D. Power Supply Restoration (Ref. Fig. 005)
Power Supply is restored :
- upon aircraft energization when safety tests running automatically are required. In that case, restoration and cut off are automatic, and power is supplied for the time that the system is under test
- when engines are started
- for maintenance purposes, by the cockpit lamp test on by the MTP for GROUND SCAN or AFS TEST/LAND TEST functions
- in operational use : through one PITCH TRIM level only ; power supply is maintained after time-delay only if one TRIM is engaged.

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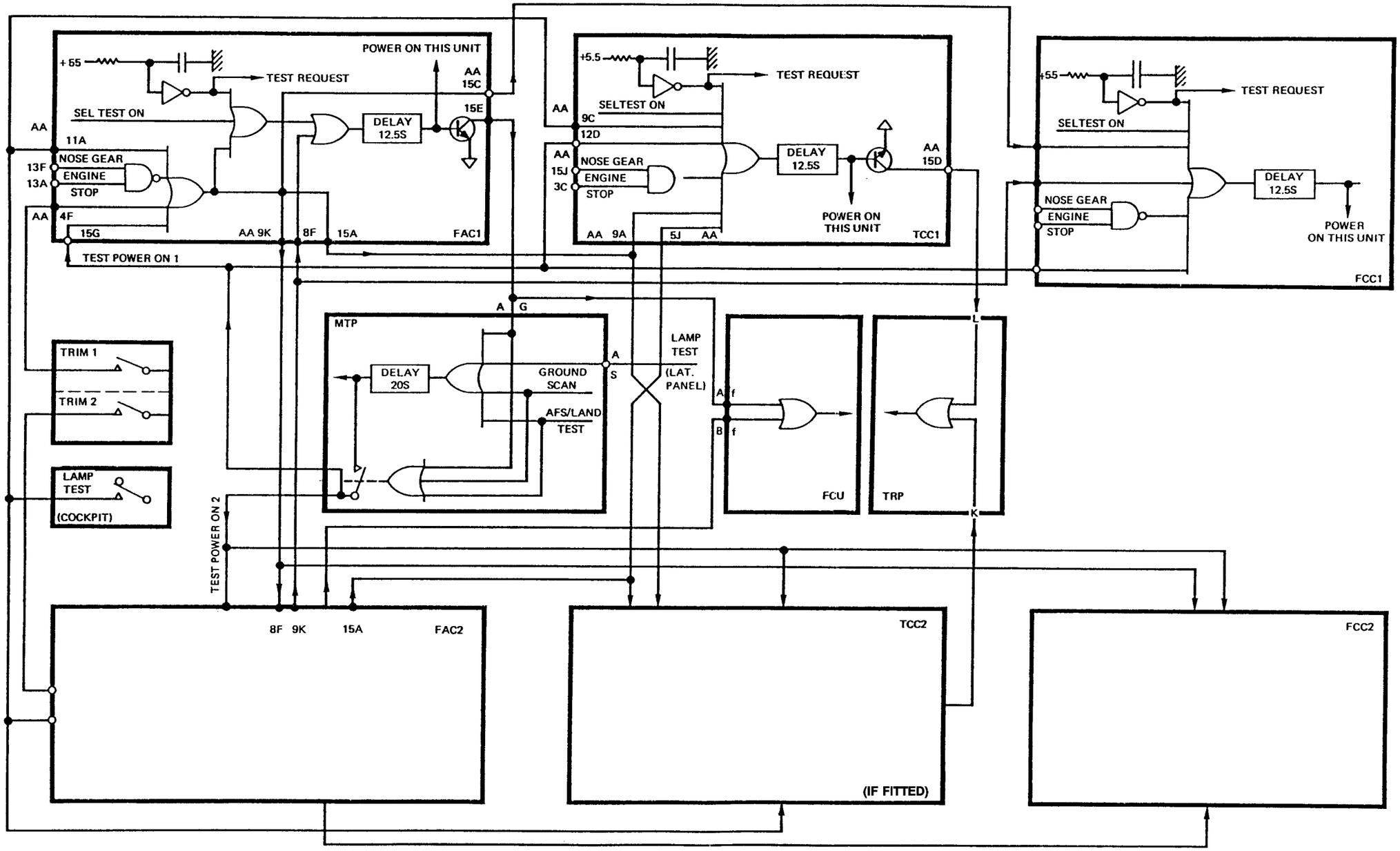

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Power Supply Cut Off on the Ground
Figure 005

1. Interface with Engine Controls

The autothrottle actuator controls the engine mechanical linkage as a function of the signals generated by TCC1 or 2.
The system mainly consists of :
- an actuator comprising a tachometer generator |
- a tee gearbox | 12CJ
- a LH coupling unit 3CJ1
- a RH coupling unit 3CJ2

A continuity circuit indicates that each component connector is correctly plugged in (ground = correct connection ; open circuit = no connection).
The lever arming logic of each TCC takes these information signals into account.
(Ref. Fig. 001)
(Ref. Fig. 002)

A. Autothrottle Actuator 12CJ - Description

The autothrottle actuator which is composed of :
- a direct-current motor
- a direct-current tachometer generator
- a three-stage speed reducer

is connected to the aircraft electrical network by a nineteen-pin connector.
The signal from TCC1 (TCC2) causes rotation of the motor and through a reduction gear of the tachometer generator providing position feedback.
Rotation is then transmitted to the tee gearbox.
Weight of the actuator is less than 2.9 kg.
Type of electrical signals :
- tachometer generator : 1.75 V/°/s
  (throttle degrees)
  range : ±12 V
- actuator :
  gradient = 0.334 °/s/V
  range = ±24 V for ±8°/s of throttle
  consumption = < 10 W in normal operation
  2A (average)
  5A (peak).

B. Tee Gearbox 12CJ - Description

On its back, the actuator is equipped with a tee gearbox which transmits the motion of the actuator to the LH and RH coupling units through torsion rods.
Transmission of the actuator torque is limited by a shear pin which breaks in case of excessive torque applied to the system.
(Ref. Fig. 003)

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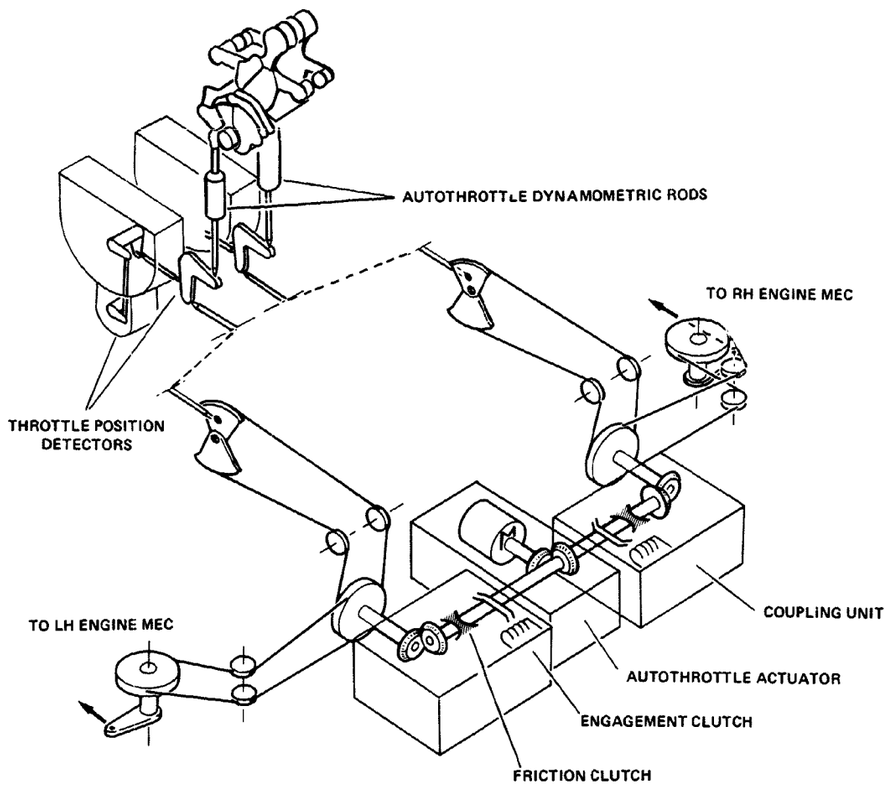

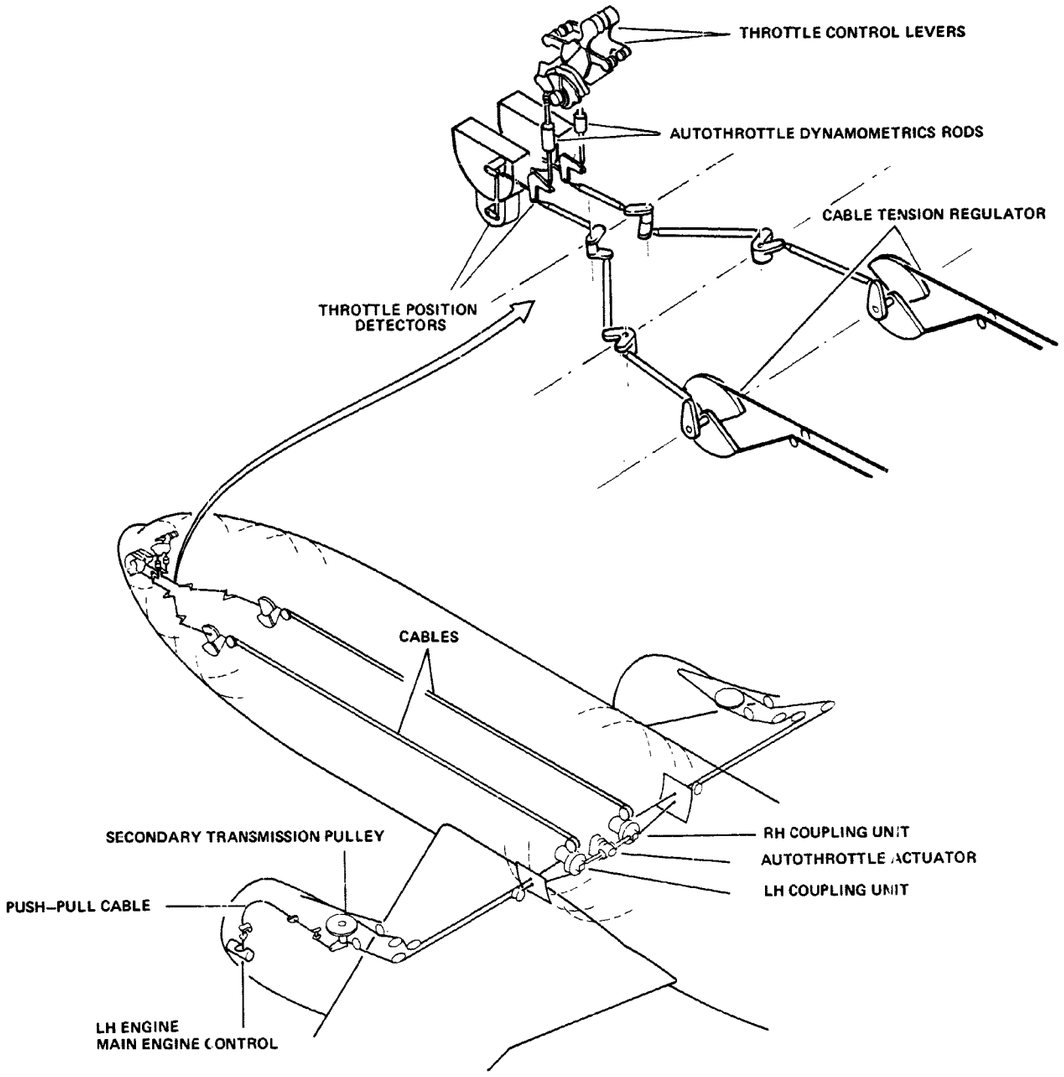

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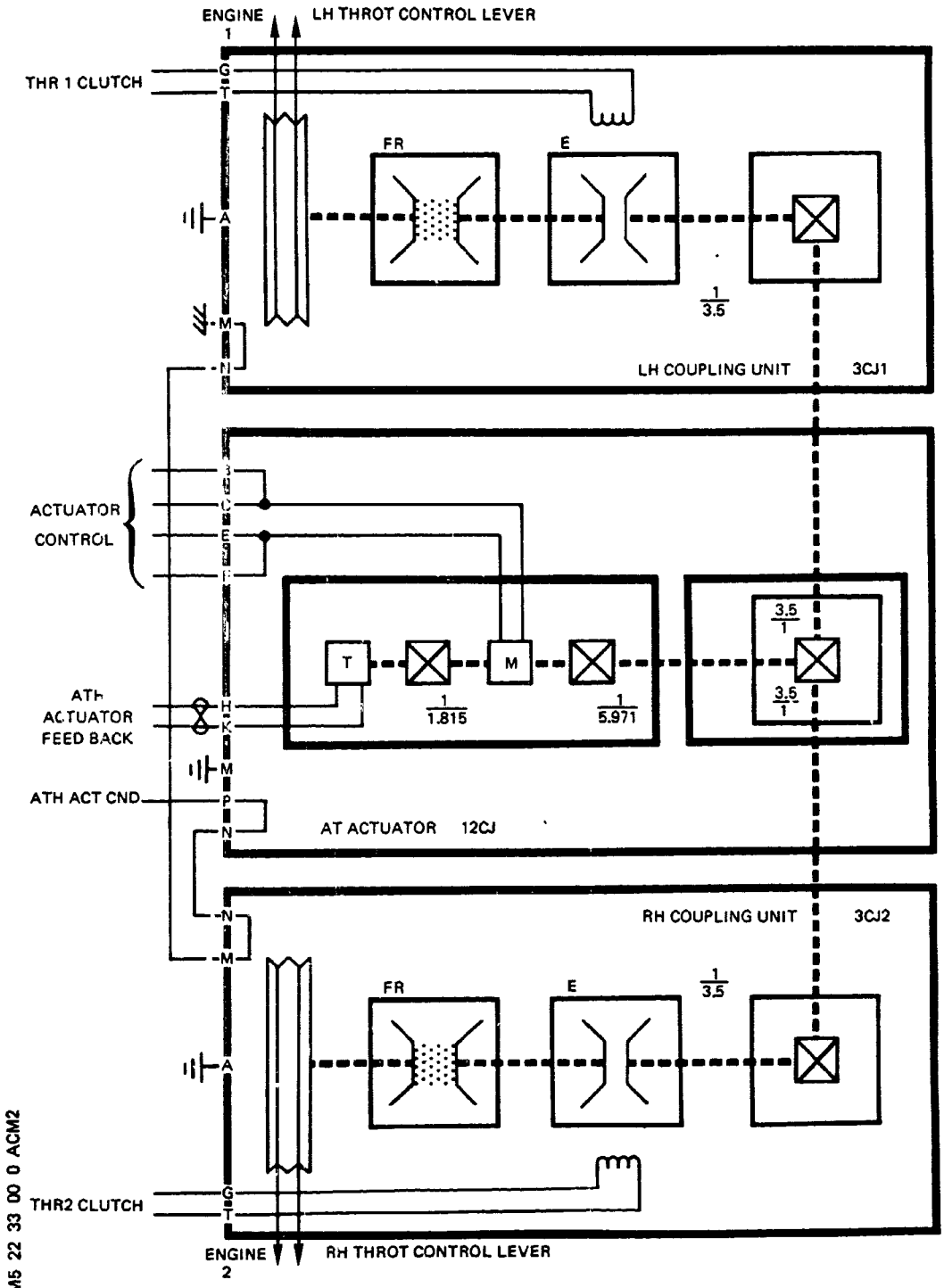
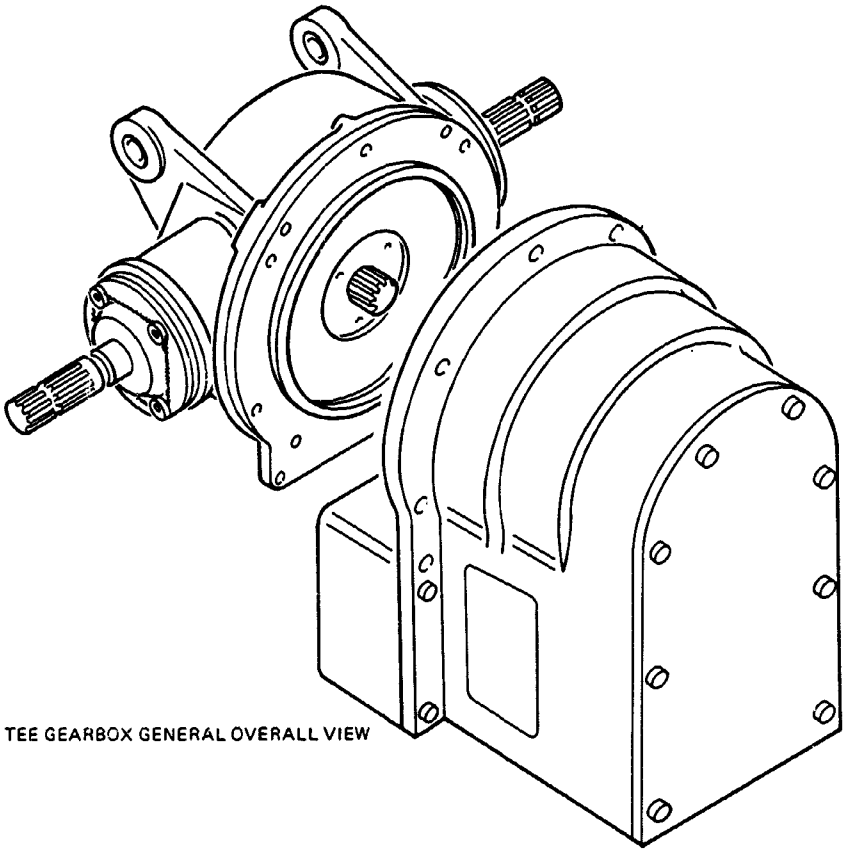
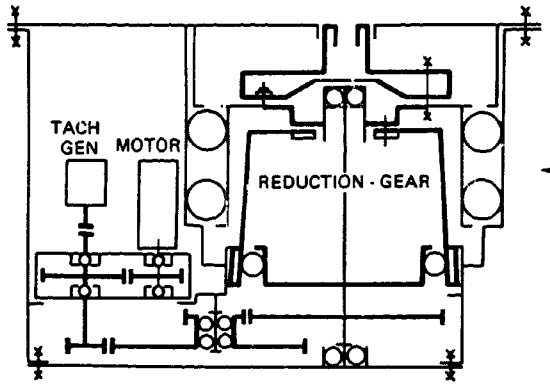
Engine Control Channel Layout
Figure 001



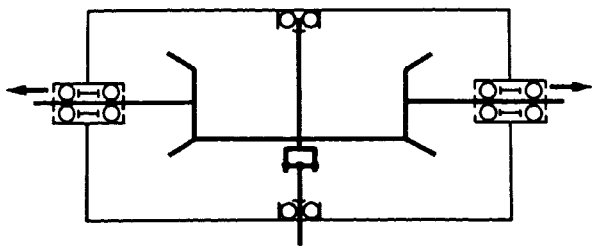
Autothrottle Actuator and Coupling Unit Power Supply
Figure 002



TEE GEARBOX GENERAL OVERALL VIEW



AT ACTUATOR SECTIONAL VIEW



TEE GEARBOX SECTIONAL VIEW

BM5 22 33 00 0 AEM2

Autothrottle Actuator - General Overall View
Figure 003

C. LH and RH Coupling Units 3CJ1 and 3CJ2 - Description

Each coupling unit is driven by the actuator through the tee gearbox and associated torsion rod.

The motion is transmitted to a bevel gear, a clutch and then a dual pulley which actuates the throttle control levers and the engine controls.

Pulley disengagement is obtained by means of the electromagnetic clutch.

After the TCCS has been armed, with a mode engaged, the electromagnetic clutch is engaged.

Declutching occurs when the pilot overrides operation of the actuator by actuating the throttle control levers.

The friction clutch acts as a torque limiter and protects the output pulley against a failure in the coupling unit.

Characteristics of each coupling unit :

Weight is less than 5.9 kg.

Power supply is +28VDC (clutch).

Power : < 21.5 W at 25.5 V.

(Ref. Fig. 004)

D. Mechanical Performance

The following performance data have been established considering the use of a DC motor with 400 g x cm minimum lock torque.

(a) Total displacement of the capstan is 116° from Full throttle to Full reverse and 71.3° from Full throttle to Full idle
The capstan "pitch" diameter is 180 mm.

(b) Output shaft speed of the coupling unit at capstan level will range between 7.7°/s and 10°/s with a maximum opposing torque of 0.77 m.daN per capstan and maximum electrical command.

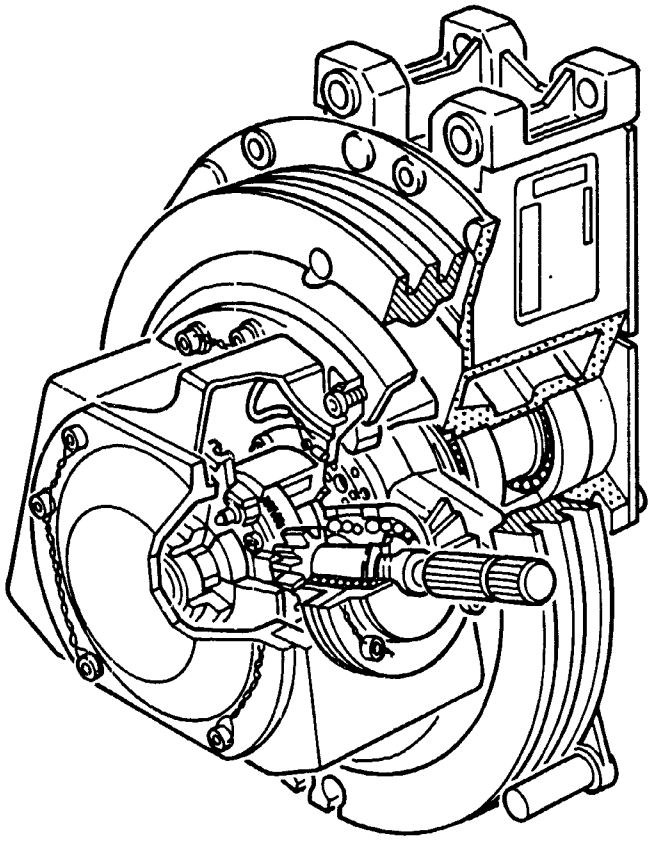
(c) The torque limiter will be set to a nominal value of 1.28 ±40% m.daN.

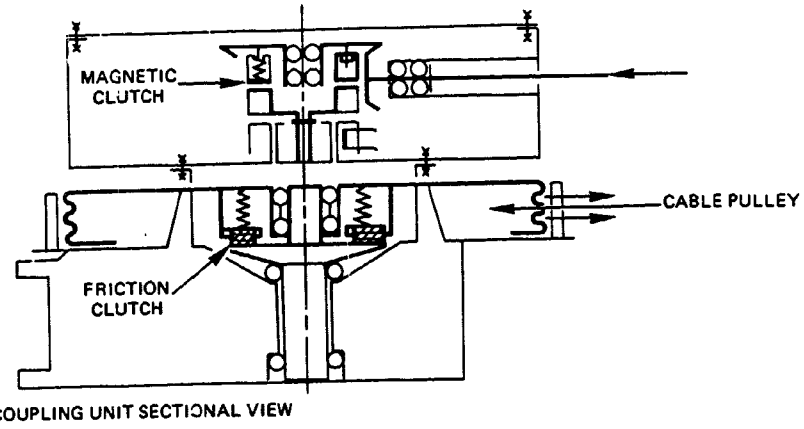
(d) The torque limiter is to be adjusted in technical laboratory.

(e) The rotational speed of motor output shafts will be approximately 3.5 that of capstan.

(f) Coupling unit residual friction at capstan (with clutch released) is lower than 0.04 m.daN.

(g) The play measured at coupling unit shaft (with capstan locked) is less than 3.5°.
(Ref. Fig. 005)

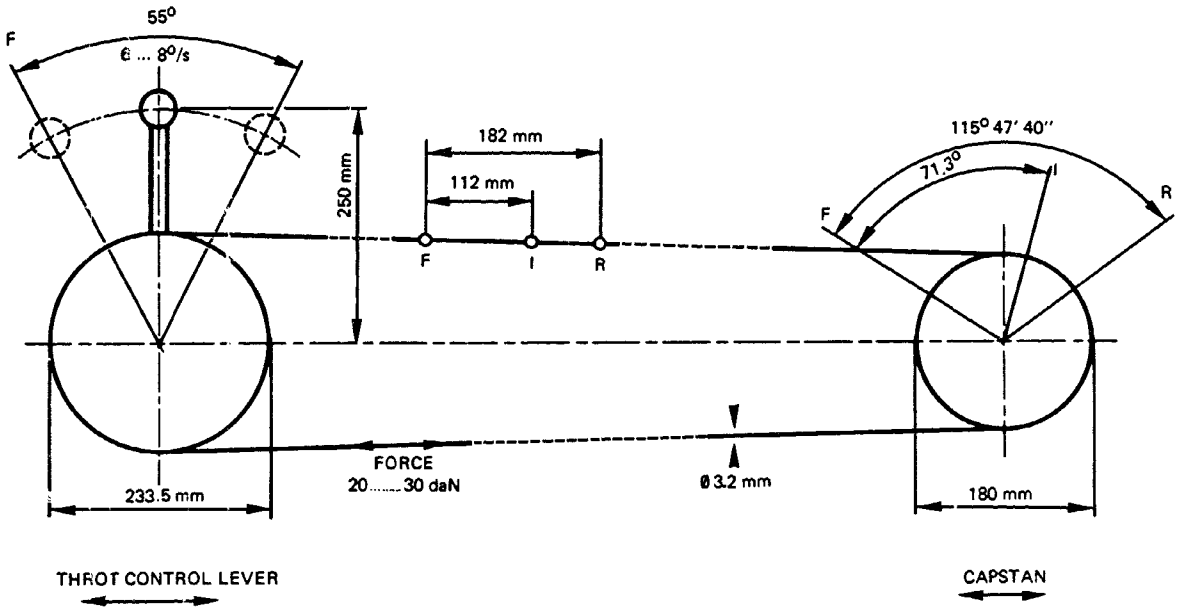




AT COUPLING UNIT SECTIONAL VIEW

BM5 22 33 00 0 AGM2

AT Coupling Unit - Cut-Away View
Figure 004



F = FULL THROTTLE
I = FULL IDLE
R = REVERSE
MN = 0.7M. daN 10% NOMINAL TORQUE

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Transmission Characteristics
Figure 005

E. Dynamometric Rods - Description

(1)Introduction

The dynamometric rods are installed in the throttle control linkage system.

Their function is to transmit electrical signals to each TCC, which vary according to the load applied on the throttle control levers by the pilot. Each rod comprises a load detector which senses any deformation of a spring.

The command thus generated causes autothrottle disconnection and enables adjustment of the lever positions without having to disengage the system.

The rods can be locked to the zero-force position to enable adjustment of the throttle control.

(2)Characteristics

Each rod weighs approximately 2.3 kg, connector and lead included.

The detector is supplied with 26 V/400 Hz.

Consumption does not exceed 0.6 VA.

Connection to the aircraft electrical network is made through a connector located at the end of a flexible lead.

A continuity circuit indicates that each rod connector is correctly plugged in (ground = correct connection ; open circuit = no connection).

The lever arming logic takes these information signals into account.

(Ref. Fig. 006)

(3)Special operational characteristics

Load detector measurement range is between -20 daN and +20 daN. Any load exceeding these limits is transmitted directly to the control linkage.

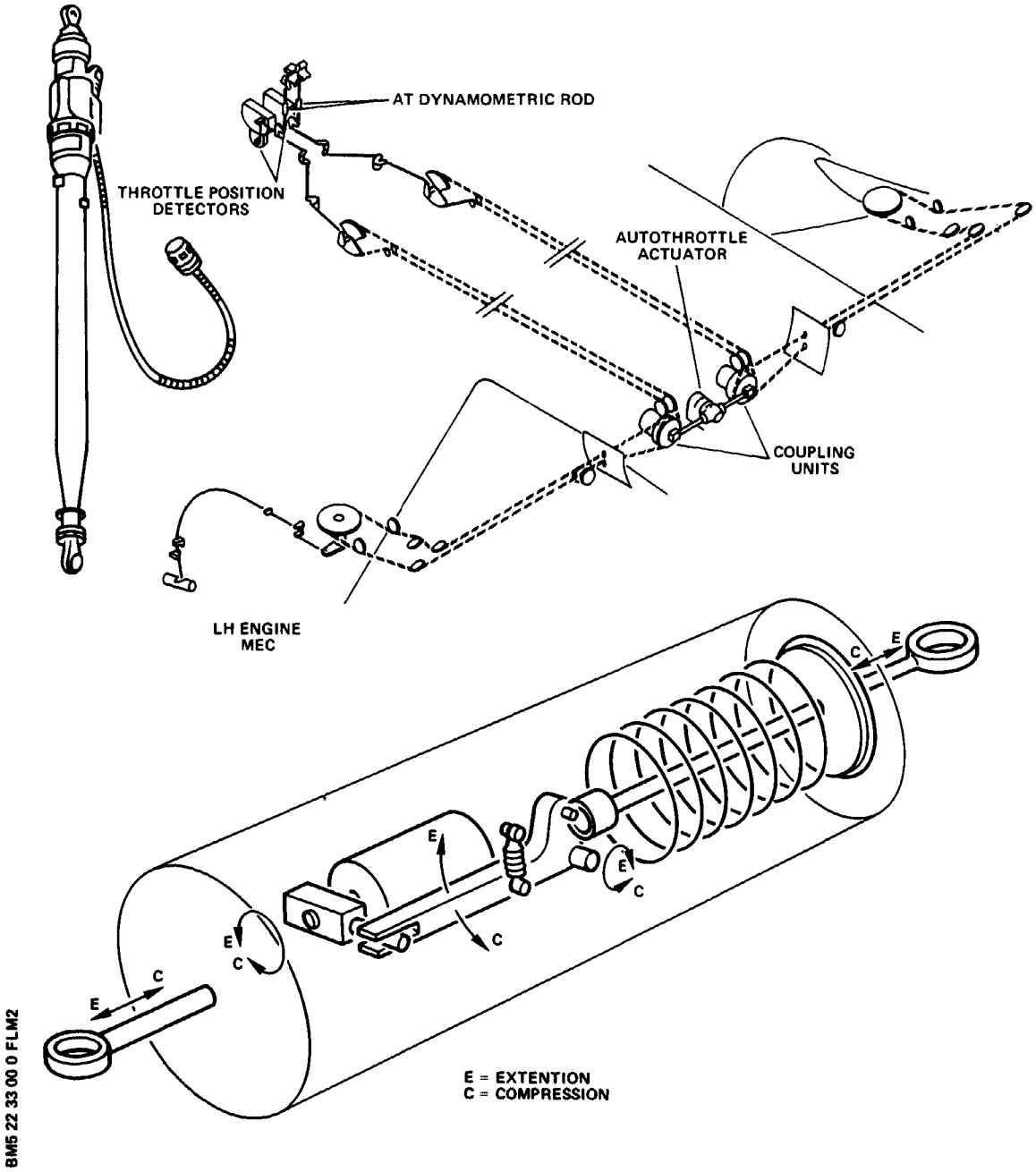
Deformation of the rod is limited to ±0.5 mm. It is sensed by two deviation detectors after mechanical amplification.

Maximum deformations of the rod correspond approximately to 3.2 V at the output of the deviation detector (±16° of the rotor rotation angle).

(4)Type of signal delivered by the load detector

Pilot load = 1.25 daN corresponds to 1° of detector rotor and to 195 mV at output.

The output signal is phase-shifted by +14.5° relative to the supply voltage.



Autothrottle Dynamometric Rods
Figure 006

(Ref. Fig. 007)

F. Throttle Position Detectors - Description

(1)General

Position of each throttle control lever is transmitted to the TCCs through a unit located under the microswitch unit.

Therefore the LH throttle control lever controls microswitch unit 112VU and LH throttle position detector 313CJ. RH throttle control lever displacement is transmitted to microswitch unit 113VU and RH throttle position detector 314CJ.

The throttle position detection system is composed of two rotary transformer resolver-type detectors

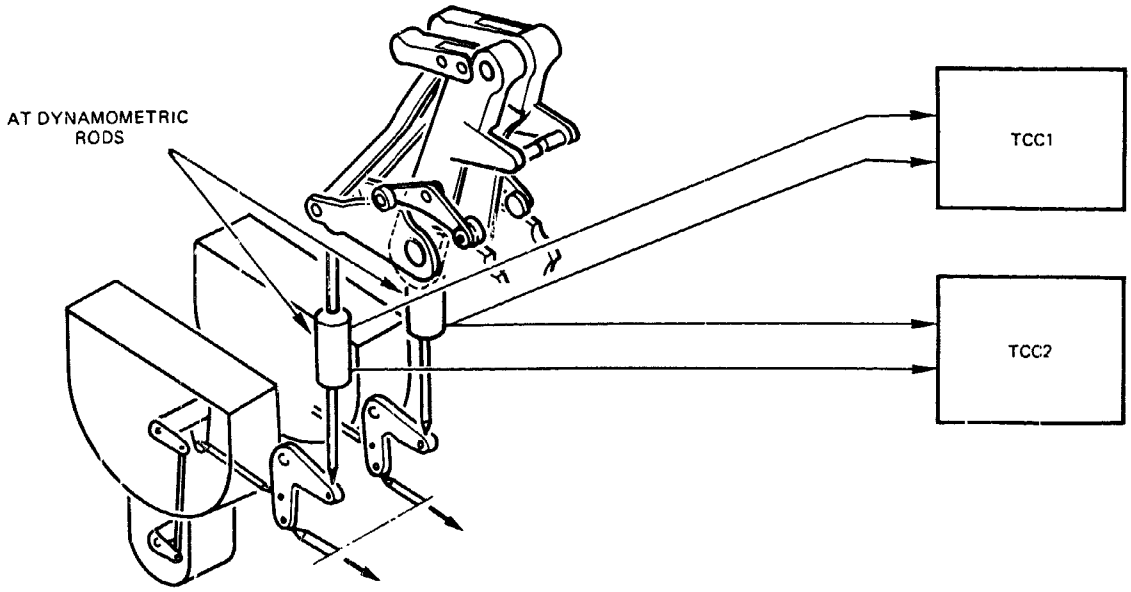
- the first detector connected directly through the input lever gives throttle control lever position to the corresponding PMC
- the second detector which receives its displacement through gears with a 1:1 ratio gives throttle control lever position to the TCCs.

(Ref. Fig. 008)

(a)Direction of rotation and setting

Direction of rotation of each detector is given in figure. Their travel is indicated in the following table.

AIRCRAFT							
LH THROTTLE CONTROL LEVER			CENTERLINE				RH THROTTLE CONTROL LEVER
FULL THROTTLE	PMC Resolver	TCC Resolver	PLA	PLA	TCC Resolver	PMC Resolver	
+55°	-84°17'	+84°17'	130°	130°	-84°17'	+84°17'	+55°
Δ = 47°17'						Δ = 47°17'	
0°	-37°	+37°	50°	50°	-37°	+37°	0°
Δ = 30°33'						Δ = 30°33'	
REVERSE	-6°27'	+6°27'	0°	0°	-6°27'	+6°27'	



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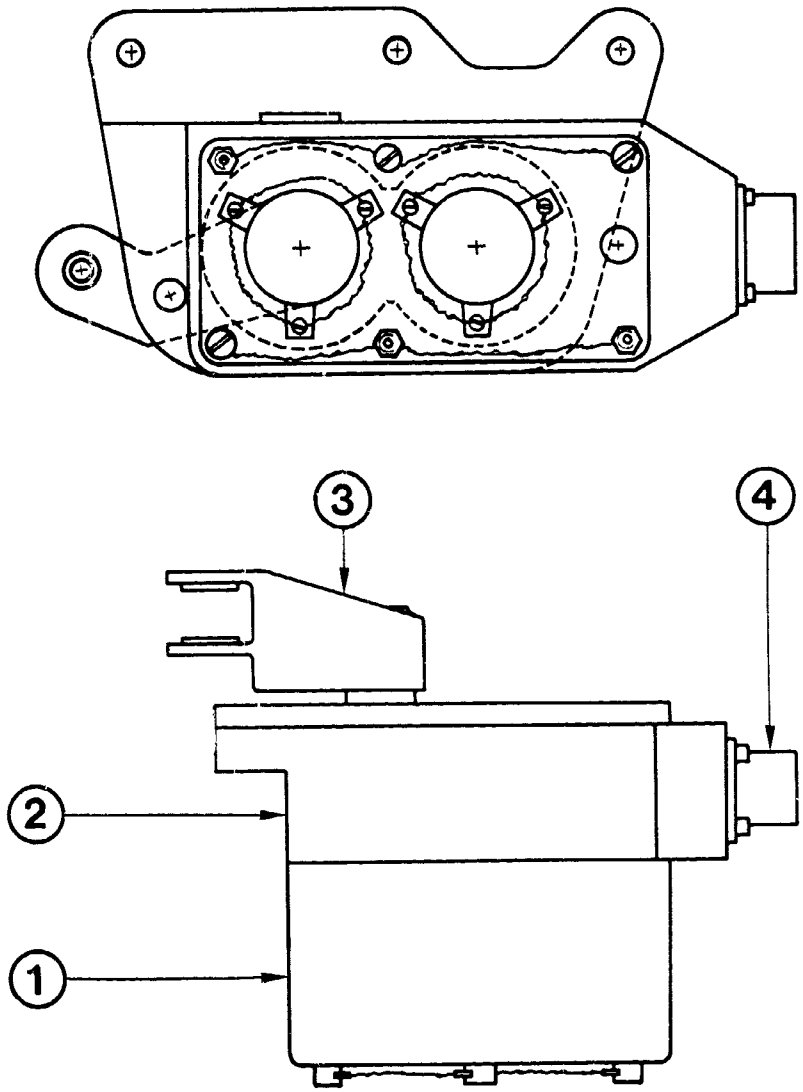
AT Dynamometric Rod Signals
Figure 007

1_COVER
2_BODY
3_INPUT LEVER
4_CONNECTOR



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LH Throttle Position Detector - Overall View
Figure 008

(b)Connector wiring
Each unit is connected to the aircraft wiring through a connector : 14-19PM for LH throttle position detector and 14-19PX for RH throttle position detector.
Connector wiring is given in figure.

(c)Power supplies
Resolvers connected to the TCCs are supplied with 26 V/400 Hz (Ref. 22-32) whereas resolvers connected to the ;PMCs are supplied directly from a special electrical system integrated in each engine (Ref. ATA 73...).

(Ref. Fig. 009)

2. Interface with Flight Controls

Each TCC receives from the FCUs two signals (lift dump 1 and 2) corresponding to extension of the spoilers in ground spoiler mode.

Each signal is a combination of the following parameters :

- wheelspeed > 85 kts
- rotation data summed with radio height condition < 5 ft.
- throttle control levers moved to the reverse position
- throttle control levers in the idle position.

These signals are fed into the logic in order to disengage the A/THR mode in the event of an acceleration stop.

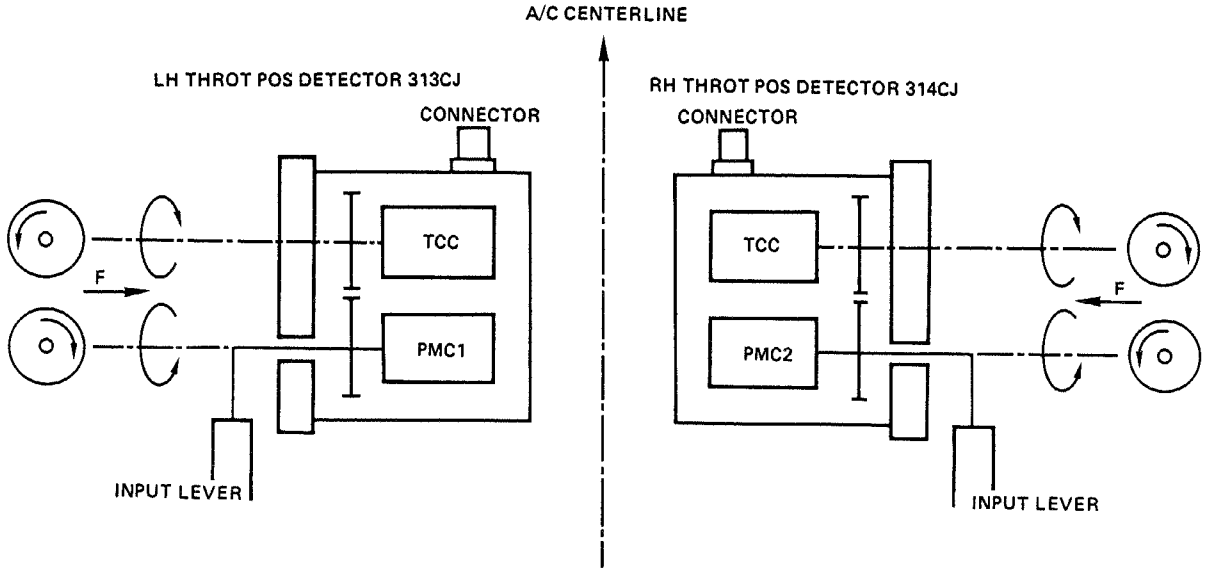
Lift dump 1 and lift dump 2 signals are only generated on the ground when spoilers 1 and 2 extend in ground spoiler mode.

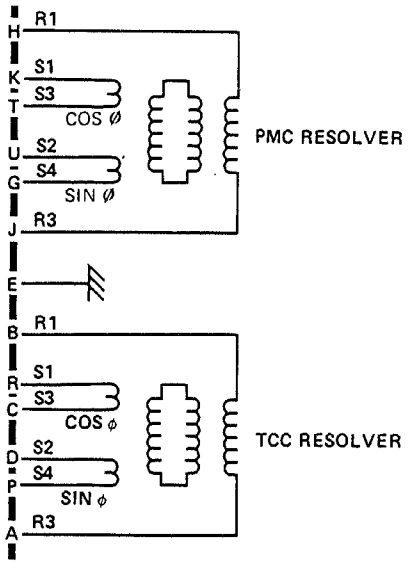
Type of signals

Spoilers extended in ground spoiler mode = ground.

Spoilers retracted = open circuit.

(Ref. Fig. 010)





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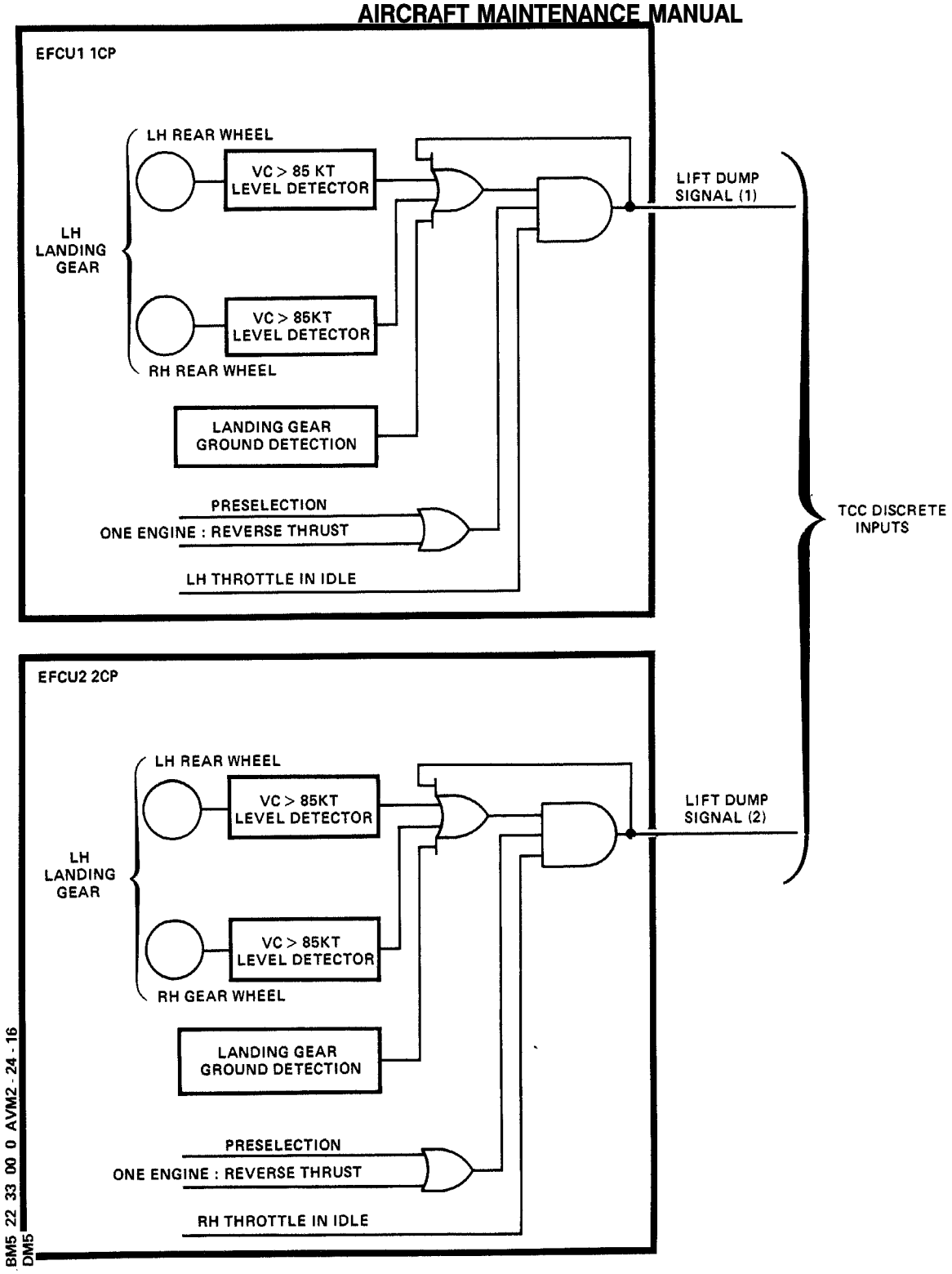
Throttle Position Detectors - Transmission Schematic
Figure 009



Interconnection between FCU and TCC
Figure 010

3. Aircraft Configuration Data

A. Flap and Slat Data

Slats ≤ 17° and flaps ≥ 20/15° data detected by the FAC are transmitted to the TCC via an ARINC data bus (ref. 22-34 FAC - TCC Arinc interface).

In the TCC each data signal (slats ≤ 17° or flaps ≥ 20/15°) is processed in the command and monitoring channels.

Slats ≤ 17° :

This position is taken into account in the AUTO mode engage logic.

Flaps ≥ 15°.

This data signal validates the engagement of GA mode and generation of the low limit logic.

B. Landing Gear Data

(1)Nose gear shock absorber compressed

Two nose gear shock absorber compressed data signals are fed to the computer from :

- relay 141GB associated with TCC2 (if fitted)
- relay 96GB associated with TCC1
  (shock absorber compressed = relay contacts closed = 28 V).

This signal is used in the TO or FLX TO mode engage logic.

(2)Main landing gears

LH and RH gear shock absorber compressed data signals are processed through an OR-gate to disengage A/THR mode as soon as the throttle control levers have reached the idle position or to validate engagement of TO or FLX TO mode.

These gear shock absorber compressed signals (28 V) are transmitted by the FAC to the TCC.

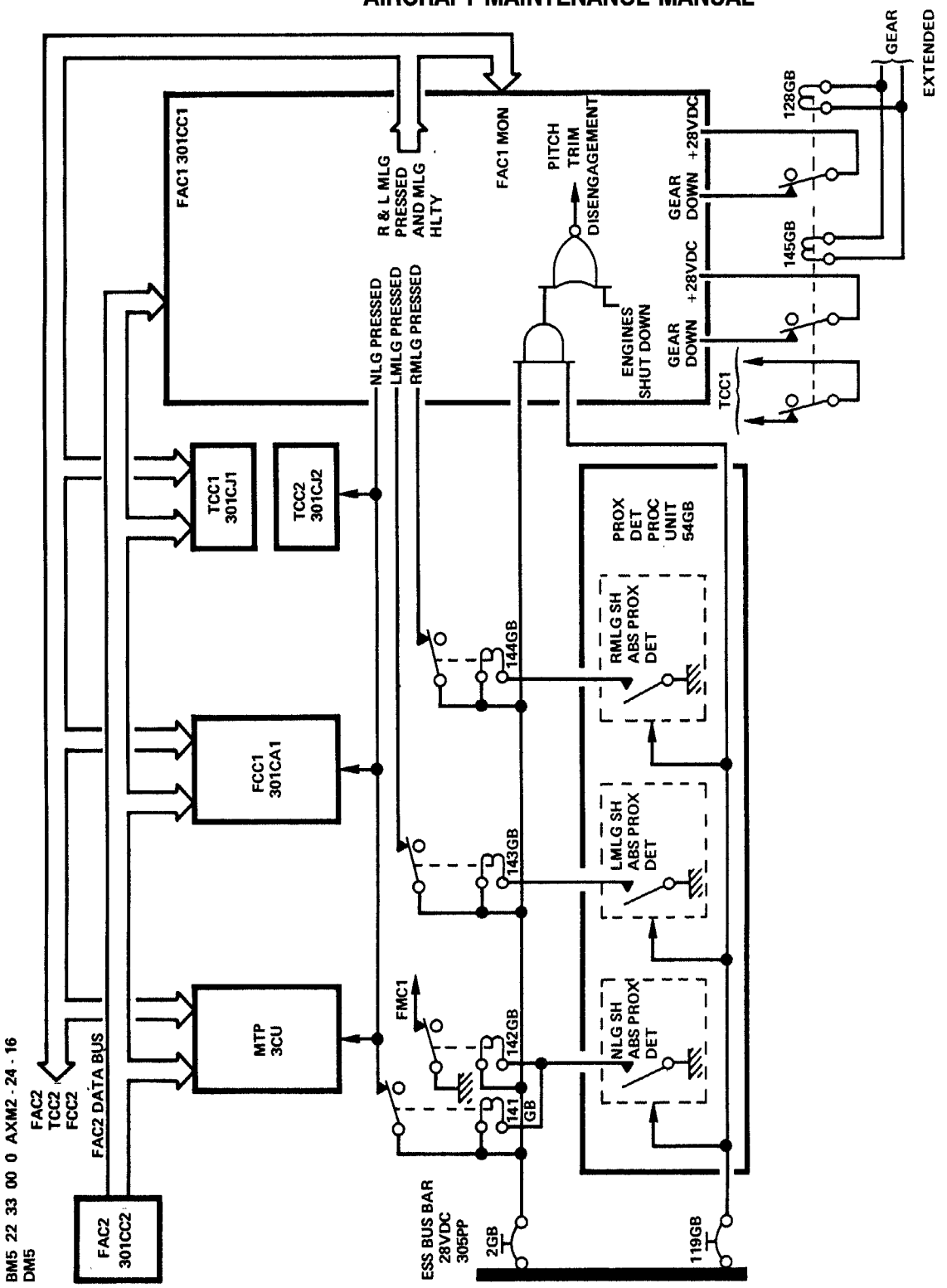
(3)Retracted gear

A gear retracted data signal is fed to the computer from relay 128GB for TCC1 and 139GB for TCC2 (if fitted). This signal is in provision and will be brought into use in computation of decelerated approaches.

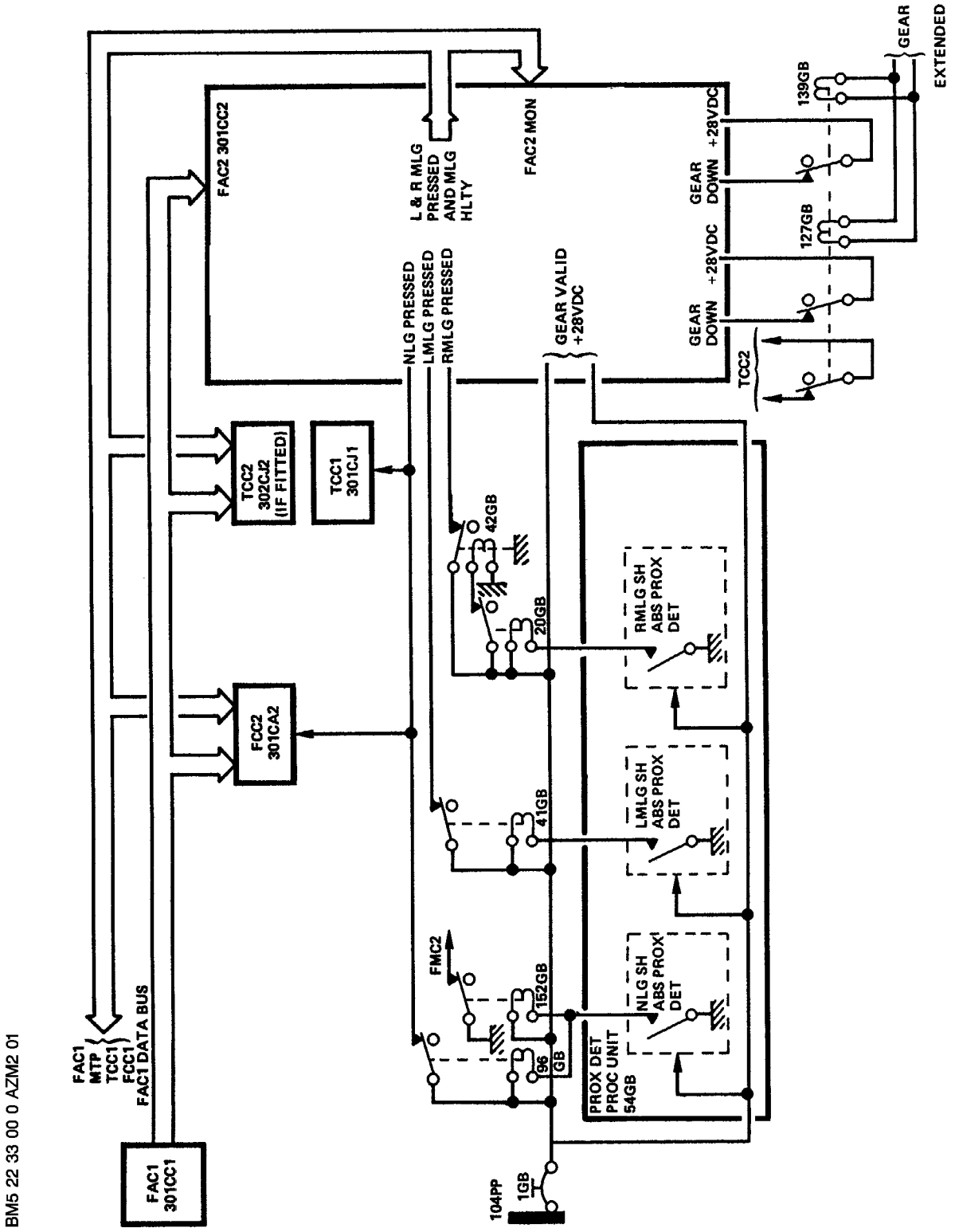
Type of signal : 28 V = downlocked gear
open circuit in all other cases.

(Ref. Fig. 011)

(Ref. Fig. 012)



Interconnection between Shock Absorber Proximity Detectors and AFS (Side 1)
Figure 011



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Interconnection between Shock Absorber Proximity Detectors and AFS (Side 2)
Figure 012

C. Air Conditioning Data - Engine and Wing Anti-Icing

For each mode, N1 lim = f(T) curves are given for the following basic air bleed conditions :

- takeoff : no air bleed for aircraft systems,
- all other modes : air is bled for normal aircraft air conditioning.

A ΔN1 correction function of altitude is introduced for any air bleed configuration different from the basic conditions mentioned above.

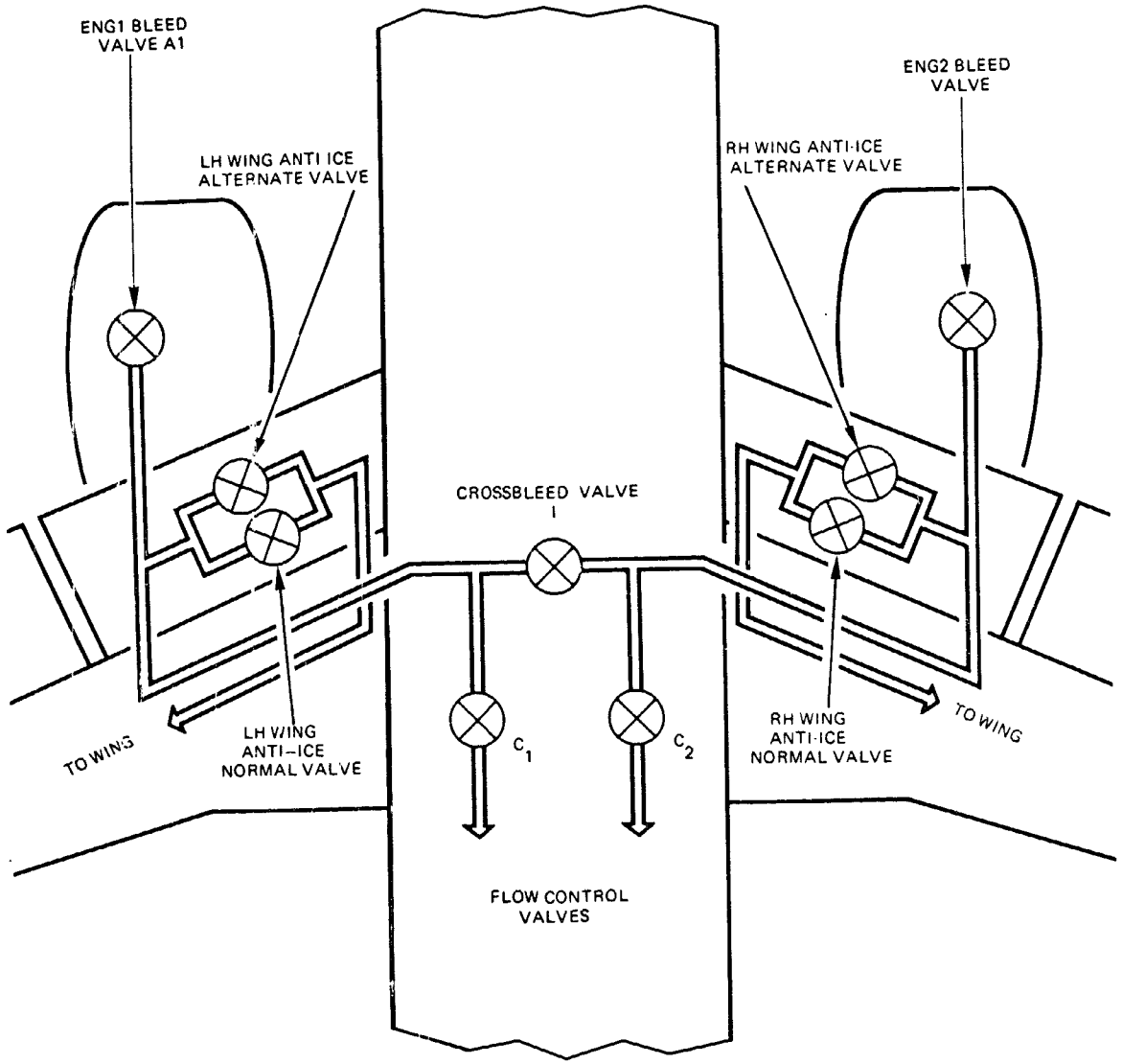
There are five air bleed configurations :

- no air bleed,
- conditioning air bleed,
- normal wing anti-icing air bleed,
- alternate wing anti-icing air bleed,
- nose cowl anti-icing air bleed.

Each one of these configurations is generated by the computer with respect to the position of the following valves :

- engine bleed valves : valve A1 for engine 1
  valve A2 for engine 2
- flow control valves : - valve C1 for pack associated with engine 1
  - valve C2 for pack associated with engine 2
- wing anti ice valves : - V1 for wing associated with engine 1
  - V2 for wing associated with engine 2
- nose cowl anti ice valves : - 1 = engine 1 nose cowl anti-icing
  - 2 = engine 2 nose cowl anti-icing
- cross feed valve 1.

The computer determines three N1 correction values via the logic circuits. (Ref. Fig. 013)



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Ice Protection and Bleed Air Supply Systems - Component Location
Figure 013

1. Interface with Engine Controls

The autothrottle actuator controls the throttle mechanical linkage as a function of the signals generated by TCC1 (or 2).
The system mainly consists of :
- an actuator comprising a tachometer generator |
- a tee gearbox | 12CJ
- a LH coupling unit 3CJ1
- a RH coupling unit 3CJ2

A continuity circuit indicates that each component connector is correctly plugged in (ground = correct connection ; open circuit = no connection).
The lever arming logic of each TCC takes these information signals into account.
(Ref. Fig. 001)
(Ref. Fig. 002)

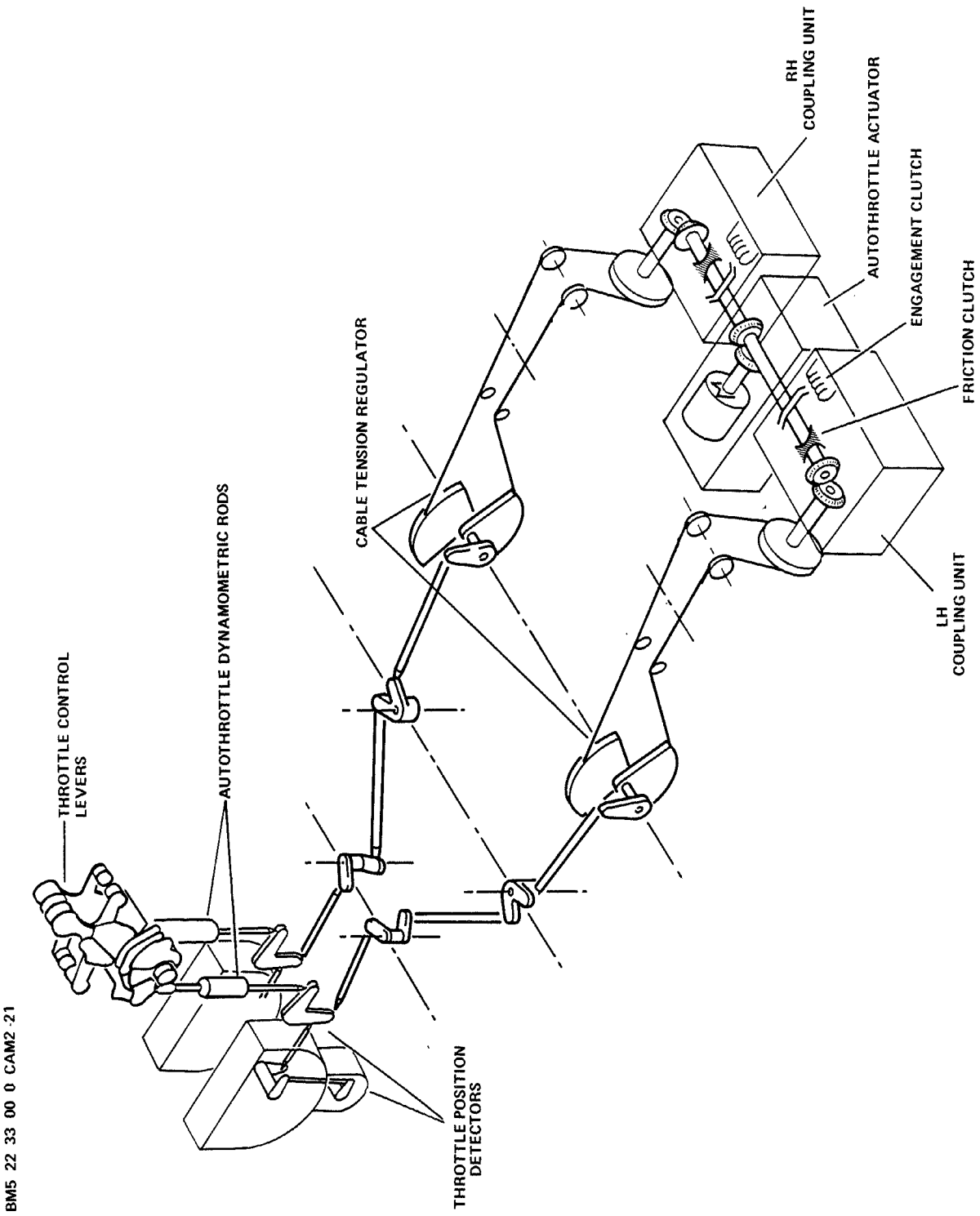
A. Autothrottle Actuator 12CJ - Description

The autothrottle actuator which is composed of :
- a direct-current motor
- a direct-current tachometer generator
- a three-stage speed reducer

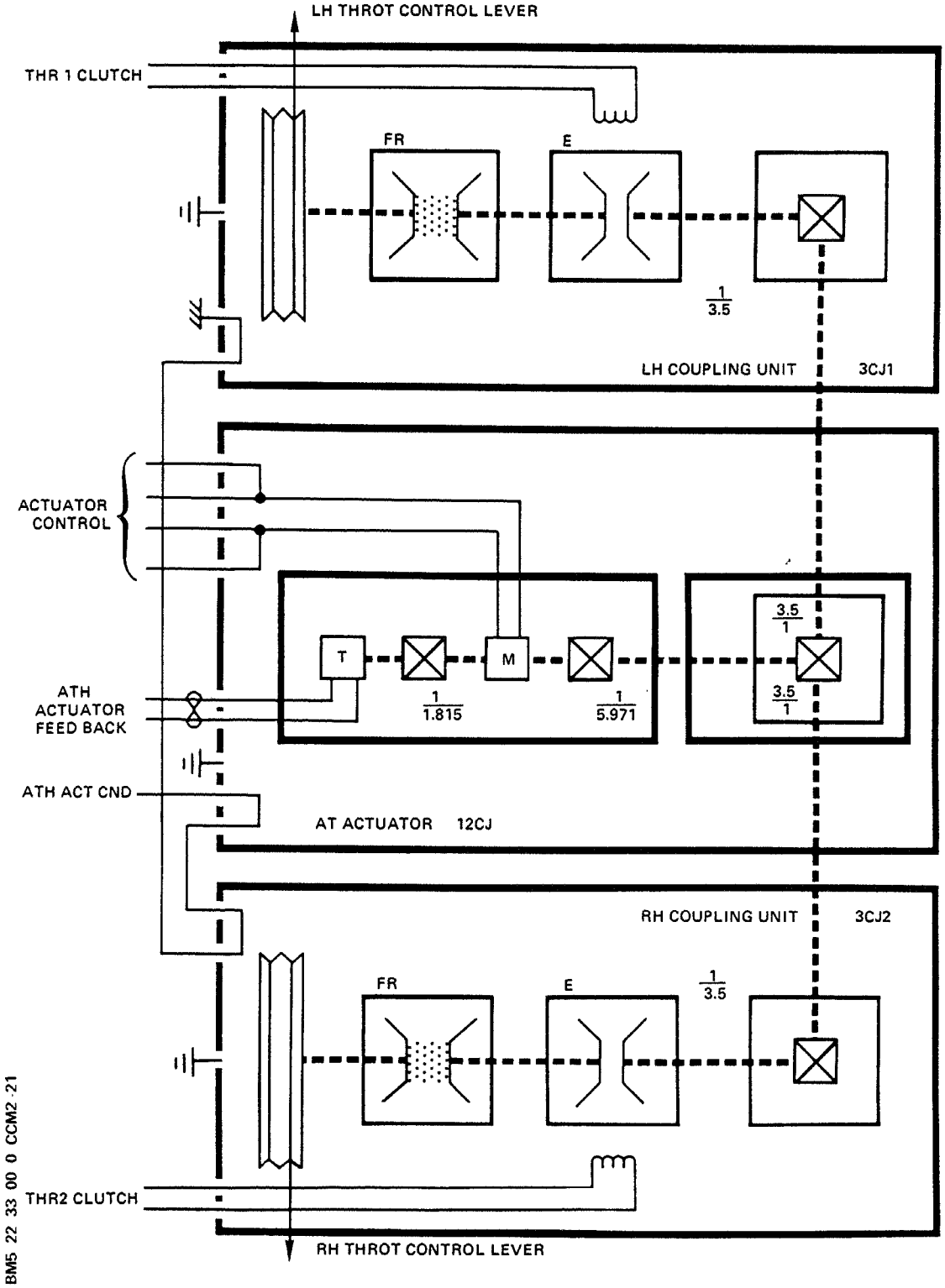
is connected to the aircraft electrical network by a nineteen-pin connector.
The signal from TCC1 (or TCC2) causes rotation of the motor and through a reduction gear of the tachometer generator providing position feedback.
Rotation is then transmitted to the tee gearbox.
Weight of the actuator is less than 2.9 kg.
Type of electrical signals :
- tachometer generator : 1.75 V/°/s
  (throttle degrees)
  range : ±12 V
- actuator :
  gradient = 0.334 °/s/V
  range = ±24 V for ±8°/s of throttle
  consumption = < 10 W in normal operation
  2A (average)
  5A (peak).

B. Tee Gearbox 12CJ - Description

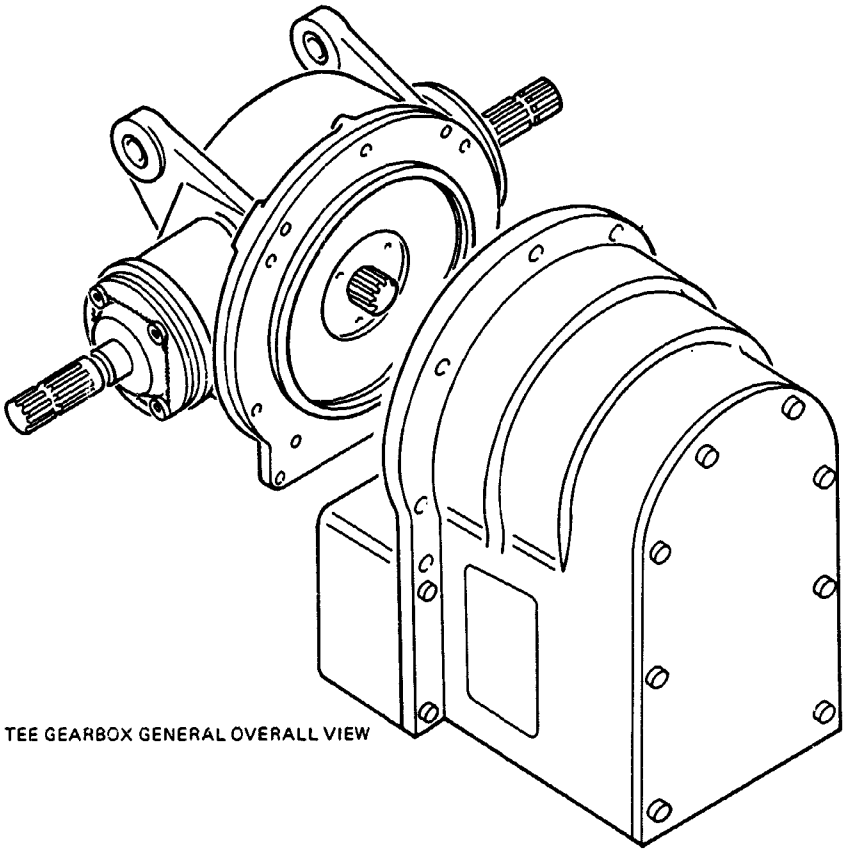
On its back, the actuator is equipped with a tee gearbox which transmits the motion of the actuator to the LH and RH coupling units through torsion rods.
Transmission of the actuator torque is limited by a shear pin which breaks in case of excessive torque applied to the system.
(Ref. Fig. 003)



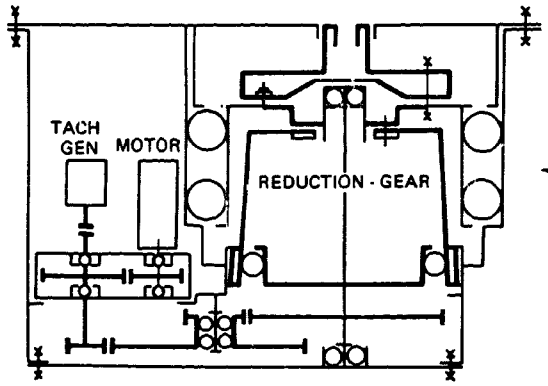
Engine Control Channel Layout
Figure 001



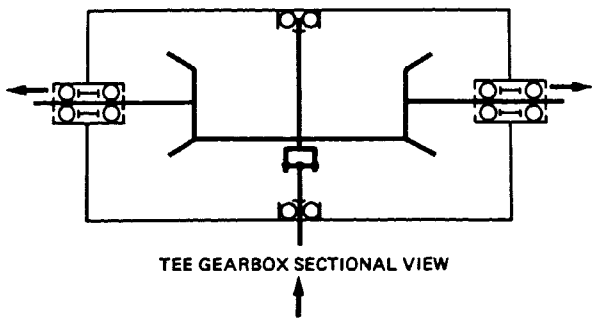
Autothrottle Actuator and Coupling Unit Power Supply
Figure 002



TEE GEARBOX GENERAL OVERALL VIEW



AT ACTUATOR SECTIONAL VIEW



TEE GEARBOX SECTIONAL VIEW

BM5 22 33 00 0 AEM2

Autothrottle Actuator - General Overall View
Figure 003

C. LH and RH Coupling Units 3CJ1 and 3CJ2 - Description

Each coupling unit is driven by the actuator through the tee gearbox and associated torsion rod.

The motion is transmitted to a level gear, a clutch and then a pulley which actuates the throttle control levers.

Pulley disengagement is obtained by means of the electromagnetic clutch.

After the TCCS has been armed, with a mode engaged, the electromagnetic clutch is engaged.

Declutching occurs when the pilot overrides operation of the actuator by actuating the throttle control levers.

The friction clutch acts as a torque limiter and protects the output pulley against a failure in the coupling unit.

Characteristics of each coupling unit :

Power supply is +28VDC (clutch).

Power : < 21.5 W at 25.5 V.

(Ref. Fig. 004)

D. Mechanical Performance

The following performance data have been established considering the use of a DC motor with 400 g x cm minimum lock torque.

(a) Total displacement of the capstan is 116° from Full throttle to Full reverse and 71.3° from Full throttle to Full idle
The capstan "pitch" diameter is 180 mm.

(b) Output shaft speed of the coupling unit at capstan level will range between 7.7°/s and 10°/s with a maximum opposing torque of 0.77 m.daN per capstan and maximum electrical command.

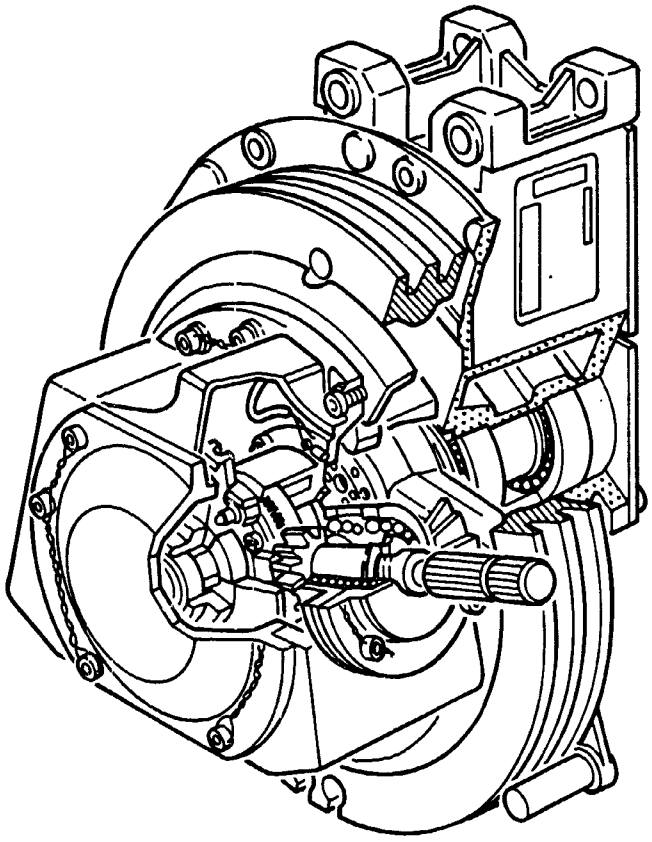
(c) The torque limiter will be set to a nominal value of 1.28 ±40% m.daN.

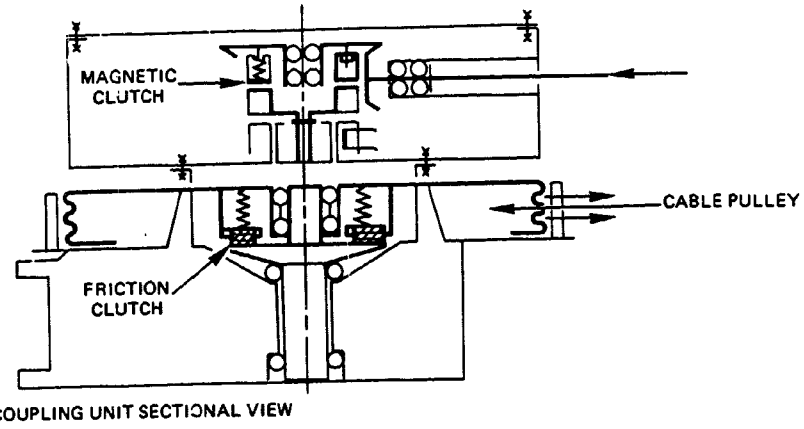
(d) The torque limiter is to be adjusted in technical laboratory.

(e) The rotational speed of motor output shafts will be approximately 3.5 that of capstan.

(f) Coupling unit residual friction at capstan (with clutch released) is lower than 0.04 m.daN.

(g) The play measured at coupling unit shaft (with capstan locked) is less than 3.5°.
(Ref. Fig. 005)

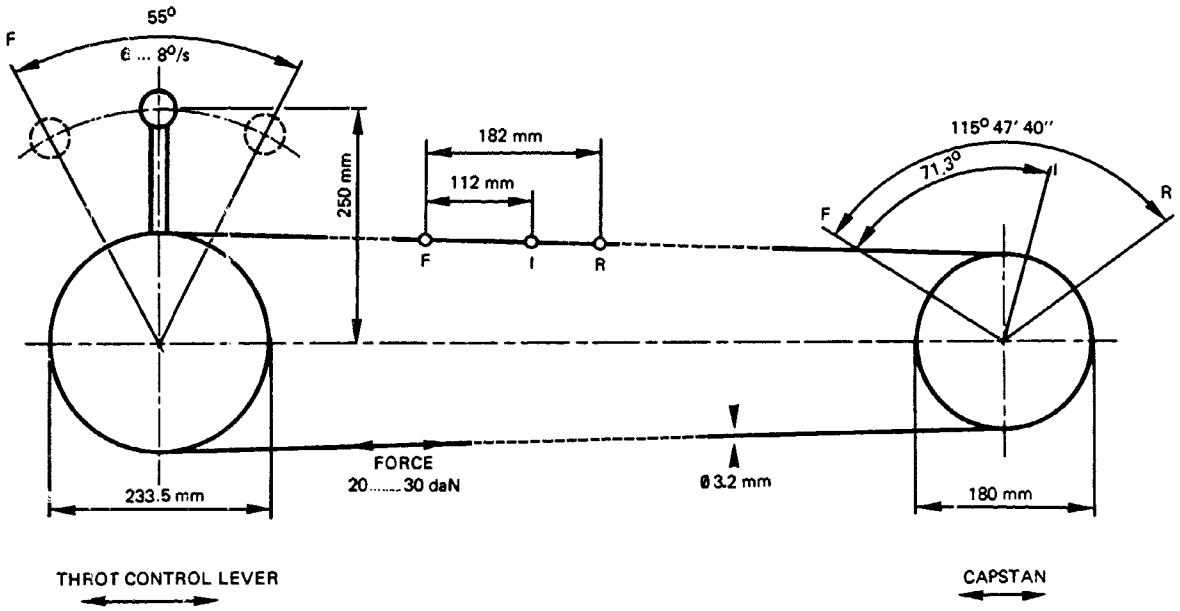




AT COUPLING UNIT SECTIONAL VIEW

BM5 22 33 00 0 AGM2

AT Coupling Unit - Cut-Away View
Figure 004



F = FULL THROTTLE
I = FULL IDLE
R = REVERSE
MN = 0.7M. daN 10% NOMINAL TORQUE

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Transmission Characteristics
Figure 005

E. Dynamometric Rods - Description

(1)Introduction

The dynamometric rods are installed in the throttle control linkage system. Their function is to transmit electrical signals to each TCC, which vary according to the load applied on the throttle control levers by the pilot. Each rod comprises a load detector which senses any deformation of a spring.

The command thus generated causes autothrottle declutch and enables adjustment of the lever positions without having to disengage the system.

The rods can be locked to the zero-force position to enable adjustment of the throttle control.

(2)Characteristics

Each rod weighs approximately 2.3 kg, connector and lead included.

The detector is supplied with 26 V/400 Hz.

Consumption does not exceed 0.6 VA.

Connection to the aircraft electrical network is made through a connector located at the end of a flexible lead.

A continuity circuit indicates that each rod connector is correctly plugged in (ground = correct connection ; open circuit = no connection).

The lever arming logic takes these information signals into account.

(3)Special operational characteristics

Load detector measurement range is between -20 daN and +20 daN. Any load exceeding these limits is transmitted directly to the control linkage.

Deformation of the rod is limited to ±0.5 mm. It is sensed by two deviation detectors after mechanical amplification.

Maximum deformations of the rod correspond approximately to 3.2 V at the output of the deviation detector (±16° of the rotor rotation angle).

(4)Type of signal delivered by the load detector

Pilot load = 1.25 daN corresponds to 1° of detector rotor and to 195 mV at output.

The output signal is phase-shifted by +14.5° relative to the supply voltage.

(Ref. Fig. 006)

F. Throttle Position Detectors - Description

(1)General

Position of each throttle control lever is transmitted to the TCCs and EEC through a unit located under the microswitch unit.
Therefore the LH throttle control lever controls microswitch unit 112VU and LH throttle position detector 313CJ. RH throttle control lever displacement is transmitted to microswitch unit 113VU and RH throttle position detector 314CJ.
The throttle position detection system is composed of three rotary transformer resolver-type detectors mounted on the same shaft.
Per throttle, one resolver sends a signal to the TCC, the two other resolvers send a signal to their respective EEC (Channels A and B).
The first and the third resolvers are supplied by the corresponding EEC and transmit the throttle control lever positions to it.
The second resolver, supplied in 26V/400Hz and mounted between the two others linked with the EEC, transmits the throttle control lever positions to the TCC.

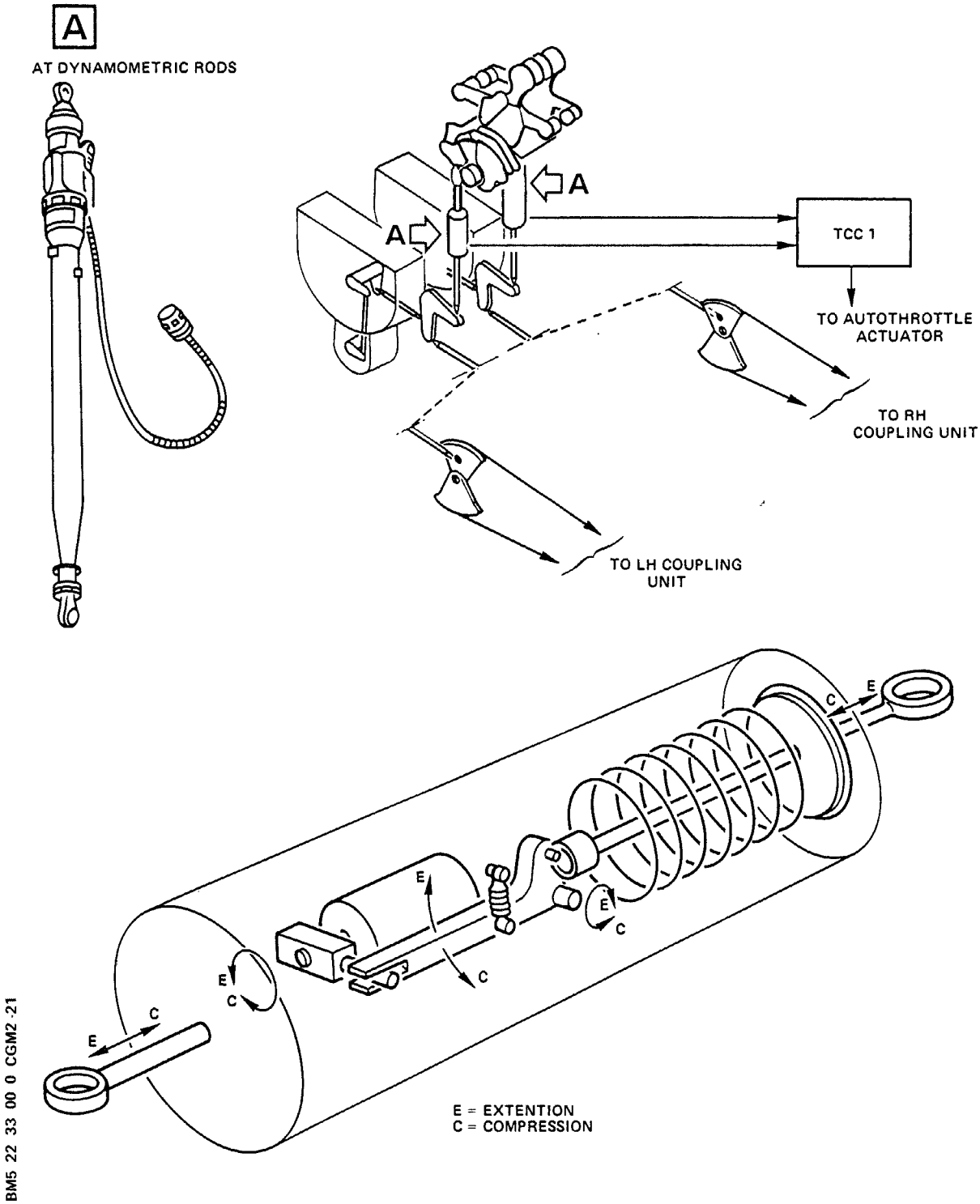
(a)Direction of rotation and setting

Direction of rotation of each detector is given in figure.
Their travel is indicated in the following table.

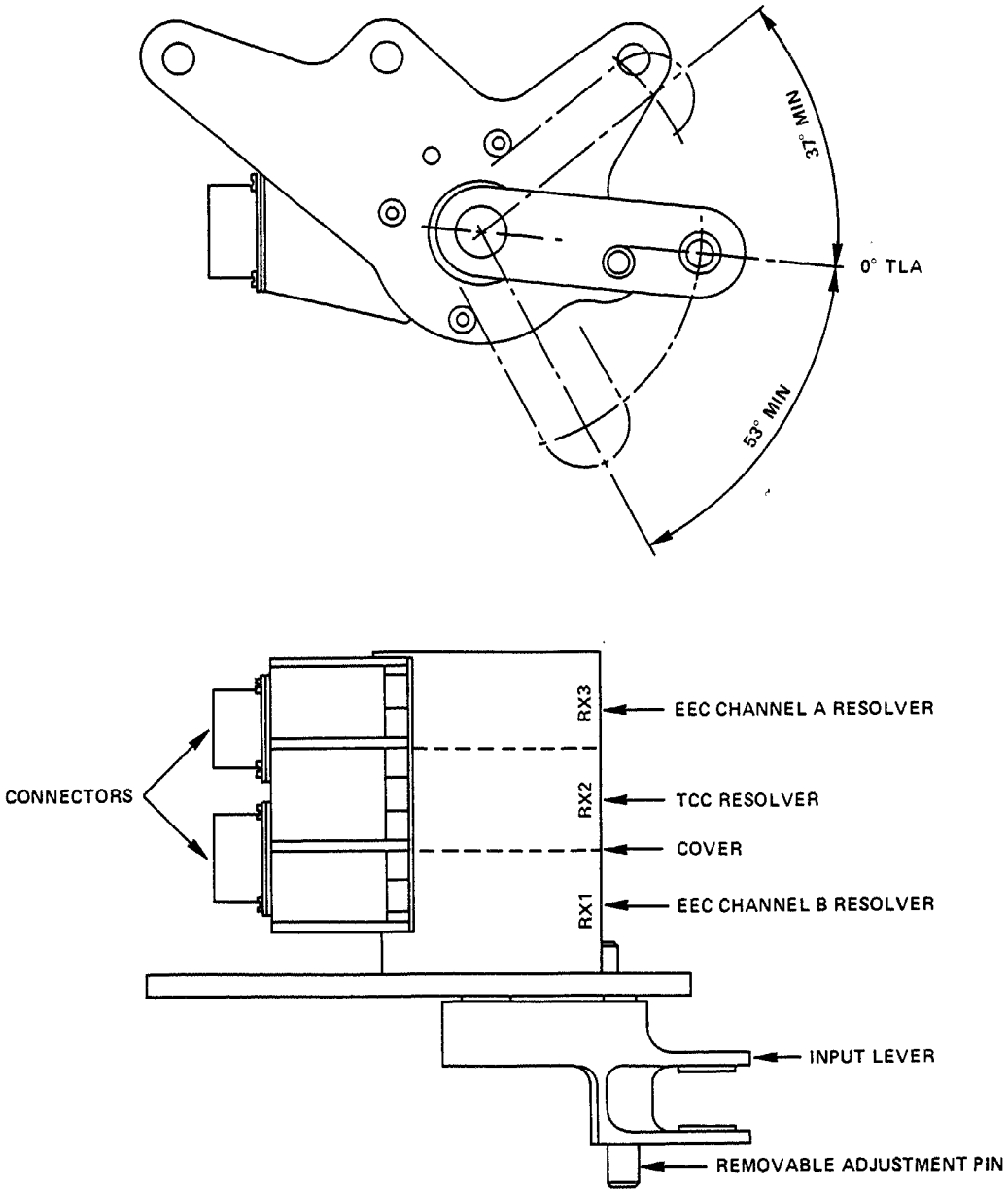
AIRCRAFT							
LH THROTTLE CONTROL LEVER			CENTERLINE				RH THROTTLE CONTROL LEVER
FULL THROTTLE	EEC Resolver	TCC Resolver	PLA	PLA	TCC Resolver	ECC Resolver	
+55°	-84°17'	+84°17'	129°45'	129°45'	-84°17'	+84°17'	+55°
Δ = 47°17'							Δ = 47°17'
0°	-37°	+37°	53°	53°	-37°	+37°	0°
Δ = 30°33'							Δ = 30°33'
REVERSE(119°40)	-6°27'	+6°27'	0°		-6°27'	+6°27'	

AIRCRAFT							
LH THROTTLE CONTROL LEVER			CENTERLINE				RH THROTTLE CONTROL LEVER
FULL THROTTLE	EEC Resolver	TCC Resolver	PLA	PLA	TCC Resolver	ECC Resolver	

(Ref. Fig. 007)
(Ref. Fig. 008)

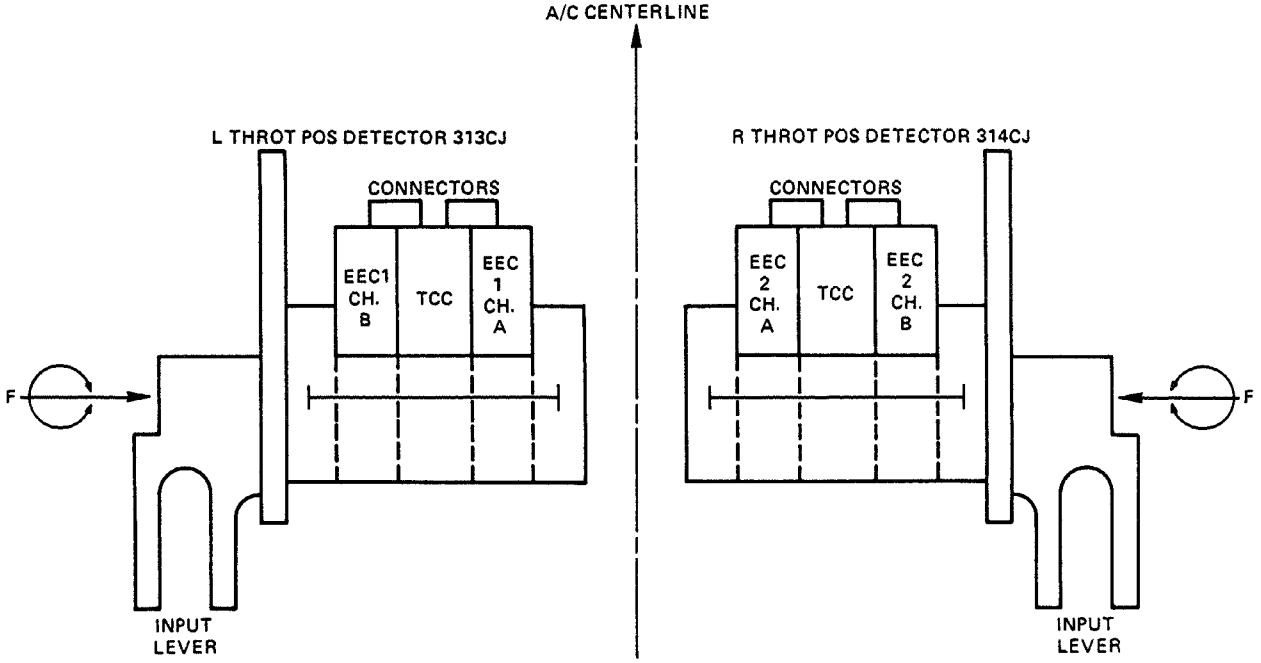


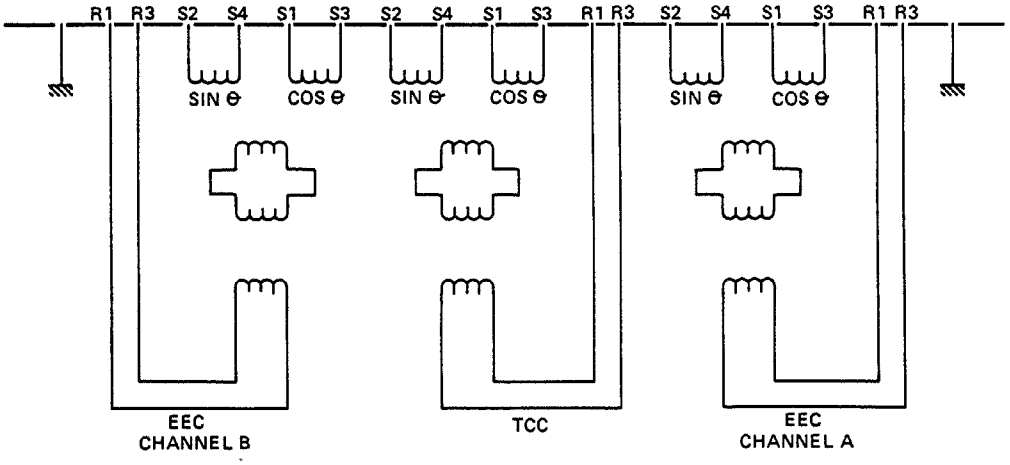
Autothrottle Dynamometric Rods
Figure 006



BM5 22 33 00 0 CJM2 21

Throttle Position Detector - Overall View
Figure 007





$$\sqcup \overrightarrow{S1S3} = \sqcup \overrightarrow{R1R3} \cos\theta$$
$$\sqcup \overrightarrow{S2S4} = \sqcup \overrightarrow{R1R3} \sin\theta$$

BM5 22 33 00 0 CLM2-21

Throttle Position Detectors - Schematic
Figure 008

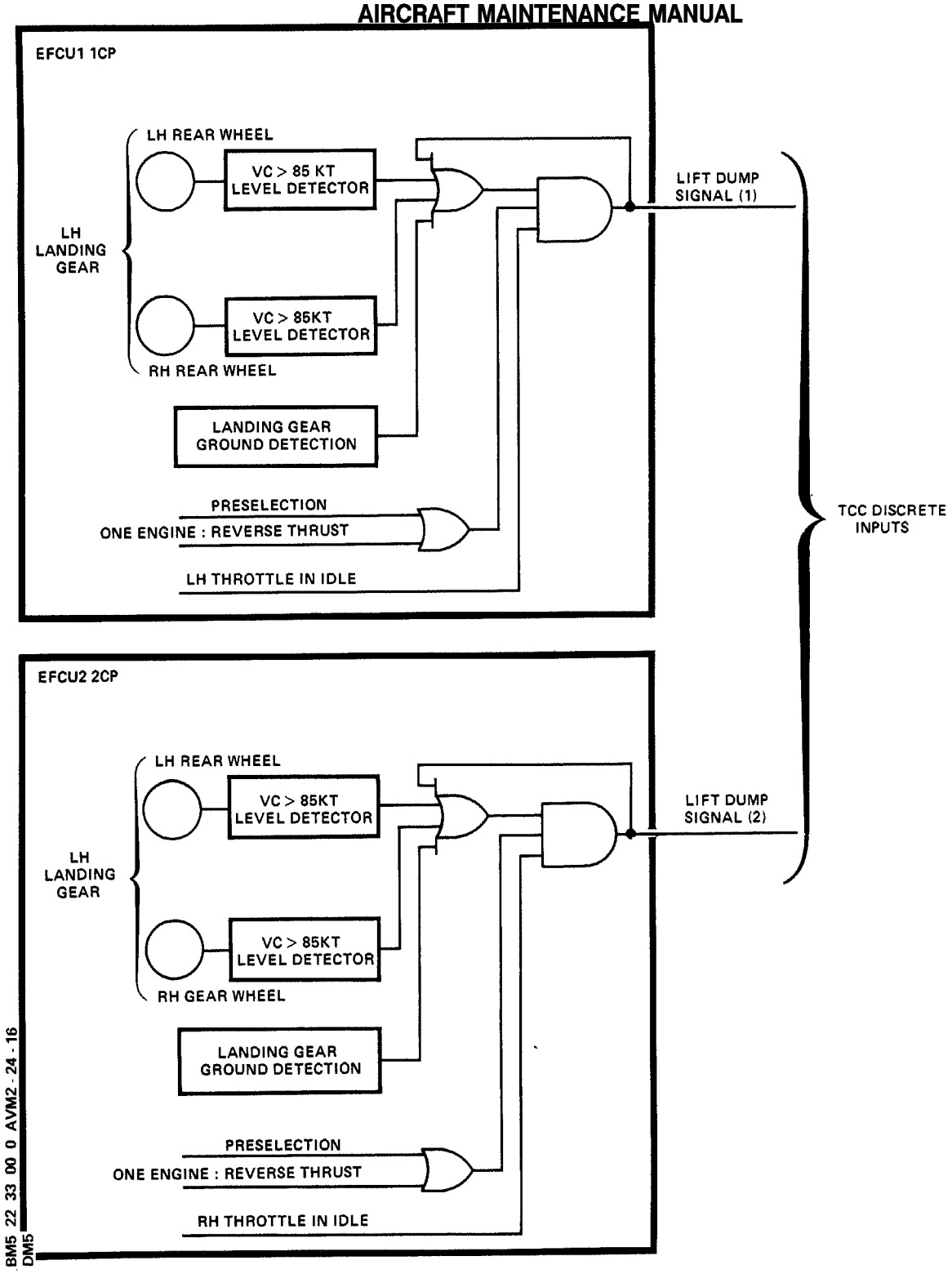
2. Interface with Flight Controls

Each TCC receives from the EFCUs two signals (lift dump 1 and 2) corresponding to extension of the spoilers in ground spoiler mode.
Each signal is a combination of the following parameters :
- wheelspeed > 85 kts
- rotation data summed with radio height condition < 5 ft.
- throttle control levers moved to the reverse position
- throttle control levers in the idle position.

These signals are fed into the logic in order to disengage the A/THR mode in the event of an acceleration stop.
Lift dump 1 and lift dump 2 signals are only generated on the ground when spoilers 1 and 2 extend in ground spoiler mode.

Type of signals

Spoilers extended in ground spoiler mode = ground.
Spoilers retracted = open circuit.
(Ref. Fig. 009)

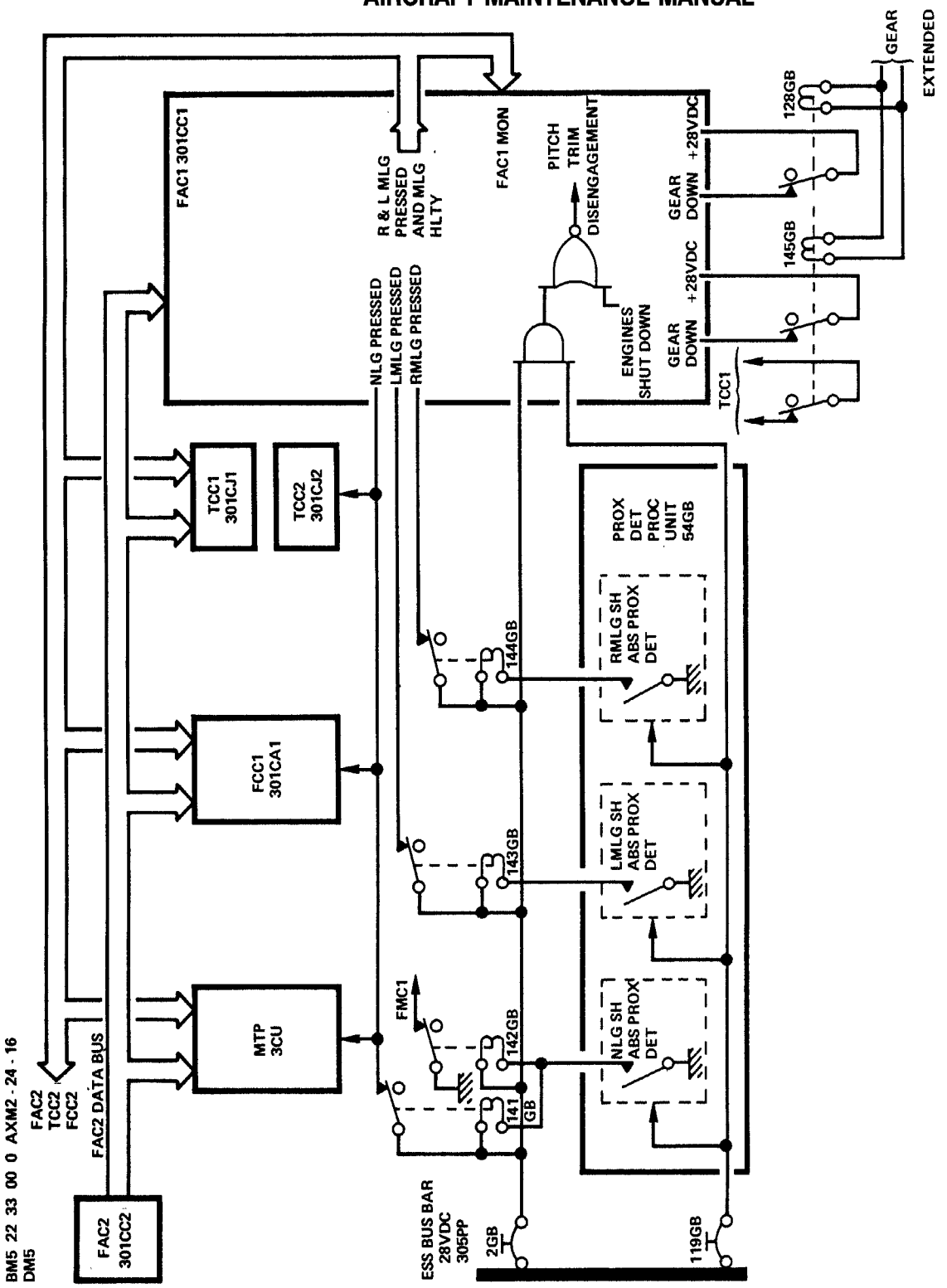


Interconnection between EFCU and TCC
Figure 009

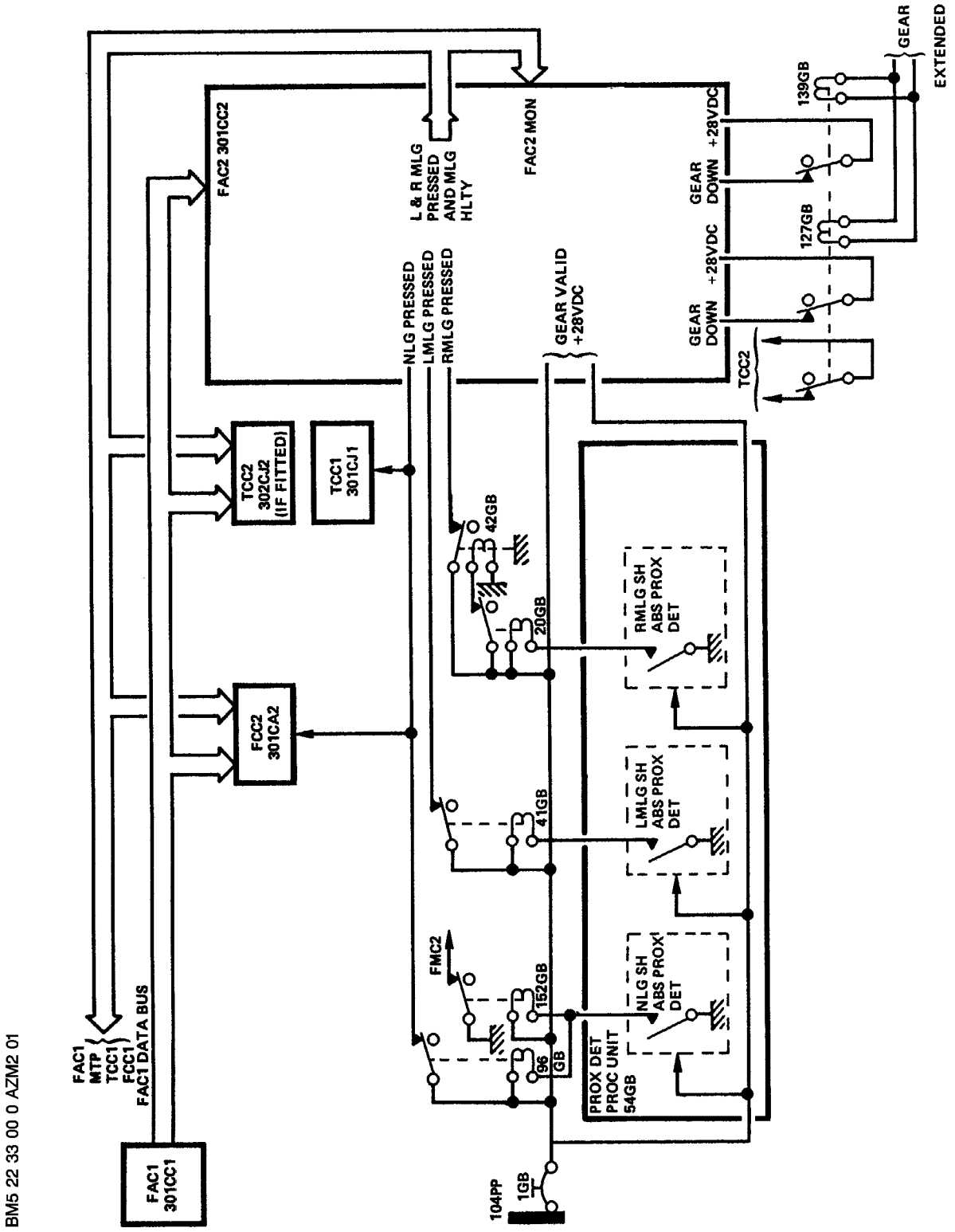
3. Aircraft Configuration Data

A. Flap and Slat Data
Slats ≤ 17° and flaps ≥ 20/15° data detected by the FAC are transmitted to the TCC via an ARINC data bus (ref. 22-34 FAC - TCC Arinc interface).
In the TCC each data signal (slats ≤ 17° or flaps ≥ 20/15°) is processed in the command and monitoring channels.
Slats ≤ 17° :
This position is taken into account in the AUTO mode engage logic.
Flaps ≥ 15° :
This data signal validates the engagement of GA mode and generation of the low limit logic.

B. Landing Gear Data
(1)Nose gear shock absorber compressed
Two nose gear shock absorber compressed data signals are fed to the computer from :
- relay 141GB associated with TCC2 (if fitted)
- relay 96GB associated with TCC1
(shock absorber compressed = relay contacts closed = 28 V).
This signal is used in the TO or FLX TO mode engage logic.
(2)Main landing gears
LH and RH gear shock absorber compressed data signals are processed through an OR-gate to disengage A/THR mode as soon as the throttle control levers have reached the idle position or to validate engagement of TO or FLX TO mode.
These gear shock absorber compressed signals (28 V) are transmitted by the FAC to the TCC.
(3)Retracted gear
A gear retracted data signal is fed to the computer from relay 128GB for TCC1 and 139GB for TCC2. This signal is in provision and will be brought into use in computation of decelerated approaches.
Type of signal : 28 V = downlocked gear
open circuit in all other cases.
(Ref. Fig. 010)
(Ref. Fig. 011)



Interconnection between Shock Absorber Proximity Detectors and AFS (Side 1)
Figure 010



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Interconnection between Shock Absorber Proximity Detectors and AFS (Side 2)
Figure 011

C. Air Conditioning Data - Engine and Wing Anti-Icing

For each mode, EPR lim = f(T) curves are given for the following basic air bleed conditions :

- takeoff : no air bleed for aircraft systems,
- all other modes : air is bled for normal aircraft air conditioning.

A THR correction function of altitude is introduced for any air bleed configuration different from the basic conditions mentioned above.

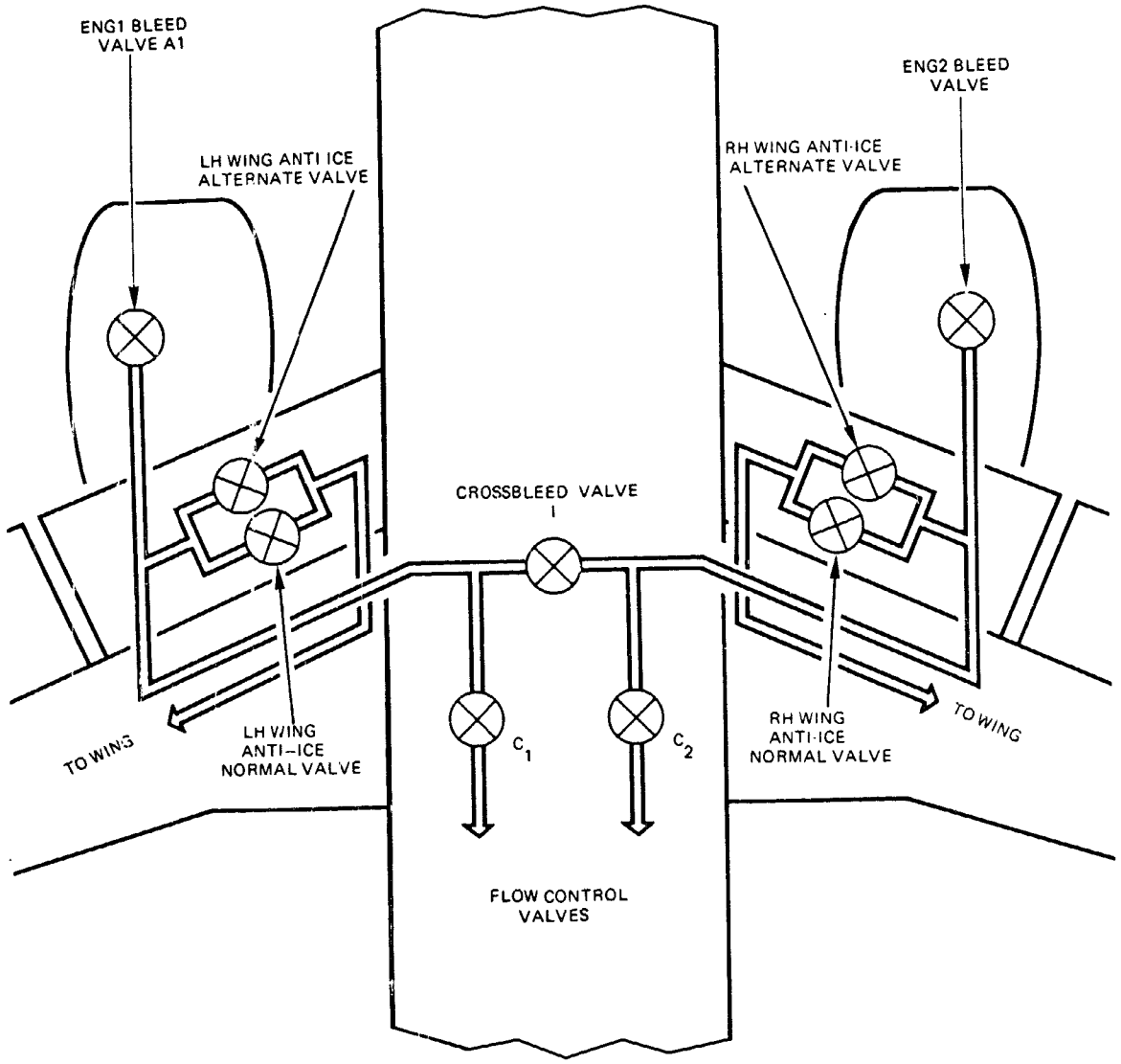
There are five air bleed configurations :

- no air bleed,
- conditioning air bleed,
- normal wing anti-icing air bleed,
- alternate wing anti-icing air bleed,
- nose cowl anti-icing air bleed.

Each one of these configurations is generated by the computer with respect to the position of the following valves :

- engine bleed valves : valve A1 for engine 1
  valve A2 for engine 2
- flow control valves : - valve C1 for pack associated with engine 1
  - valve C2 for pack associated with engine 2
- wing anti ice valves : - V1 for wing associated with engine 1
  - V2 for wing associated with engine 2
- nose cowl anti ice valves : - E1 = engine 1 nose cowl anti-icing
  - E2 = engine 2 nose cowl anti-icing
- cross feed valve 1.

The computer determines three EPR correction values via the logic circuits. (Ref. Fig. 012)



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Ice Protection and Bleed Air Supply Systems - Component Location
Figure 012

1. Deactivation of the autothrottle actuator 12CJ

A. Reason for the Job
Ref. MMEL 22-30-01 3 - Autothrottle Actuator (FAA only)

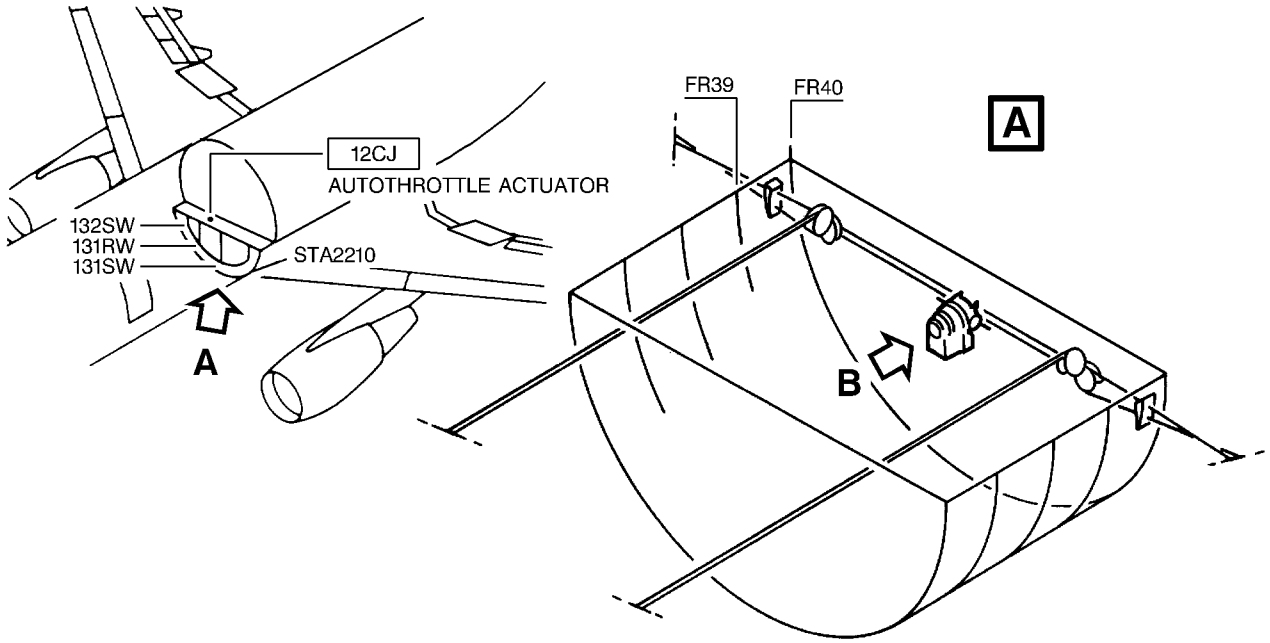
B. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 2,7 m (9 Ft.)
B.	Circuit Breaker Safety Clips
C.	Blanking Caps
D.	Warning Notice
E. NSA 935401-XX	Tie-Cable
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 53-10-55, P. Block 401	Removal of FWD Cargo Compartment Fixed Partition

C. Procedure
(1)Job Set-Up
(a)Energize the ground service network.
(b)Make certain that ATS lever(s) is (are) in OFF position.
(c)Display warning notice on the mechanical center pedestal 9VU prohibiting operation of the throttle control levers.
(d)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-TCC SYS1/28VDC	306CJ1	102/H 8
21VU	AFS-TCC SYS2/28VDC	306CJ2	102/H12

(e)Put the access platform in position below the FWD cargo compartment door.
(f)Open the FWD cargo compartment door.
(g)Remove sidewall panels 131SW, 132SW and 131RW.
(2)Deactivation of the Autothrottle Actuator 12CJ
(Ref. Fig. 901).
(a)Disconnect electrical connector (2) from receptacle (3).
(b)Put blanking caps on the disconnected connector.
(c)Attach the electrical connector with tie-cable on the structure (1) located above the actuator.
R (d)Remove the safety clips and tags and close circuit breakers 306CJ1
R and 306CJ2.
(3)Close-up
(a)Remove warning notice.
(b)On the FAC/ATS engage units, place placard near ATS lever(s) warning flight crew that autothrottle is inoperative.



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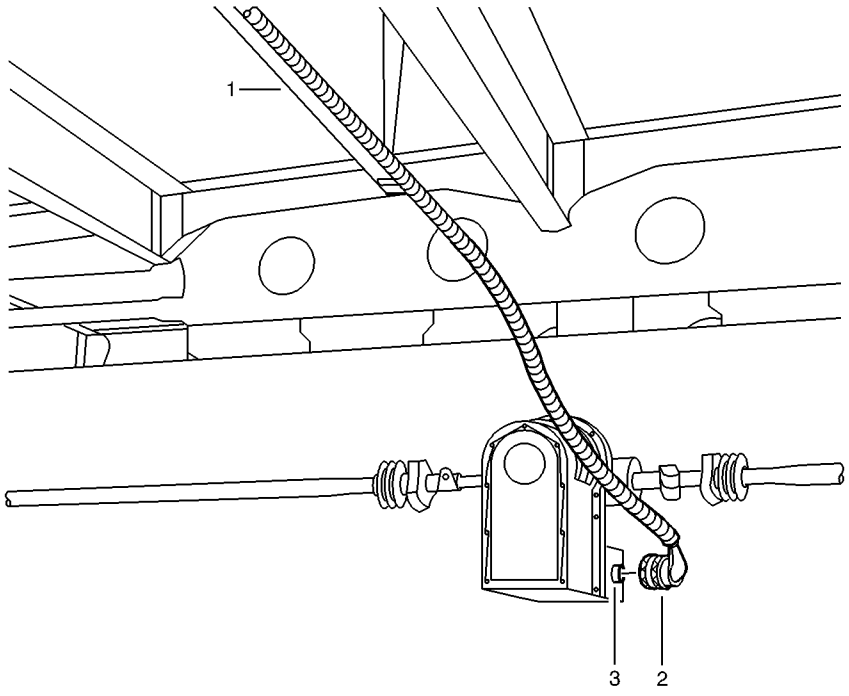

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Autothrottle Actuator
Figure 901

(c)Make certain that the work area is clean and clear of tool(s) and other items.
(d)De-energize the aircraft.
(e)In forward cargo compartment install sidewall panels 131SW, 132SW and 131RW.
(f)Close the FWD cargo compartment door.
(g)Remove the access platform.
(h)Move throttle control levers from FORWARD position to IDLE position several times to check the throttle control levers operate manually.

2. Reactivation of the autothrottle actuator

A. Equipment and Materials

ITEM	DESIGNATION
A.	Access Platform 2,7 m (9 Ft.)
B.	Circuit Breaker Safety Clips
C.	Warning Notice
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 22-30-00, P. Block 501	Autothrottle Operational Test

B. Procedure
(1)Job Set-Up
(a)Energize the ground service network.
(b)Display warning notice on the mechanical center pedestal 9VU prohibiting operation of the throttle control levers.
(c)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-TCC SYS1/28VDC	306CJ1	102/H 8
21VU	AFS-TCC SYS2/28VDC	306CJ2	102/H12

(d)Put the access platform in position below the FWD cargo compartment door.
(e)Open the FWD cargo compartment door.
(f)Remove sidewall panels 131SW, 132SW and 131RW.
(2)Reactivation of the Autothrottle Actuator 12CJ (Ref. Fig. 901)
(a)Detach electrical connector on the structure (1).
CAUTION : Do not cut cable located along the structure
(b)Remove the blanking caps from the electrical connector (2).
(c)Connect electrical connector (2) to receptacle (3).
R (d)Remove the safety clips and tags and close circuit breakers 306CJ1
R and 306CJ2.
(3)Test
Carry out the autothrottle operational test (Ref. 22-30-00, P. Block 501 § 1.A.).
(4)Close-up
(a)Remove warning notice.

(b)On the FAC/ATS engage units, place placard near ATS lever(s)
(c)Make certain that the work area is clean and clear of tool(s) and other items.
(d)De-energize the aircraft.
(e)In forward cargo compartment install sidewall panels 131SW, 132SW and 131RW.
(f)Close the FWD cargo compartment door.
(g)Remove the access platform.

1. Removal/Installation of Throttle Control Lever

A. Equipment and Materials

ITEM	DESIGNATION
(1)	Circuit Breaker Safety Clips
(2)65301A76110413	Special Screwdriver
R (3)Material No. 08-007A	Bonding and Adhesive Compounds (Ref. 20-31-00)

(Ref. Fig. 401)

B. Procedure

(1)Job set-up

(a)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-AP1 FD1 FCC1 28VDC	306CA1	102/H3
21VU	AFS-AP2 FD2 FCC2 28VDC	306CA2	102/H6
21VU	AFS-TCC SYS 1 28VDC	306CJ1	102/H8
21VU	AFS-TCC SYS 2 28VDC (IF FITTED)	306CJ2	102/H12
21VU	AFS-TCC SYS 1 115VAC	308CJ1	102/H9
21VU	AFS-TCC SYS 2 115VAC (IF FITTED)	308CJ2	102/H11

(2)Removal

(a)On center pedestal:

For the removal of right control lever :
- Remove screw (24) securing handle (25).
- Remove handle (25).
- Remove screws (26) securing cover plate (27).
- Remove cover plate (27).

For the removal of left control lever :
- Remove pin (20) and handle (21).
- Remove screws (22) securing cover plate (23).
- Remove cover plate (23).

(b)On throttle control lever :
- Remove screws (1), (2) and cup washer (3).
- Remove cover assy (4).
- Remove special bolt (19) with special screwdriver.
- Remove screws (14), washers (15) securing tab (16).
- Remove tab (16).
- Remove cotter pins (5), castle nuts (6) and support assy (7).
- Remove hex head bolts (13) and washers (12).
- Extract bushes (11).

(c)Rock throttle control lever (17) forward to gain access to electrical connector (10). Remove screws (9) and disconnect electrical connector (10).

(d)Remove throttle control lever (17) from link (18).

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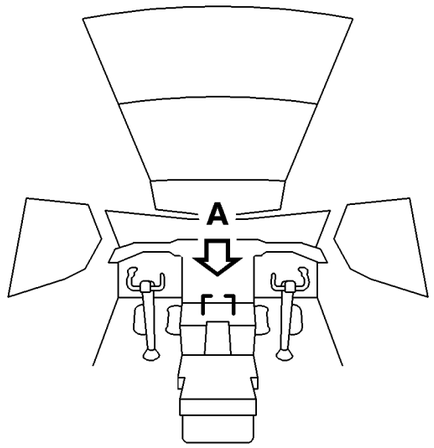
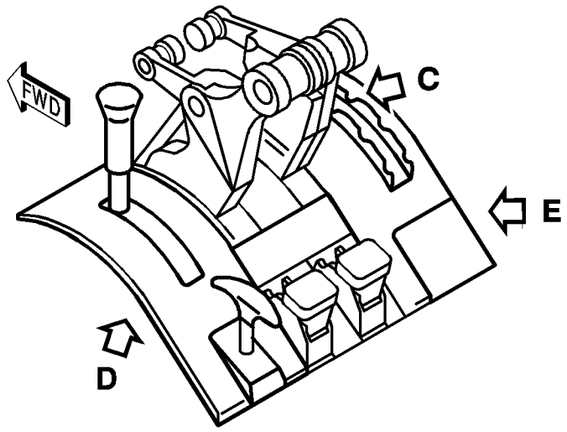
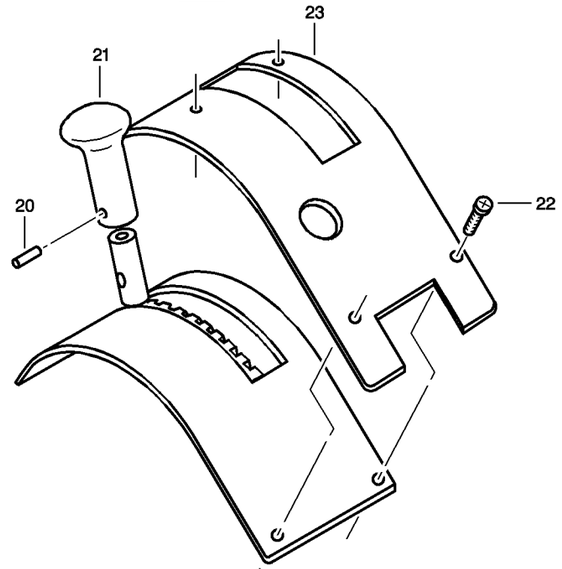
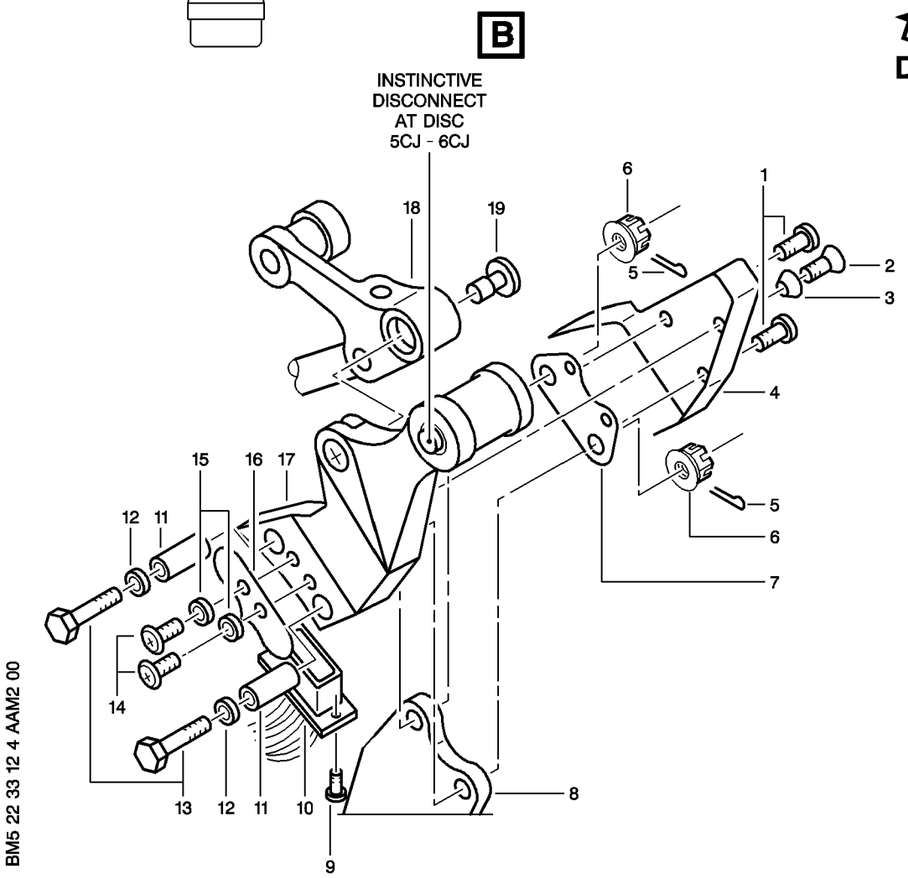
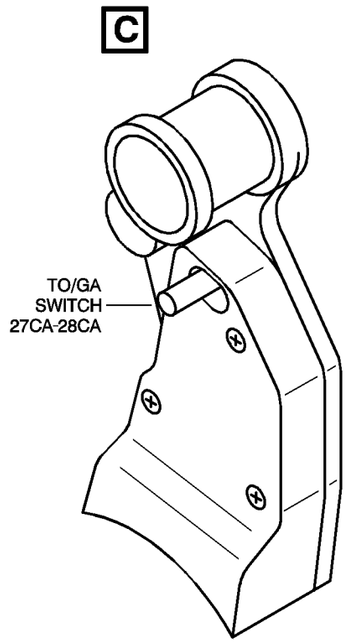
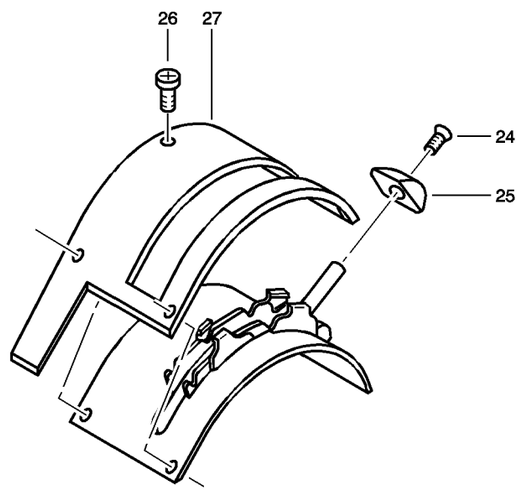












Throttle Control Lever
Figure 401

(3)Preparation of replacement component
(a)Make certain that replacement throttle control lever is clean (no trace of protective coating) and free of dents and corrosion.
(b)Clean and inspect component adjacent area for correct condition.
(4)Installation
R NOTE : All screws and bolt (19) are secured with bonding and adhesive
R compound (Material No. 08-007A).
(a)Install throttle control lever (17) on link (18).
(b)Connect electrical connector (10) and secure with screws (9).
(c)Install throttle control lever (17) on its support (8).
(d)Insert bushes (11) in previous holes.
(e)Install hex head bolts (13), washers (12) and support assy (7), secure with castle nuts (6) and cotter pins (5).
(f)Install tab (16), washers (15) and secure with screws (14).
(g)Install special bolt (19) with special screwdriver.
(h)Install cover assy (4), cup washer (3) and secure with screws (1), (2).
(i)For the installation of right side control lever :
- Install cover plate (27) and secure with screws (26).
- Install handle (25) and secure with screw (24).
For the installation of left side control lever :
- Install cover plate (23) and secure with screws (22).
- Install handle (21) and insert pin (20).
(j)Remove safety clips and tags and close circuit breakers 306CA1, 306CA2, 306CJ1, 306CJ2 (If fitted), 308CJ1, 308CJ2 (If fitted).
(5)Test
Carry out operational test of autothrottle (Ref. 22-30-00, P. Block 501 Para. 1.B).
(6)Close-up
(a)De-energize the aircraft electrical network and disconnect electrical ground power unit.
(b)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(c)Restore system and aircraft to normal operating condition.

R THROTTLE POSITION DETECTORS 313CJ, 314CJ - REMOVAL/INSTALLATION

R

NOTE : Removal/Installation procedures for throttle position detectors 313CJ and 314CJ being identical, only procedure concerning detector 313CJ is described.

1. Equipment and Materials

ITEM	DESIGNATION
A. 99A76101003000	Rigging Pin - Throttle Control Lever
B. 16987886-003	Drift Pin - Microswitch Unit
C.	Warning Notice (Prohibiting Operation of Throttle Control Levers)
D.	Blanking Caps
E.	Circuit Breaker Safety Clips
F.	Lockwire
G.	Cotter Pins
Referenced Procedure - 22-30-00, P. Block 501	Autothrottle

2. Procedure

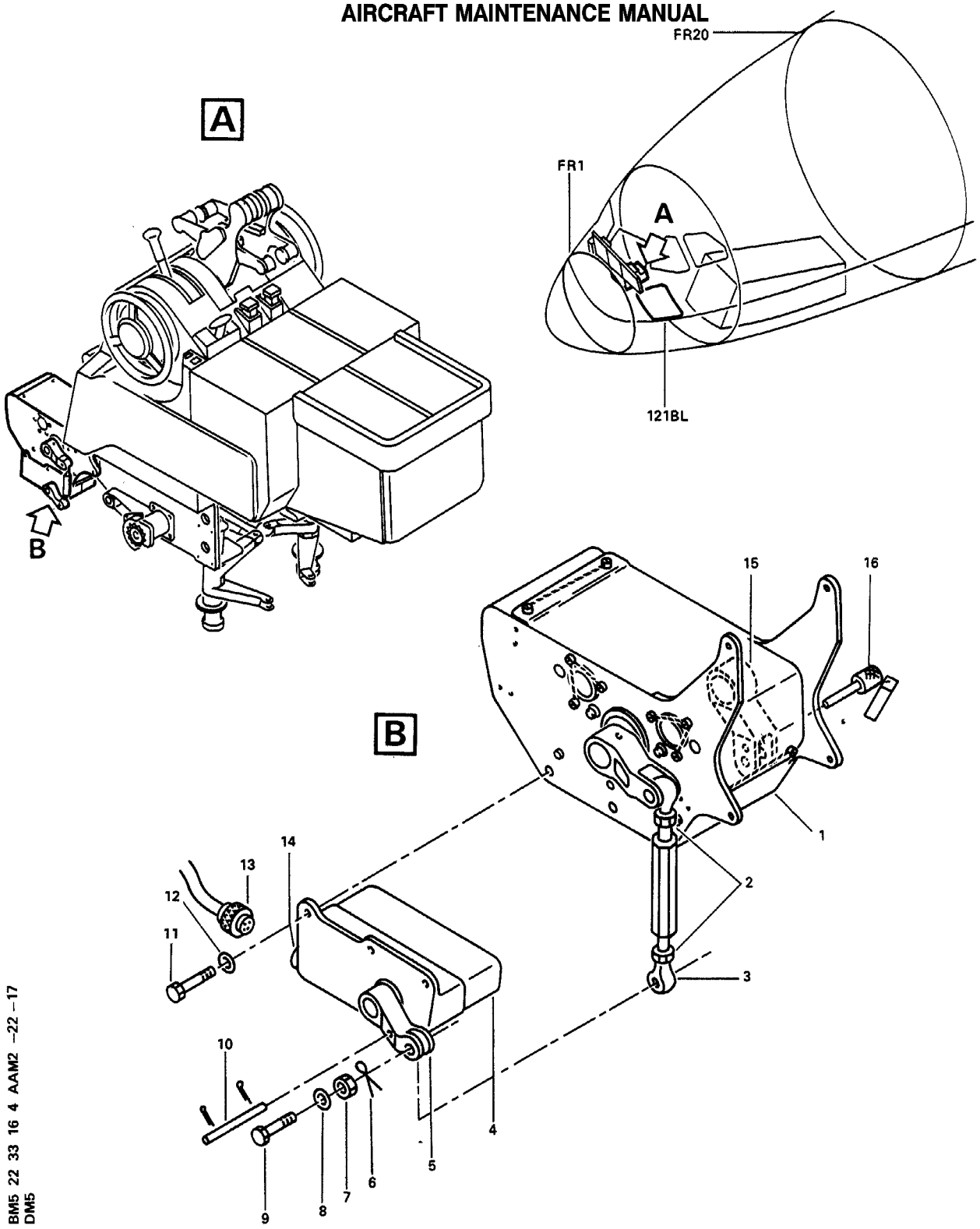
A. Job Set-Up
(1)On mechanical center pedestal 9VU place throttle control levers in idle position and display warning notice prohibiting their operation.
(2)Open, safety and tag the following circuit breaker :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS - TCC SENSORS	305CJ1	102/H10

(3)Open access door 121BL and gain access to throttle position detector to be removed.

B. Removal (Ref. Fig. 401)
(1)Disconnect electrical connector (13) and cap connectors (13) and (14).
(2)Disengage rod (3) from lever (5) as follows :
Remove and discard cotter pin (6), remove and retain nut (7) with washer (8) and bolt (9).
(3)Remove rod (3) from lever (5).
(4)Cut and remove lockwire from the three bolts (11). Remove and retain the three bolts (11), and the three washers (12).
(5)Remove throttle position detector (4).

C. Installation
(1)Clean and inspect throttle position detector interface and/or adjacent area.
(2)Make certain that replacement component is clean and free from damage



Throttle Position Detector
Figure 401

or corrosion.
(3)Position throttle position detector (4), install the three washers (12) and the three bolts (11) and tighten. Safety with lockwire.
(4)Engage rod (3) in lever (5) yoke.
(5)Install bolt (9), washer (8) and nut (7) and tighten. Safety with cotter pin (6).
(6)Adjust detector ("0" mechanical position).
(a)Remove drift pin (10) with its cotter pins on detector (4).
NOTE : The detector is delivered by the supplier with drift pin installed.
(b)Rig bellcrank (15) with rigging pin (16) on microswitch unit (1).
(c)Rig detector (4) with drift pin (10) and make certain that detector can be easily rigged; if not, adjust rod (3) with both nuts (2).
(d)Tighten both nuts (2) and safety with lockwire.
(e)Remove drift pin (10) and rigging pin (16).
(7)Remove blanking caps from electrical connectors (13) and (14) and connect.
(8)Remove safety clip and tag and close circuit breaker 305CJ1.

D. Test
NOTE : This test is carried out by means of DADC test set. If this test set is not available, the ground pressure generator must be connected to the main air data systems.
(1)Reason for the job
To check correct rigging of throttle position detectors 313CJ and 314CJ.
(2)Equipment and Materials

ITEM	DESIGNATION
(a)35M5M01150001	Test Set-DADC or Generator - Ground Pressure with Tolerance ≤ 0.30 mbar.
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure
(a)Job Set-up

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

1 Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
2 Make certain that electronics rack ventilation is correct.
3 Make certain that flaps and slats are retracted.
4 Make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU.
5 Align the IRS 1, 2 and 3 (Ref. 34-25-00, P. Block 501).
For this test, it is not necessary :

- To continue the ALIGN procedure after the red HDG flags on NDs disappear
- To enter current position of the aircraft.

6 Place FD/FPV selector switch in OFF position on panels 412VU and 413VU

7 Make certain that ADS is operating.

8 Make certain that PITCH TRIM 1 or 2 lever is in ON position.

9 With DADC test set :
- Connect the DADC test set to electrical connectors 62FL1 (located in electronics rack 90VU) and 62FL2 (located in electronics rack 80VU)
- Place S1 switches in OFF position
- Place S2 switches in OFF position
- Place S3 switches in VC position
- Turn Hp/Vc RATE knobs to middle position.

or

With ground pressure generator :
- Remove blanking covers from CAPT and F/O static probes
- Remove slip-on covers from CAPT and F/O pitot probes
- Connect ground pressure generator to CAPT and F/O static probes and to CAPT and F/O pitot probes with appropriate equipment
- Start up ground pressure generator.

(b)Test

NOTE : Test after removal/installation of the two throttle position detectors being identical, only test concerning detector 313CJ is described. For detector 314CJ, read information between parentheses.

ACTION	RESULT
1. On overhead panel 20VU on the FAC/ATS engage unit : - Place ATS1(2) arming lever in ON position.	
2. On center pedestal : - Move Left (Right) throttle control lever to 50° TLA position and Right (Left) one to 47° TLA position.	The throttle control lever remains in this position. On Primary Flight Display (PFD) : - Amber MAN THR is displayed.
3. On DADC test set : - Place S1 switches in ON position - Place S2 switches in BIT INITIATE position - Place S3 switches in Vc position - Using Hp/Vc RATE knobs, select an airspeed of 150 Kts on Captain's and First Officer's PFDs. or On ground pressure generator,	On CAPT and F/O PFDs, airspeed scale

ACTION	RESULT
slowly increase the impact pressure Qc to display an airspeed of 150 Kts on CAPT and F/O PFDs.	stops scrolling and remains stable at 150 Kts.
4. On Flight Control Unit (FCU) : - Turn the SPEED/MACH selector knob until 150 is displayed in the SPD/MACH display window.	On CAPT and F/O PFDs : - The cyan selected airspeed symbol is displayed next to 150 Kts.
5. On FCU : - Press A/THR pushbutton switch.	On FCU : - On the A/THR pushbutton switch, the green bars come on. On CAPT and F/O PFDs, on the FMA : - SPD is displayed.
6. On DADC test set : - Slightly turn the Hp/Vc RATE knob to increase airspeed (by about 3 Kts) until throttle control levers begin to move back. or On pressure generator, slowly increase the impact pressure Qc by about 3 Kts until throttle control levers begin to move back.	On center pedestal : - Both throttle control levers move gradually to the idle position - When the Right (Left) throttle control lever is at the idle position, the Left (Right) throttle control lever is at 6° TLA.
7. Place ATS arming lever in OFF position.	
8. De-energize AFS components by placing PITCH TRIM levers in OFF position.	

(c)Close-up
1 Disconnect DADC test set
or
Disconnect ground pressure generator:
- Install blanking covers on CAPT and F/O static probes
- Install slip-on covers on CAPT and F/O pitot probes.

2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.
4 Make certain that throttle control levers are in idle position.

E. Close-up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

(2)On mechanical center pedestal 9VU, remove warning notice.
(3)Close access door 121BL.

THROTTLE POSITION DETECTORS 313CJ, 314CJ - REMOVAL/INSTALLATION

NOTE : Removal/Installation procedures for throttle position detectors 313CJ and 314CJ being identical, only procedure concerning detector 313CJ is described.

1. Equipment and Materials

	ITEM	DESIGNATION
R	A. 99A76101003000	Rigging Pin - Throttle Control Lever
R	B. P13736	Drift Pin - Microswitch Unit
	C.	Warning Notice (Prohibiting Operation of Throttle Control Levers)
	D.	Blanking Caps
	E.	Circuit Breaker Safety Clips
	F.	Lockwire
	G.	Cotter Pins
	Referenced Procedure - 22-30-00, P. Block 501	Autothrottle

2. Procedure

A. Job Set-Up

(1) On mechanical center pedestal 9VU place throttle control levers in idle position and display warning notice prohibiting their operation.

(2) Open, safety and tag the following circuit breaker :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS - TCC SENSORS	305CJ1	102/H10

(3) Open access door 121BL and gain access to throttle position detector to be removed.

B. Removal (Ref. Fig. 401)

R (1) Disconnect electrical connector (14) and cap connectors (14) and (15).

(2) Disengage rod (3) from lever (5) as follows :

- Remove and discard cotter pin (6), remove and retain nut (7) with washer (8) and bolt (9).

(3) Remove rod (3) from lever (5).

R (4) Cut and remove lockwire from the three bolts (13).

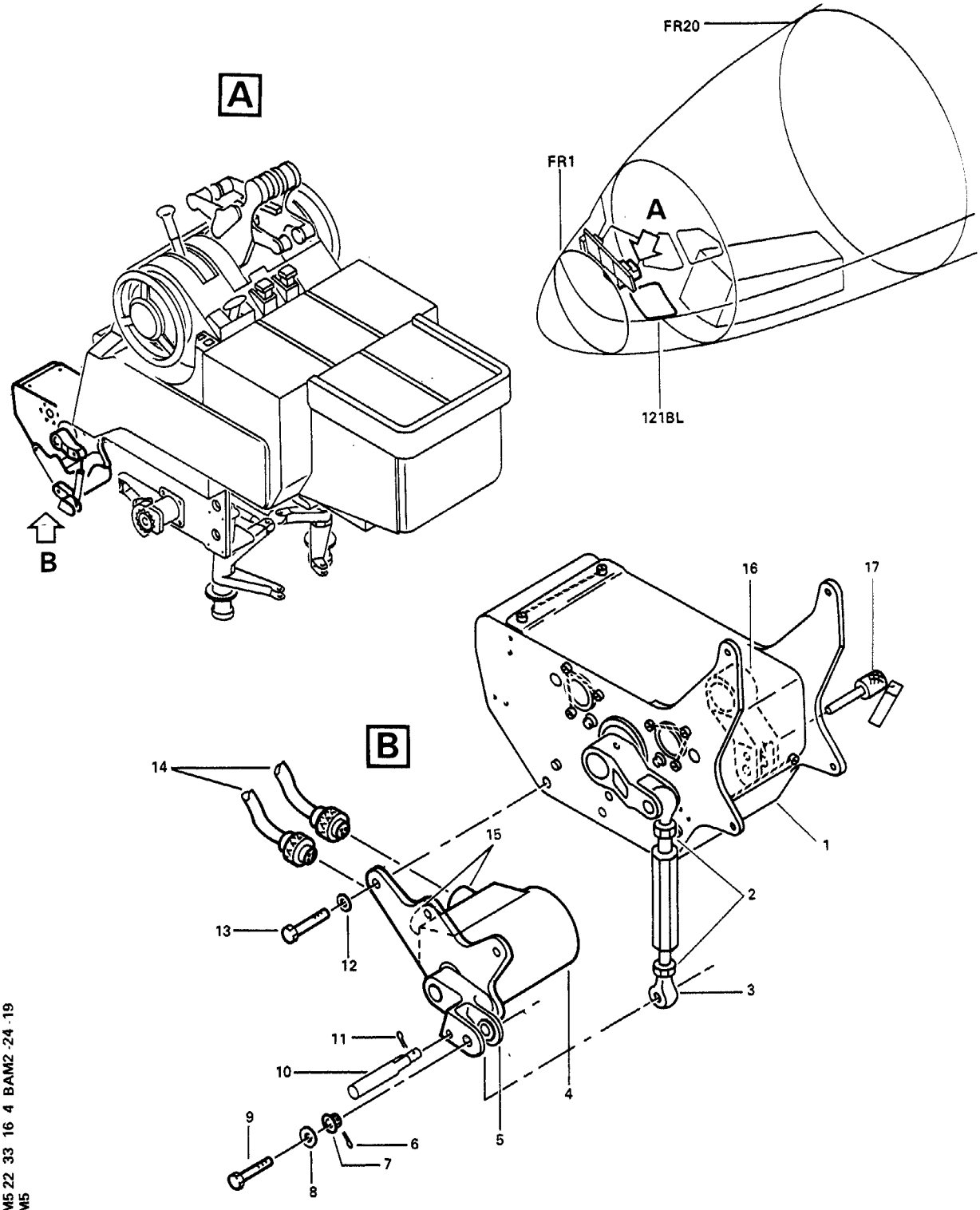
R (5) Remove and retain the three bolts (13) and the three washers (12).

(6) Remove throttle position detector (4).

C. Installation

(1) Clean and inspect throttle position detector interface and/or adjacent area.

(2) Make certain that replacement component is clean and free from damage or corrosion.



Throttle Position Detector
Figure 401

(3)Position throttle position detector (4), install the three bolts (13) and the three washers (12) and tighten. Safety with lockwire.
(4)Engage rod (3) in lever (5) yoke.
(5)Install bolt (9), nut (7) with washer (8) and tighten. Safety with cotter pin (6).
(6)Adjust detector (4) ("0" mechanical position).
NOTE : The detector is delivered by the supplier with drift pin (10) installed.
(a)Remove drift pin (10) with its cotter pins on detector (4).
(b)Rig bellcrank (16) with rigging pin (17) on microswitch unit (1).
(c)Rig detector (4) with drift pin (10) and its cotter pin and make certain that detector can be easily rigged; if not, adjust rod (3) with both nuts (2) (after cutting and removing lockwire).
(d)Tighten both nuts (2) and safety with lockwire.
(e)Remove drift pin (10) and rigging pin (17).
(7)Remove blanking caps from electrical connectors (14) and (15) and connect.
(8)Remove safety clip and tag and close circuit breaker 305CJ1.

D. Test
NOTE : This test is carried out by means of DADC test set. If this test set is not available to the operator, the pressure generator must be connected to CAPT and F/O static probes and to CAPT and F/O pitot probes with appropriate equipment.
(1)Reason for the job
To check correct rigging of throttle position detectors 313CJ and 314CJ.
(2)Equipment and Materials

ITEM	DESIGNATION
(a)35M5M01150000 or 35M5M01150001	Test Set-DADC
Referenced Procedures	
- 24-41-00, P. Block 301	AC External Power Control
- 34-25-00, P. Block 501	Inertial Reference System

(3)Procedure
(a)Job Set-up

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

1 Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
2 Make certain that electronics rack ventilation is correct.
3 Make certain that flaps and slats are retracted.
4 Make certain that EFIS is operating and adjust PFD image brightness on panels 412VU and 413VU.
5 Align the IRS 1, 2 and 3 (Ref. 34-25-00, P. Block 501).
For this test, it is not necessary :

- To continue the ALIGN procedure after the red HDG flags on NDs disappear
- To enter current position of the aircraft.

6 Make certain that ADS is operating.
7 Make certain that PITCH TRIM 1 or 2 lever is in ON position.
8 Connect the DADC test set to electrical connector 62FL1 (located in electronics rack 90VU) and 62FL2 (located in electronics rack 80VU).
9 On DADC test set :
- Place S1 switches in OFF position
- Place S2 switches in OFF position
- Place S3 switches in VC position
- Turn Hp/Vc RATE knobs to middle position.

(b)Test

NOTE : Test after removal/installation of the two throttle position detectors being identical, only test concerning detector 313CJ is described. For detector 314CJ, read information between parentheses.

ACTION	RESULT
1. On overhead panel 20VU on the FAC/ATS engage unit : - Place ATS1(2) arming lever in ON position.	
2. On center pedestal : - Move Left (Right) throttle control lever to 50° TLA position and Right (Left) one to 47° TLA position.	The throttle control lever remains in this position. On Primary Flight Display (PFD) : - Amber MAN THR is displayed.
3. On DADC test set : - Place S1 switches in ON position - Place S2 switches in BIT INITIATE position - Place S3 switches in Vc position - Using Hp/Vc RATE knobs, select an airspeed of 150 Kts on CAPT and F/O PFDs. or On ground pressure generator, slowly increase the impact pressure Qc to display an airspeed of 150 Kts on CAPT and F/O PFDs.	On CAPT and F/O PFDs, airspeed scale stops scrolling and remains stable at 150KTS.
4. On Flight Control Unit (FCU) : - Turn the SPEED/MACH selector knob until 150 is displayed in the SPD/MACH display window.	On CAPT and F/O PFDs : - The cyan selected airspeed symbol is displayed next to 150 Kts.

R
R

ACTION	RESULT
5. On FCU : - Press A/THR pushbutton switch.	On FCU : - On the A/THR pushbutton switch, the green bars come on. On CAPT and F/O PFDs, on the FMA : - SPD is displayed.
6. On DADC test set : - Slightly turn the Hp/Vc RATE knob to increase airspeed (by about 3 Kts) until throttle control levers begin to move back. or On ground pressure generator, slowly increase the impact pressure Qc by about 3 Kts until throttle control levers begin to move back.	On center pedestral : - Both throttle control levers move grdually to the idle position - When the Right (Left) throttle control lever is at the idle position, the Left (Right) throttle control lever is at 6° TLA.
7. Place ATS arming lever in OFF position.	
8. De-energize AFS components by placing PITCH TRIM levers in OFF position.	

(c)Close-up
1 Disconnect DADC test set.
2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
3 Restore aircraft and system to normal operating condition.
4 Make certain that throttle control levers are in idle position.

E. Close-up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)On mechanical center pedestal 9VU, remove warning notice.
(3)Close access door 121BL.

AUTOTHROTTLE ACTUATOR 12CJ - REMOVAL/INSTALLATION

CAUTION : BEFORE ATTEMPTING REMOVAL/INSTALLATION PROCEDURES FOR THE AUTO-THROTTLE ACTUATOR, DISPLAY A WARNING NOTICE PROHIBITING OPERATION OF THE THROTTLE CONTROL LEVERS.

1. Equipment and Materials

ITEM	DESIGNATION
A. 98A76101003000	Pin - Rigging, Autothrottle Coupling Unit
B.	Access Platform 2.7 m (9 ft.)
C.	Circuit Breaker Safety Clips
D.	Blanking Caps
E.	Cotter Pins
F.	Warning Notice (prohibiting operation of throttle control levers)
G.	Lockwire Dia. 0.6 mm (0.023 in.)
H. Material No. 04-004	Common Greases (Ref. 20-31-00)
I.	Bolts and Captive Washer Locknuts
Referenced Procedures	
- 22-30-00, P. Block 501	Autothrottle
- 20-28-11, P. Block 1 (Ref. Fig. 401).	Electrical Bonding

2. Procedure

A. Job Set-Up

(1)Open, safety and tag the following circuit breakers :

R **ON A/C 101-200,

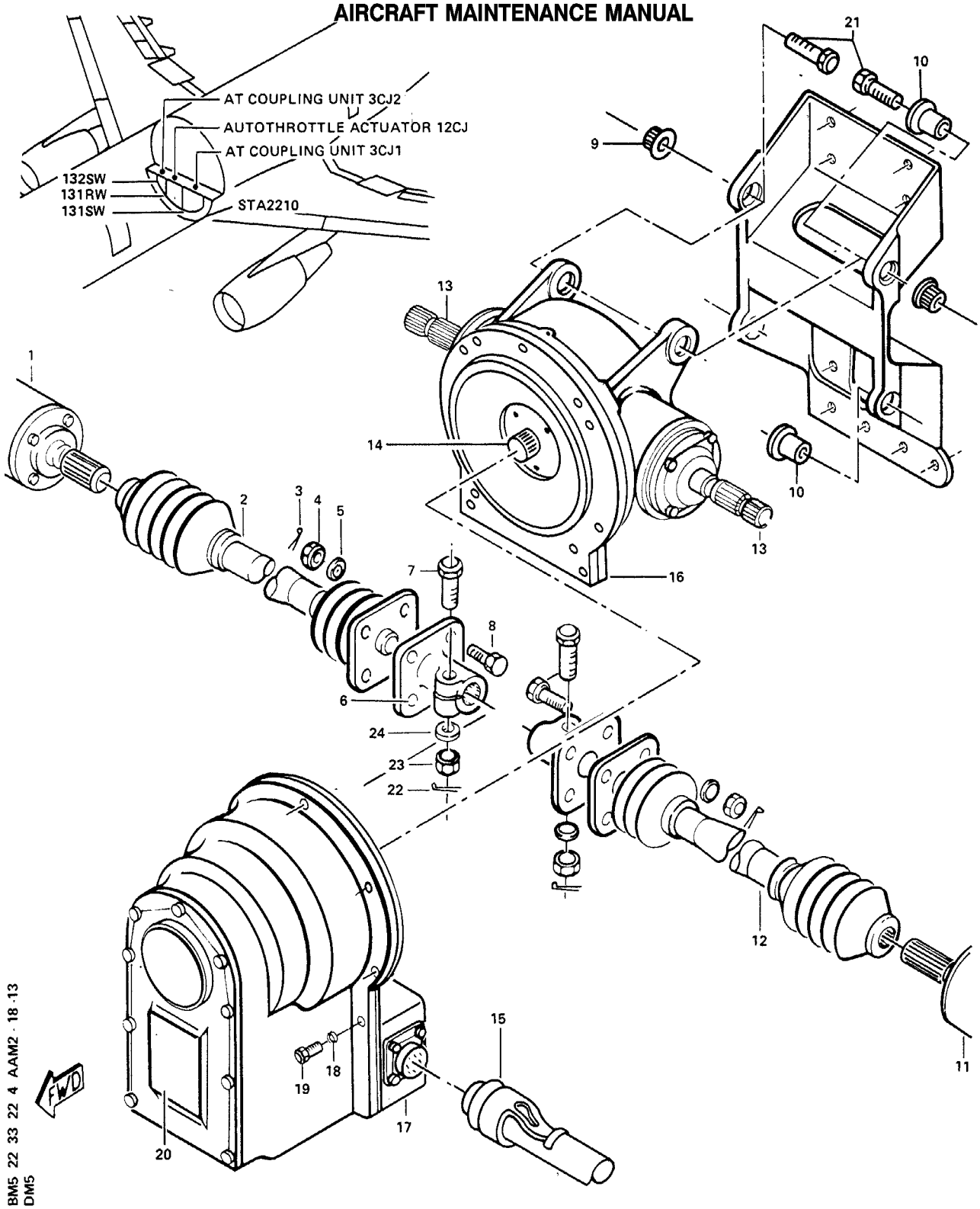
PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-TCC SYS1/28VDC	306CJ1	102/H 8

R **ON A/C 001-100,

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-TCC SYS1/28VDC	306CJ1	102/H 8
21VU	AFS-TCC SYS2/28VDC	306CJ2	102/H12

R **ON A/C ALL

(2)Display warning notice on mechanical center pedestal 9VU prohibiting operation of throttle control levers.
(3)Position access platform at forward cargo compartment door.
(4)Open FWD cargo compartment door.
(5)In forward cargo compartment remove sidewall panels 131SW, 131RW and 132SW.
(6)Install rigging pins on left and right autothrottle coupling units



Autothrottle Actuator
Figure 401

(1) and (11).

B. Removal
(1)Removal of Actuator
(a)Disconnect electrical plug (15) from receptacle (17) and protect with blanking cap.
(b)Cut and remove lockwire and remove the seven attaching screws (19) and washers (18) from actuator (20) on output gear (16). Retain screws (19) and washers (18).
(c)Remove actuator (20) from output gear (16).
(d)Protect splined shaft (14) on output gear with blanking cap.
(2)Removal of output gear
(a)Uncouple rigid transmission shafts (2) and (12).
1 Remove and discard the eight cotter-pins (3) ; remove and retain the eight nuts (4), the eight washers (5) and the eight bolts (8).
2 Separate rigid transmission shafts from splined shafts on both coupling units (1) and (11).
Take care not to damage rubber end pieces.
(b)Remove output gear (16).
1 Remove and discard the four captive washer locknuts (9) and the four bolts (21).
2 Remove output gear (16).
3 Check that the two spacers (10) are installed in mounting flange.
4 After output gear has been removed, remove and discard the two cotter-pins (22). Remove the two nuts (23), the two washers (24), the two bolts (7) securing the two flanges (6) to output gear output shafts (13).
5 Remove and retain the two flanges (6) on the output gear.

C. Preparation of Replacement Component
(1)Make certain that replacement component is clean and free from damage and corrosion.
(2)Clean and inspect actuator interface and adjacent area.
(3)Check condition of rigid transmission shafts (2) and (12) and make certain that universal joints are lubricated (Ref. Material No. 04-004).
(4)Position the two flanges (6) on splined output shafts of output gear (13).
(5)Install the two bolts (7), the two washers (24) and the two nuts (23), tighten nuts and lock with two cotter pins (22).
(6)Check bonding of output gear (upper L side attachment). (Ref. 20-28-11, P. Block 1).

D. Installation
(1)Installation of output gear.
(a)On support mount R side, check that the two spacers (10) are in place.
(b)Position output gear on its support mount.
(c)Install two new bolts (21) on support mount L side and two new captive washer locknuts (9). Tighten nuts.
(d)On support mount R side, install two new bolts (21) and two new captive washer locknuts (9). Tighten nuts.
(e)Install rigid transmission shafts (2) and (12).

(f)Engage end of universal joint on splined shaft on coupling unit (1) ; make certain that rubber bellows end covers splined shaft tip undercut.
(g)Mount flange of rigid transmission shaft (2) on flange (6) of actuator ; install the four bolts (8), the four washers (5) and the four nuts (4). Tighten and safety with cotter pins (3).
(h)Repeat operations (f) and (g) to install L rigid transmission shaft (12) on splined shaft of coupling unit (11).

(2)Installation of actuator.
(a)Remove blanking caps from splined shaft (14) on output gear (16).
(b)Install actuator (20) on output gear (16).
(c)Secure by means of the seven screws (19) and washers (18). TORQUE to 0.14 m.daN (12.4 lbf.in.).
(d)Wirelock the seven screws (19).
(e)Remove blanking cap and connect electrical plug (15) to receptacle (17).
(f)Remove rigging pins on both autothrottle coupling unit (1) and (11).

R **ON A/C 101-200,

(g)Remove safety clip and tag and close circuit breaker 306CJ1.

R **ON A/C 001-100,

(g)Remove safety clips and tags and close circuit breakers 306CJ1, 306CJ2.

R **ON A/C ALL

(h)Remove warning notice prohibiting operation of throttle control levers.

E. Test
Carry out an autothrottle operational test (Ref. 22-30-00, P. Block 501, para.1.A.).

F. Close-up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)In forward cargo compartment install sidewall panels 131SW, 132SW and 131RW.
(3)Close FWD cargo compartment door.
(4)Remove access platform from forward cargo compartment door.

AT COUPLING UNIT 3CJ1 (3CJ2) - REMOVAL/INSTALLATION

WARNING : BEFORE STARTING AT COUPLING UNIT REMOVAL/INSTALLATION PROCEDURES, DISPLAY A WARNING NOTICE PROHIBITING OPERATION OF THROTTLE CONTROL LEVER.
DISPLAY A WARNING NOTICE PROHIBITING OPERATION OF FLAP AND SLAT CONTROL LEVER.

NOTE : Removal/Installation procedures for left and right AT coupling units being identical, only procedure concerning left coupling unit 3CJ1 is described.

1. Equipment and Materials

ITEM	DESIGNATION
A. 98A27901005000	Pin-Rigging, Cable Tension Regulator
B. 98A76101003000	Pin-Rigging, Autothrottle Coupling Unit
C. 98A76103000000	Bar-Locking, Throttle Control, Cable Tension Regulator "0" Position
D. 99A76101501000	Pin-Rigging, Primary Transmission Crank Lever
E.	Warning Notice prohibiting operation of throttle control lever
F.	Warning Notice prohibiting operation of flap and slat control lever
G. Material No. 04-004	Common Greases (Ref. 20-31-00)
H.	Tensiometer
J.	Environment Thermometer
K.	Circuit Breaker Safety Clips
L.	Turnbuckle Safety Pins
M.	Cotter Pins
N.	Access Platform 2.70 m (9 ft.)
P.	Access Platform 4.10 m (13 ft. 5 in.)
Q.	Cable Clamps
R.	Bolts and Captive Washer Lock nuts
S.	Torque Wrench
Referenced Procedures	
- 22-30-00, P. Block 501	Autothrottle
- 76-11-00, P. Block 501	Throttle Controls

2. Procedure

A. Job Set-Up

(1)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-TCC SYS1/28VDC	306CJ1	102/H 8
21VU	AFS-TCC SYS2/28VDC	306CJ2	102/H12

(2)On mechanical center pedestal 9VU, position warning notices prohibiting operation of throttle control levers and flap and slat control lever.

(3)Make certain that throttle control lever is in 0° position.

(4)Position access platform on R side of engine pylon.

(5)Open access door 414GR on left engine pylon (424GR on right engine pylon) to gain access to primary transmission crank lever.

(6)Immobilize primary transmission crank lever (20) with rigging pin 99A76101501000 (21).

(7)Position access platform at FWD cargo compartment door and open FWD cargo compartment door.

(8)In forward cargo compartment at STA2210, remove sidewall panels 131SW, 132SW and 131RW.

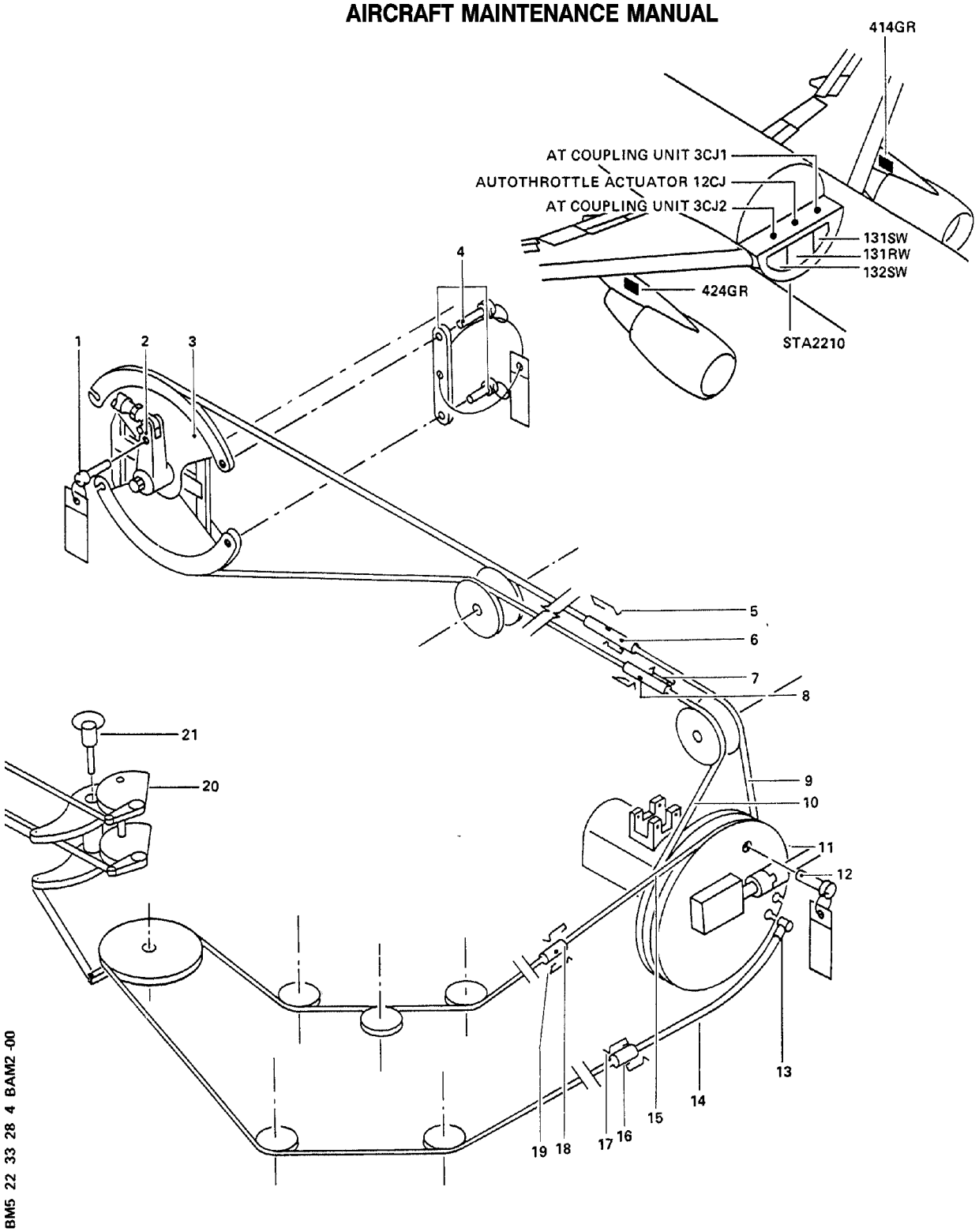
(9)Immobilize tension regulator (3) with rigging pin 98A27901005000 (1).

(10)Immobilize coupling unit pulley (11) with rigging pin 98A76101003000 (12).

(11)Extract safety pins (5), (7), (17) and (19).

(12)Install clamps on cables bearing against structure in order to keep cables tensioned upstream of coupling unit during removal of cables from coupling unit.

(13)Loosen turnbuckles (6) and (8) by the same amount until cable tension regulator locking bar 98A76103000000 (4) can be installed.
(Ref. Fig. 401)

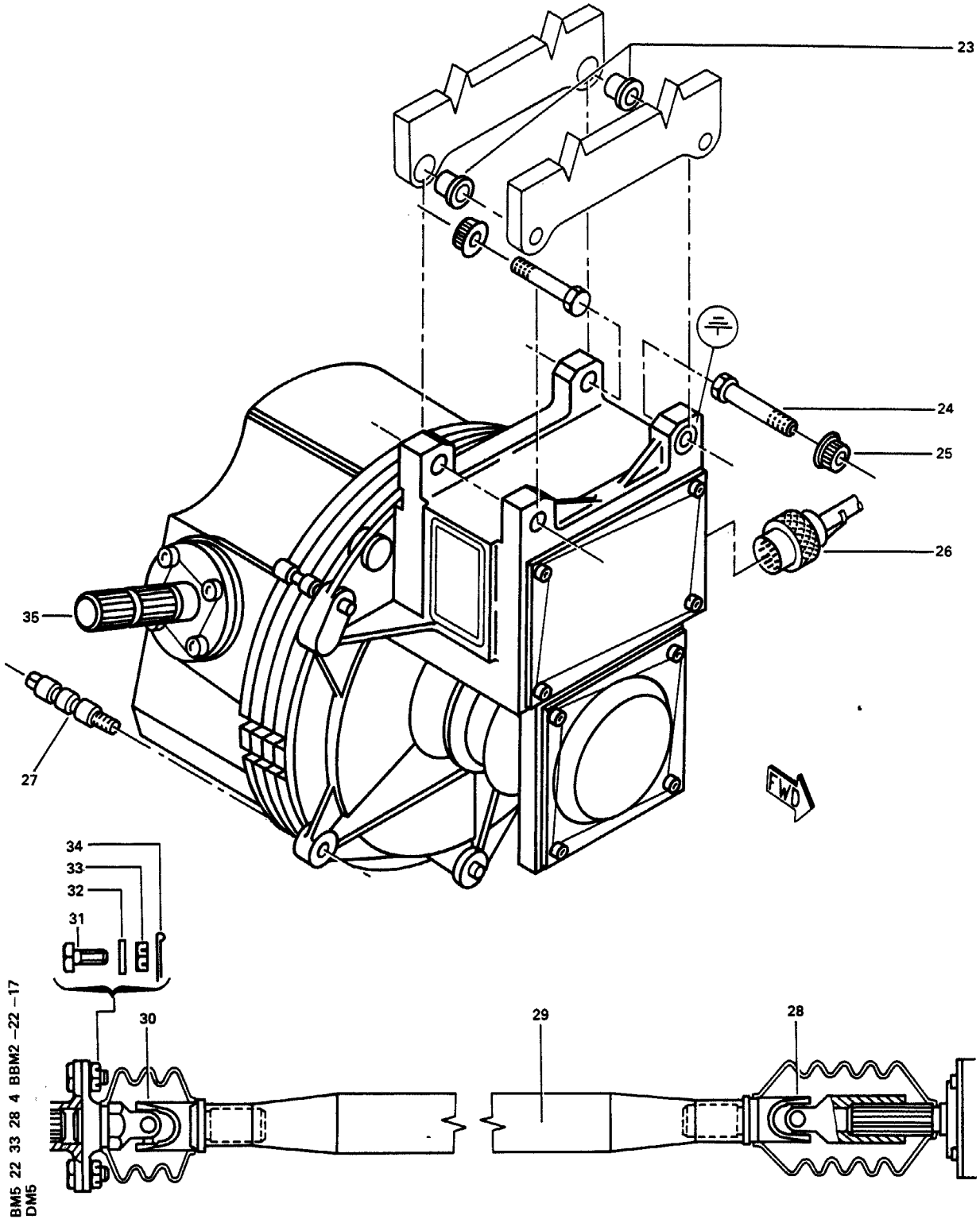


Throttle Control Linkage
Figure 401

B. Removal (Ref. Fig. 402)
(1)Disconnect electrical plug (25).
(2)Uncouple rigid transmission shaft (29).
(a)Remove and discard the four cotter pins (34), remove and retain the four nuts (33), the four washers (32) and the four bolts (31).
(b)Remove rigid transmission shaft (29) from coupling unit (35) (take care not to damage rubber bellows covering gimbal joints).
(3)Remove the seven cable guides (27) from coupling unit.
(4)Remove cables from pulleys. Loosen turnbuckles (6), (8), (16) and (18) by the same amount and by rotating barrels (13) a quarter turn, push cables outwards. Note cable location.
(5)Remove and discard the four captive washer lock nuts (26) and the four bolts (24). Remove coupling unit.
(6)Remove the two spacers (23).

C. Preparation of Replacement Component (Ref. Fig. 402)
(1)Make certain replacement component is clean and free of dents or corrosion.
(2)Check conditions of gimbal joints (28), (30) of linkage and make certain they are lubricated (Ref. Material No. 04-004).
(Ref. Fig. 402)

D. Installation (Ref. Fig. 402)
(1)Install the two spacers (23)
(2)Position coupling unit on its support. Insert the four bolts (24) and the four captive washers lock nuts (26) ; tighten nuts, being careful to position spacers correctly.
(3)Check correct bonding.
(4)Install cables in coupling unit pulleys.
(5)Make certain that cables are not crossed and are in correct position.
(6)Install the seven cable guides (27). Tighten using torque wrench. TORQUE to 0.25 m.daN (22 lbf. in.).
(7)Install rigid transmission shaft (29).
(a)Insert gimbal joint output end (28) on splined shaft of coupling unit (35). Check that rubber belows end covers splined shaft tip undercut.
(b)Position rigid transmission shaft cover against that of actuator. Install the four bolts (31) with washers (32) and nuts (33) ; tighten and safety with cotter pins (34).
(8)Connectelectrical plug (25).
(9)Tighten turnbuckles (16), (18), remove cable clamps and continue to tighten gently and progressively until cable friction is no longer present.
(10)Tighten turnbuckles (6), (8), remove cable clamps and continue to tighten gently and progressively to allow tension regulator locking tool (4) to be extracted.
(11)Remove locking tool (4).
(12)Adjust cable tension (Ref. 76-11-00, P. Block 501).
(a)Note ambient temperature. Adjust tension of cables (9), (10) in accordance with the fuselage cable tension graph (Ref. Fig. 403) by means of tension adjusters (6), (8). Control pointer on tension regulator scale.



AT Coupling Unit
Figure 402

(b)During and after cable tensioning, flex the two cables equally and release rapidly. During cable deflection check that turnbuckle element of tension regulator slides freely.

(c)Note ambient temperature. Adjust tension of cables (14), (15) in accordance with the wing tension cable graph (Ref. Fig. 404) by means of tension adjusters (16), (18) according to tensiometer. Balance tension on both cable runs.

(d)Check that rigging pins (1), (12) and (21) slide freely.

(13)Remove rigging pins (1), (12) and (21).

(14)Install safety pins (5), (7), (17), (19) on turnbuckles (6), (8), (16) and (18).

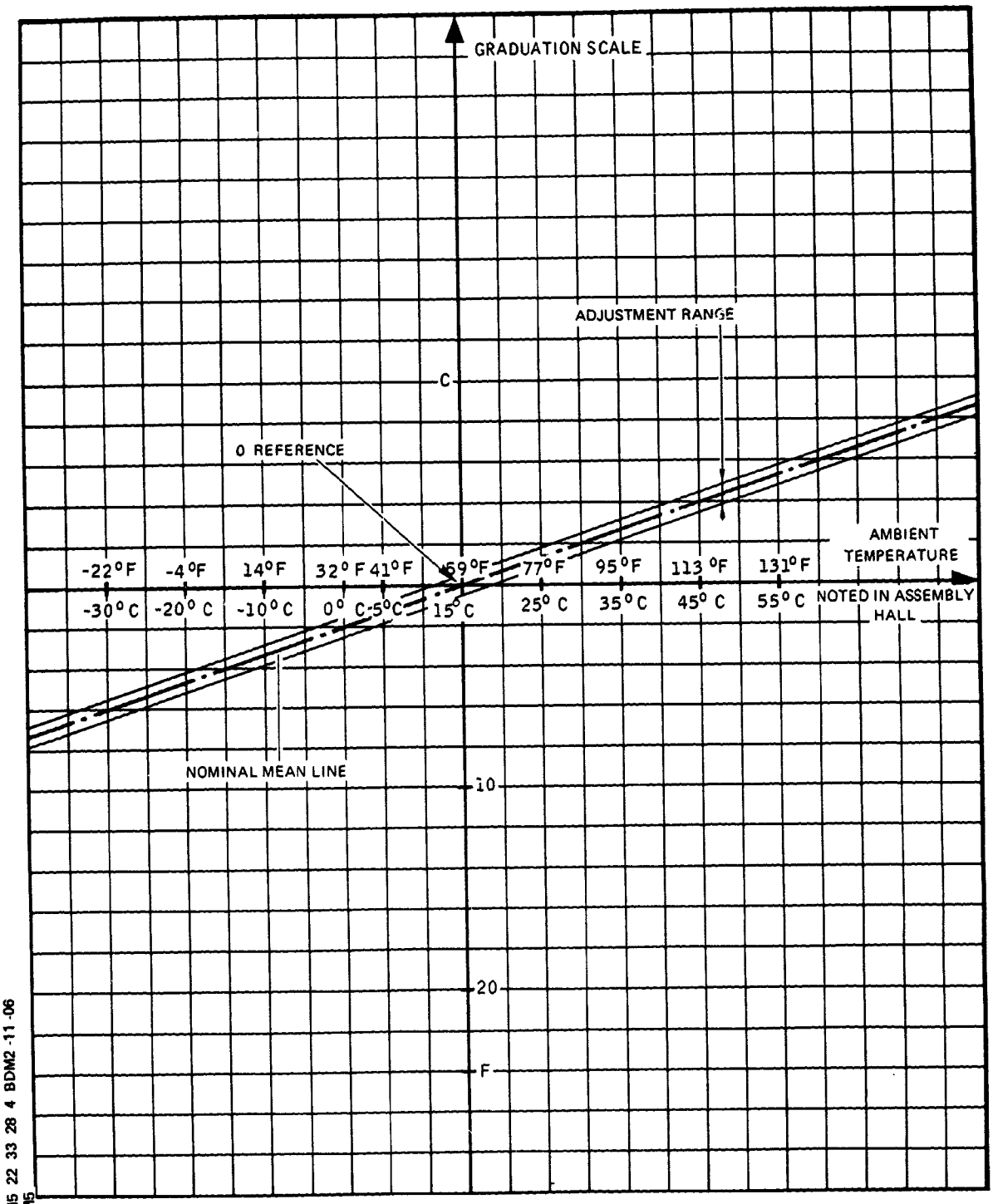
(15)Remove warning notices (throttle control and flap and slat control levers).

(16)Move throttle control lever several complete cycles ; check that there is no friction point, then bring throttle control lever back to idle position.

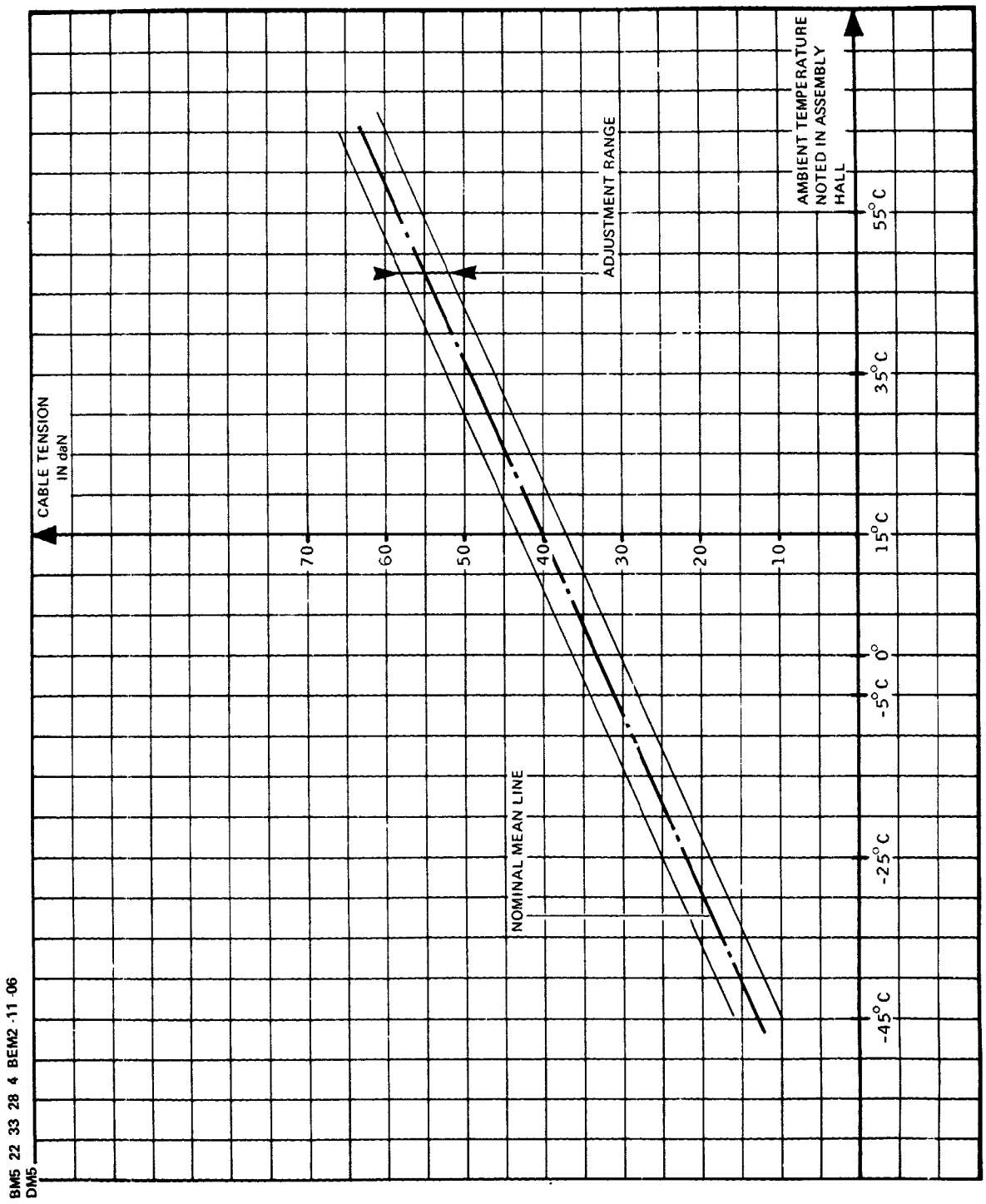
(17)Remove safety clips and tags and close circuit breakers 306CJ1 and 306CJ2.

(Ref. Fig. 403)

(Ref. Fig. 404)



Fuselage Section - Cable Tension Adjustment Graph
Figure 403



Wing Section - Cable Tension Adjustment Graph
Figure 404

E. Test
(1)For left AT coupling unit 3CJ1
Carry out an autothrottle operational test using left throttle control lever (Ref. 22-30-00, P. Block 501, paragraph 1.B.).
(2)For right AT coupling unit 3CJ2
Carry out an autothrottle operational test using right throttle control lever (Ref. 22-30-00, P. Block 501, paragraph 1.B.).

F. Close-Up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Close access door 414GR on left engine pylon (424GR on right engine pylon).
(3)Install sidewall panels 131SW, 132SW and 131RW in forward cargo compartment and close FWD cargo compartment door.
(4)Remove access platforms.

R AT COUPLING UNIT 3CJ1 (3CJ2) - REMOVAL/INSTALLATION

NOTE : Removal/Installation procedures for left and right AT coupling units being identical, only procedure concerning left coupling unit 3CJ1 is described.

1. Equipment and Materials

ITEM	DESIGNATION
A. 98A27901005000	Pin-Rigging, Cable Tension Regulator
B. 98A76101003000	Pin-Rigging, Autothrottle Coupling Unit
C. 98A76103000000	Bar-Locking, Throttle Control, Cable Tension Regulator
D.	Warning Notice prohibiting operation of throttle control lever
E. Material No. 04-004	Common Greases (Ref. 20-31-00)
F.	Environment Thermometer
G.	Circuit Breaker Safety Clips
H.	Cotter Pins
J.	Access Platform 2.70 m (9 ft.)
K.	Cable Clamps
L.	Bolts and Captive Washer Locknuts
M.	Torque Wrench
Referenced Procedures	
- 22-30-00, P. Block 501	Autothrottle
- 76-11-00, P. Block 501	Throttle Controls

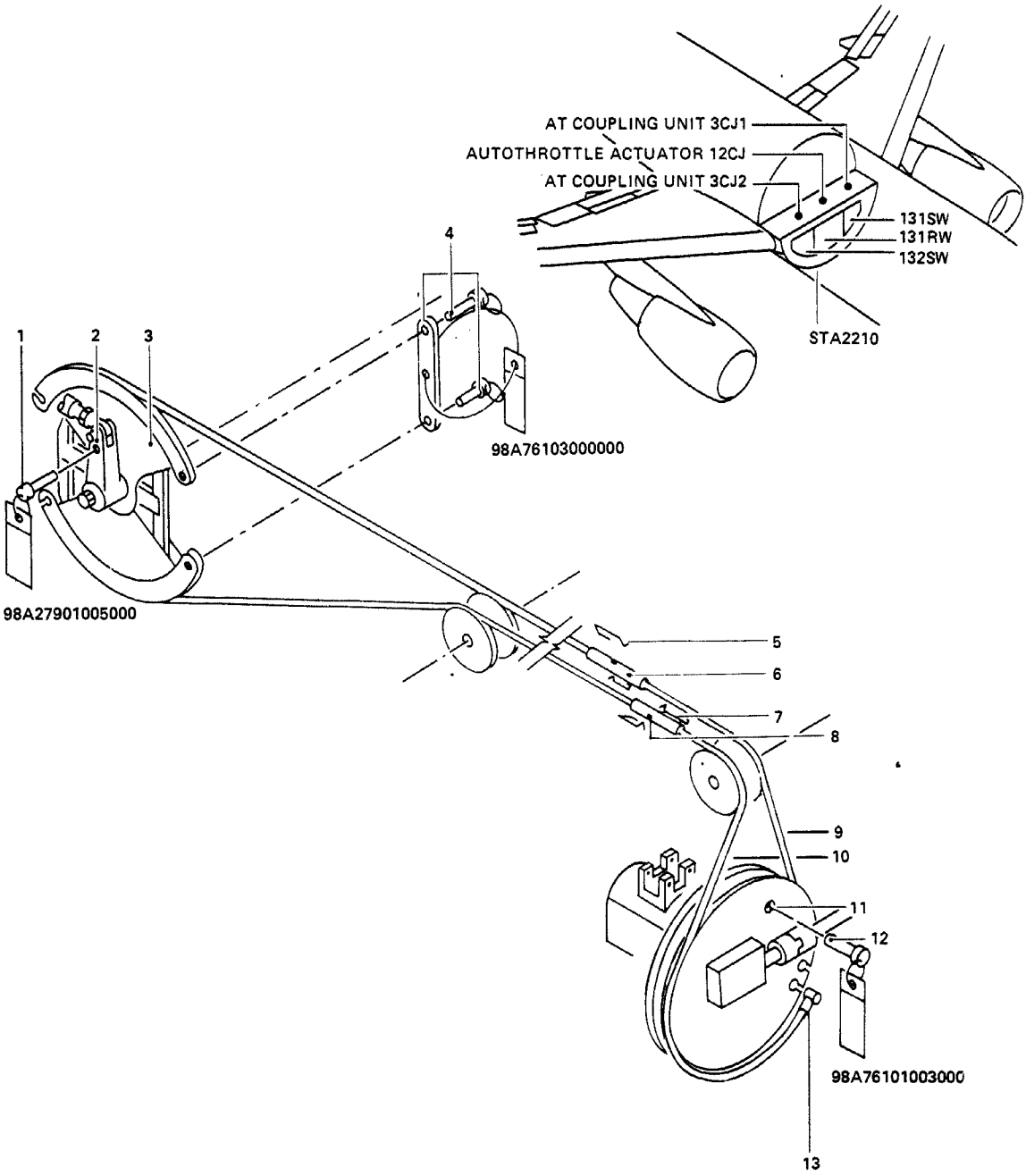
2. Procedure (Ref. Fig. 401)

A. Job Set-Up

(1)Open, safety and tag the following circuit breaker :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS-TCC SYS1/28VDC	306CJ1	102/H 8

(2)Make certain that throttle control levers are in 0° position.
(3)On mechanical center pedestal 9VU, position warning notice prohibiting operation of throttle control levers.
(4)Position access platform at FWD cargo compartment door.
(5)Open FWD cargo compartment door.
(6)In forward cargo compartment at STA2210, remove sidewall panels 131SW, 132SW and 131RW.
(7)Immobilize tension regulator (3) with rigging pin 98A27901005000 (1).
(8)Immobilize coupling unit pulley (11) with rigging pin 98A76101003000 (12).
(9)Extract safety pins (5) and (7).
(10)Install clamps on cables bearing against structure in order to keep cables tensioned upstream of coupling unit during removal of cables from coupling unit.
(11)Loosen turnbuckles (6) and (8) by the same amount until cable tension regulator locking bar 98A76103000000 (4) can be installed. (Ref. Fig. 401)



BM5 22 33 28 4 CAM2-22

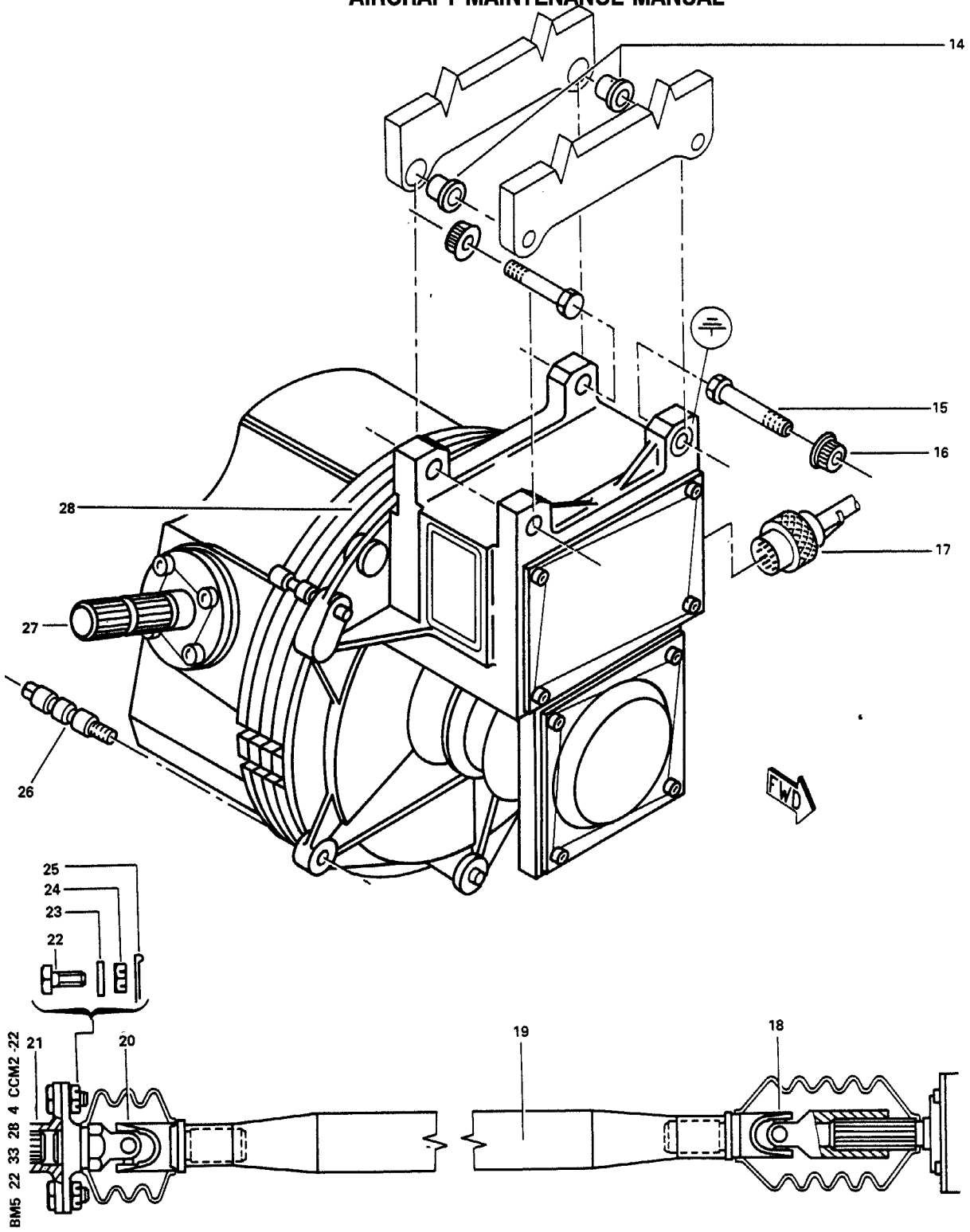
Throttle Control Linkage
Figure 401

B. Removal (Ref. Fig. 402)
(1)Disconnect electrical plug (17).
(2)Uncouple rigid transmission shaft (19) as follow :
(a)Remove and discard the four cotter pins (25), remove and retain the four nuts (24), the four washers (23) and the four bolts (22).
(b)Remove rigid transmission shaft (19) from coupling unit (27) (take care not to damage rubber bellows covering gimbal joints).
(3)Remove the seven cable guides (26) from coupling unit.
(4)Loosen turnbuckles (6) and (8) by the same amount and remove cable from coupling unit pulley (28) by rotating barrels (13) a quarter turn.
(5)Remove and discard the four captive washer locknuts (16) and the four bolts (15). Remove coupling unit.
(6)Remove the two spacers (14).

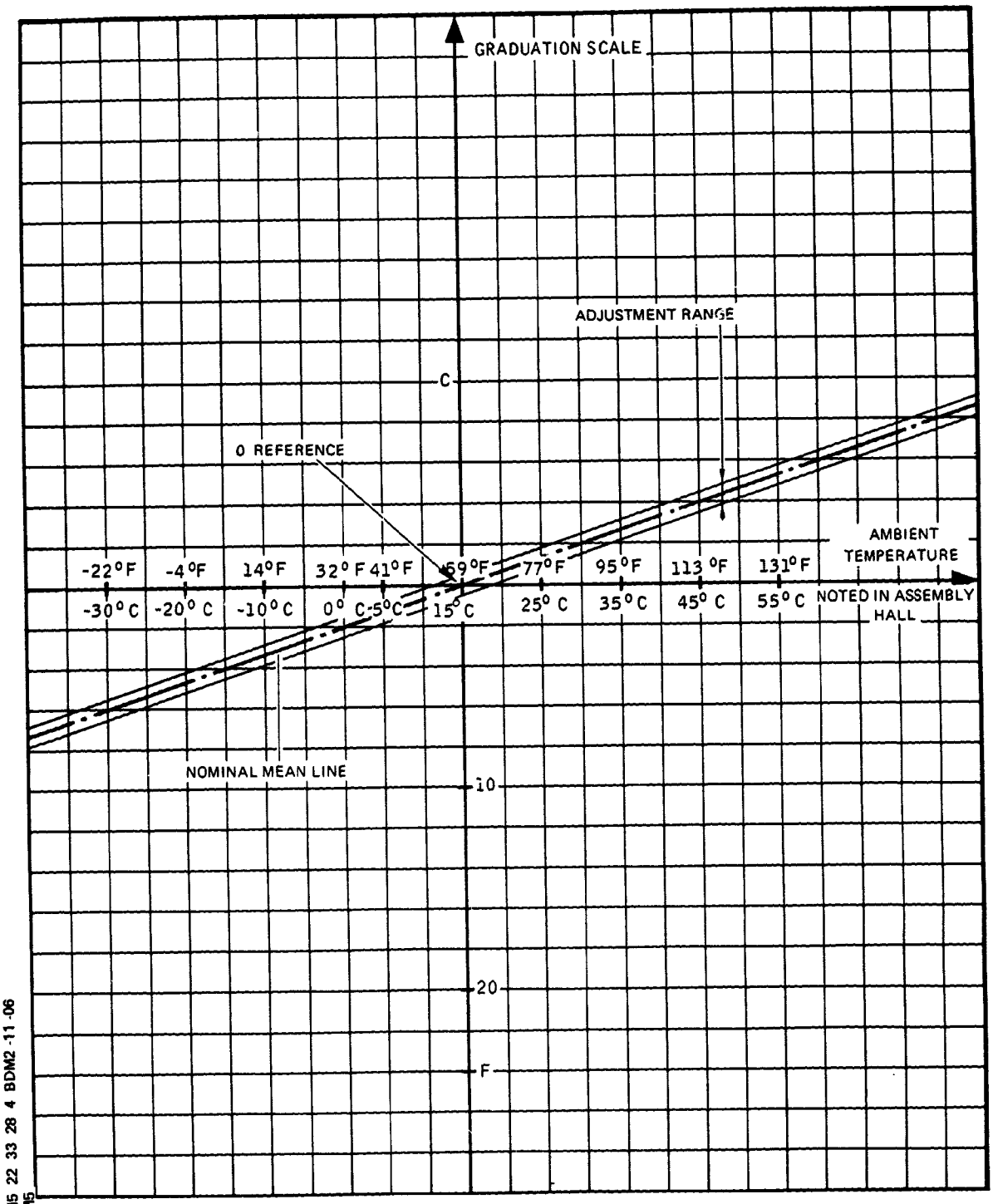
C. Preparation of Replacement Component (Ref. Fig. 402)
(1)Make certain that replacement component is clean and free of dents or corrosion.
(2)Check conditions of gimbal joints (18) and(20) of linkage and make certain they are lubricated (Ref. Material No. 04-004).

D. Installation (Ref. Fig. 401, 402)
(1)Install the two spacers (14).
(2)Position coupling unit on its support. Insert the four bolts (15) and the four captive washer locknuts (16) and tighten them, being careful to position spacers correctly.
(3)Check correct bonding.
(4)Install cable in corresponding coupling unit pulley (28).
(5)Install the seven cable guides (26). Tighten using torque wrench. TORQUE to 0.25 m.daN (22 lbf. in.).
(6)Install rigid transmission shaft (19) as follow :
(a)Insert gimbal joint output end (18) on splined shaft of coupling unit (27). Check that rubber belows end covers splined shaft tip undercut.
(b)Position rigid transmission shaft cover against that of actuator (21). Install the four bolts (22) with washers (23) and nuts (24) ; tighten and safety with cotter pins (25).
(7)Connect electrical plug (17).
(8)Tighten turnbuckles (6) and (8), remove cable clamps and continue to tighten gently and progressively to allow tension regulator locking tool (4) to be extracted.
(9)Remove locking tool 98A76103000000 (4).
(10)Adjust cable tension (Ref. 76-11-00, P. Block 501).
(a)Note ambient temperature. Adjust tension of cables (9) and (10) in accordance with the fuselage cable tension graph (Ref. Fig. 403) by means of tension adjusters (6) and (8). Control pointer on tension regulator scale.
(b)During and after cable tensioning, flex the two cables equally and release rapidly. During cable deflection check that turnbuckle element of tension regulator slides freely.
(c)Check that rigging pins (1) and (12) slide freely.
(11)Remove rigging pins 98A27901005000 (1) and 98A76101003000 (12).
(12)Install safety pins (5) and (7) on turnbuckles (6) and (8).

(13)Remove warning notices on throttle control levers.
(14)Move throttle control lever several complete cycles ; check that there is no friction point, then bring throttle control lever back to idle position.
(15)Remove safety clip and tag and close circuit breaker 306CJ1.
(Ref. Fig. 402)
(Ref. Fig. 403)



At Coupling Unit
Figure 402



Fuselage Section - Cable Tension Adjustment Graph
Figure 403

E. Test

(1)Carry out autothrottle operational test (Ref. 22-30-00, P. Block 501, para. 1.B.) using left or right throttle control lever for coupling unit 3CJ1 or 3 CJ2.

F. Close-Up

(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Install sidewall panels 131SW, 132SW and 131RW in forward cargo compartment.
(3)Close FWD cargo compartment door.
(4)Remove access platform.

1. Interface with Peripheral Systems

TCC ARINC Input Signals

Data together with their characteristics are given in figure for :

TRP
FCU
FCC1(2)
FAC1(2)
ADC1(2)
MTP
associated R/A1(2)
FMC1(2)
PMC1(2)
opposite TCC (if fitted)
associated IRU
IRU3 (single TCC version)

TCC ARINC Output Signals

TRP
FCU
FMC1(2)
MTP
opposite TCC (if fitted)
EFIS SGU1(2)(3)
PMC1(2)
Engine 1 (2) N1 indicator
FCC1(2)

**ON A/C 004-100,

DFDAU

**ON A/C 001-003,

(Ref. Fig. 001)

**ON A/C 004-100,

(Ref. Fig. 002)

R **ON A/C 001-100,

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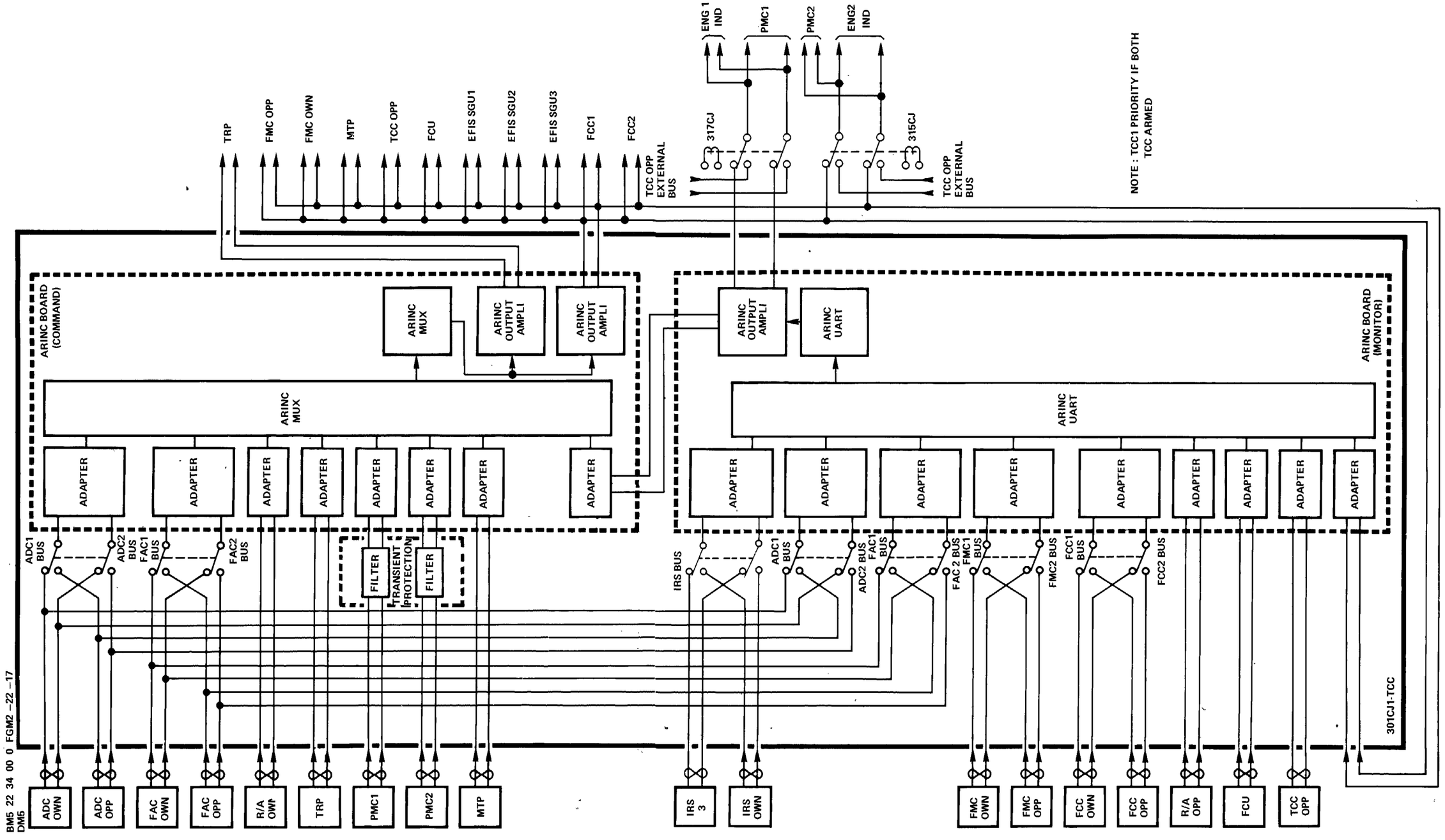
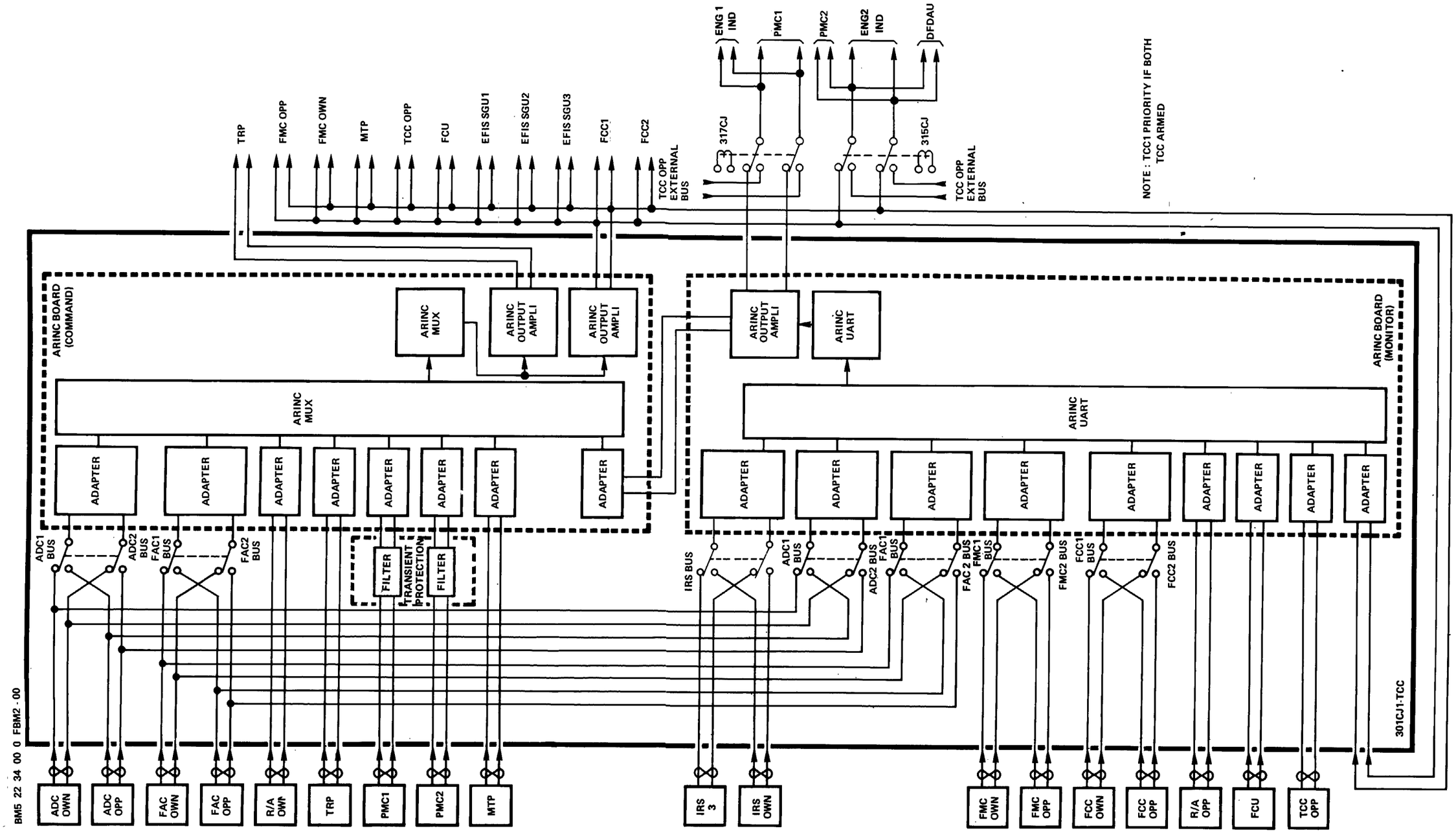


TCC - ARINC Inputs and Outputs
Figure 001



TCC - ARINC Inputs and Outputs
Figure 002

A. Air Data Computer 1(2)

Switching of one ADC to the other is automatic when the arming lever returns to the OFF position : as soon as the arming lever is in the OFF position, after detection of ADC failure, ADC2 parameters are transmitted automatically to the PMCs and TRP.

(1)ADC1 failure When the pilot operates the arming lever, the system checks for correct status of ADC1. In the event of a fault, the ADC data transmitted via the TCC bus are the labels found valid on ADC1. The faulty labels are replaced by the latest refreshed labels, i.e. ADC2 labels.

If ADC1 is found faulty and after failure detection has been checked, the system is reconfigured on ADC2 and generates all the labels using ADC2.

(2)ADC2 failure

If ADC2 failure occurs when acquisition is made by the TCC, the arming levers returns to the OFF position and the system checks for correct status of ADC1.

- If the latter operates correctly, ADC1 parameters are transmitted through the data bys and ADC1 bus selection will be confirmed when the system is re-armed by the pilot.
- If ADC1 is faulty, the system cannot be re-armed.

ARING 429 Word Characteristics - ADC to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
Altitude (ALTSTD)	203	ft.	131072	17	1	62.5		X		X	N1 limit computation
Computed airspeed (CMPSPD)	206	kt.	1024	14	0.0625	125		X		X	N1 limit computation and SPEED mode
Mach (MACH)	205	m Mach	4096	16	0.0625	125		X		X	N1 limit computation and MACH mode
Total air temperature	211	°C	512	11	0.25	500		X		X	N1 limit computation Transmission of TAT to TRP
Total pressure	242	mb	2045	16	0.031	125		X		X	N1 limit computation
Impact pressure	215	mb	512	14	0.031	125		X		X	N1 limit computation

B. Inertial Reference Unit

TCC1 is connected to IRU1 and TCC2 is connected to IRU2.

ARINC 429 Word Characteristics - IRU to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BC	CODE BN	FUNCTIONS
Body long acceleration (ACC)	331	g	4 g	12	0.001	20	X			X	Computed acceleration SPD/MACH mode (Gamma-t)
Pitch angle (PITCH)	324	Deg	180°	14	0.01	20	X			X	Computed acceleration SPD/MACH mode (Gamma-a)
Vertical acceleration (Z ACC)	364	g	4 g	12	0.001	20	X			X	Computed acceleration SPD/MACH mode (Gamma-t)
Roll angle	325	Deg	±180	14	0.01	20	X			X	Computed acceleration SPD/MACH mode (Gamma-a)

C. Radio Altimeter Transceivers

The radio altimeter signal is used to detect altitude of 30 ft. to initiate the throttle retard phase for the final flate configuration.

The TCC receives this information signal from the two radio altimeter transceivers and computed altitudes are used in the retard logic through an OR circuit.

ARINC 429 Word Characteristics - R/A to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BC	CODE BN	FUNCTIONS
Radio height (RH)	164	ft.	8192	16	0.125	50		X		X	Retract

The radio altimeter signals (400 ft.) are used to validate mode request engagement performed by the FMC.

D. Flight Management Computer 1(2)
The TCC receives the data buses from the two FMCs. The selection logic of FMC1 or FMC2 data bus depends on the FCC1 or FCC2 engagement status.

ARINC 429 Word Characteristics - FMC to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BC D	CODE BN R	FUNCTIONS
Target thrust -TRP	021	% RPM	0.3000	4	1	200		X	X		TARGET display on TRP
Target thrust -ATS	343	% RPM	256	14	0.015	200		X		X	Reference value for TCC
Target Speed	103	kt	512	11	0.25	200		X		X	Reference value for TCC
FMC DW	271			21		100					Profile status

FMC STATUS WORD LABEL 271

BIT No.	FUNCTION	BIT STATUS	
		1	0
1	label		
2			
3			
4			
5			
6			
7			
8			
9			
10	SDI		
11	PCLB Armed	Armed	Not armed
12	PDES Armed	Armed	Not armed
13	PCLB Engaged	Engaged	Not engaged
14	PALT Engaged	Engaged	Not engaged
15	PDES Engaged	Engaged	Not engaged
16	PSPD	Active	Not active
17	PMACH	Active	Not active
18	PTHR	Active	Not active
19	Throttle Idle	Idle	Not idle
20	CL	Active	Not active
21	CR	Active	Not active
22	MCT	Active	Not active
23	Vertical Mode 1		
24	Vertical Mode 2		
25	Vertical Mode 3		
26	Vertical Mode 4		
27	THR Control	Speed	Thrust
28	Altitude Alert	ON	OFF

E. Power Management Control

The TCC receives signals from PMC1 and PMC2.

ARINC 429 OUTPUTS

SIGNAL	OCTAL LABEL	SIG BITS	UNITS	UPDATE INTERVAL
N1 command	341	14	% rpm	200 ms
N1 limit (max)	342	14	% rpm	200 ms
N1 actual	346	14	% rpm	200 ms
Resolver angle	133	12	TLA °	200 ms
System discrete No. 1	270	12	-	125 ms

2. Signals Exchanged with AFS Sub-Systems and Components

A. Analog Signals

(1)Throttle position detectors

Depending on throttle position detector data the computers define the following specific positions :

(a)Idle position

This position is detected to inhibit any throttle back command in automatic flight.

The idle value corresponds to the autothrottle idle position throughout the flight phase. The idle position is active in the level change, descent or FMC IDLE phase and in the final flare configuration (RETARD) where the throttle control levers move to the idle position.

At touchdown disengagement of the A/THR mode is effective when both throttle control levers are in the idle position and when LH or RH gear microswitches are activated.

(b)17° throttle position

With the SPEED mode engaged and following a failure of the N1 limit computation the TCCS is disarmed if :

- one throttle control lever is in a position lower than 17°,
- either microswitch of the N1 indicators has tripped ;

in other words, if one engine has reached the N1 limit value while the throttle control lever is under 17°, this ensures protection against indicator microswitch failures (data not monitored).

(c)33° throttle position

This position is used in the go around mode without PMC.

When this mode is engaged the throttle control levers move at maximum speed (8°/s) until the microswitches operate.

They remain in this position as long as the N1 deviation is greater than 20% N1 ; this enables an increase in engine rating without overboosting when the levers reach the GA position. The levers then move at 3°/s when the levers have passed the 33° position.

(d)Upper limit

NOTE : Upper limit stop is provided by the N1 limit mode or in case of failure of the latter by the N1 indicator microswitches.

(Ref. Fig. 003)

(2)Dynamometric rods

Each dynamometric rod is connected to a TCC in which the alternating signal is rectified and becomes active through an hysteresis on the clutching logic.

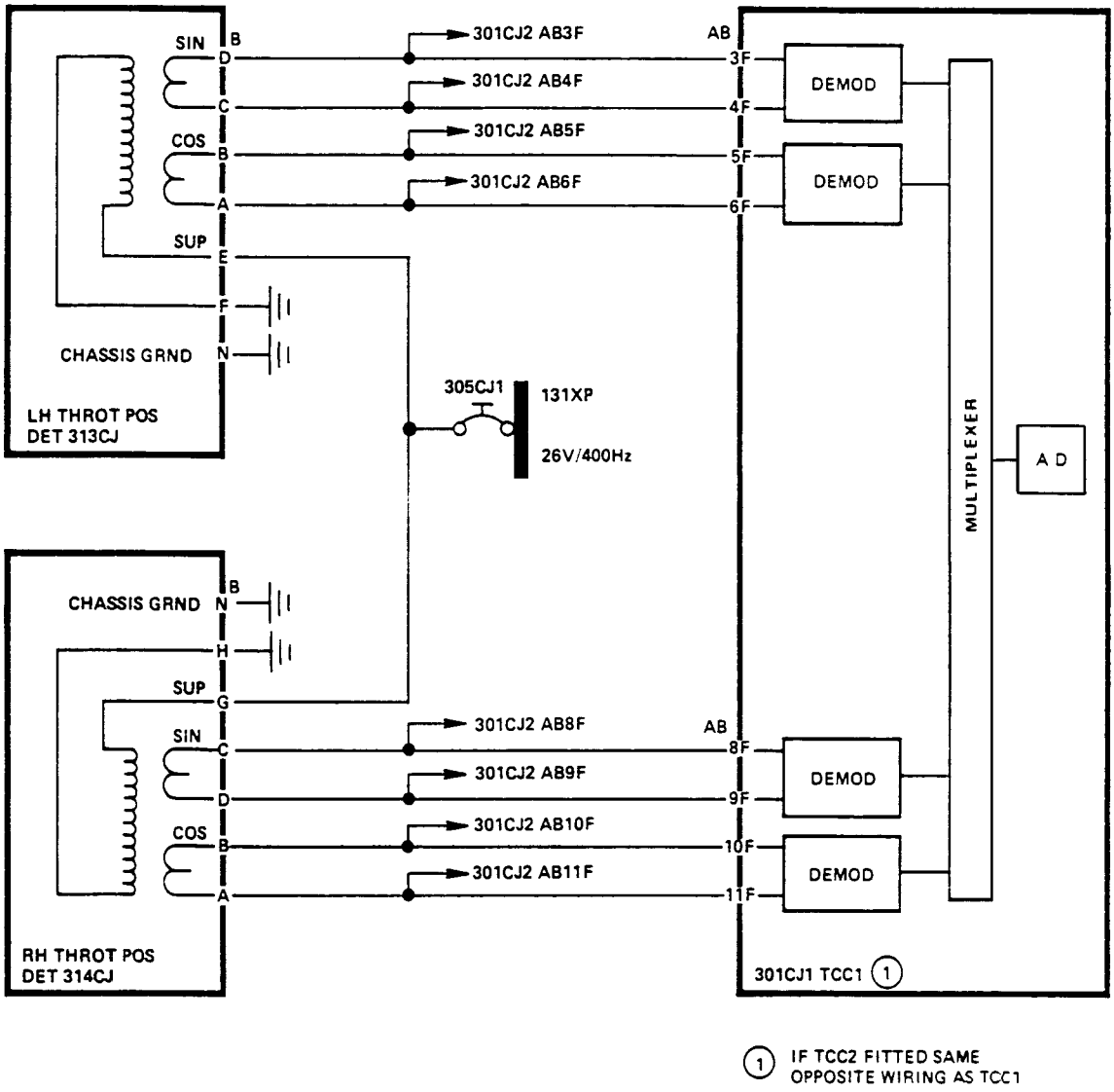
When the pilot applies a load to a throttle control lever, connection between the latter and the actuator is interrupted by the clutch during load application.

Particular case : If the level change mode is active and if the pilot applies a load to one throttle control lever during the retard phase, both levers declutch and remain declutched until another mode is selected.

(Ref. Fig. 004)

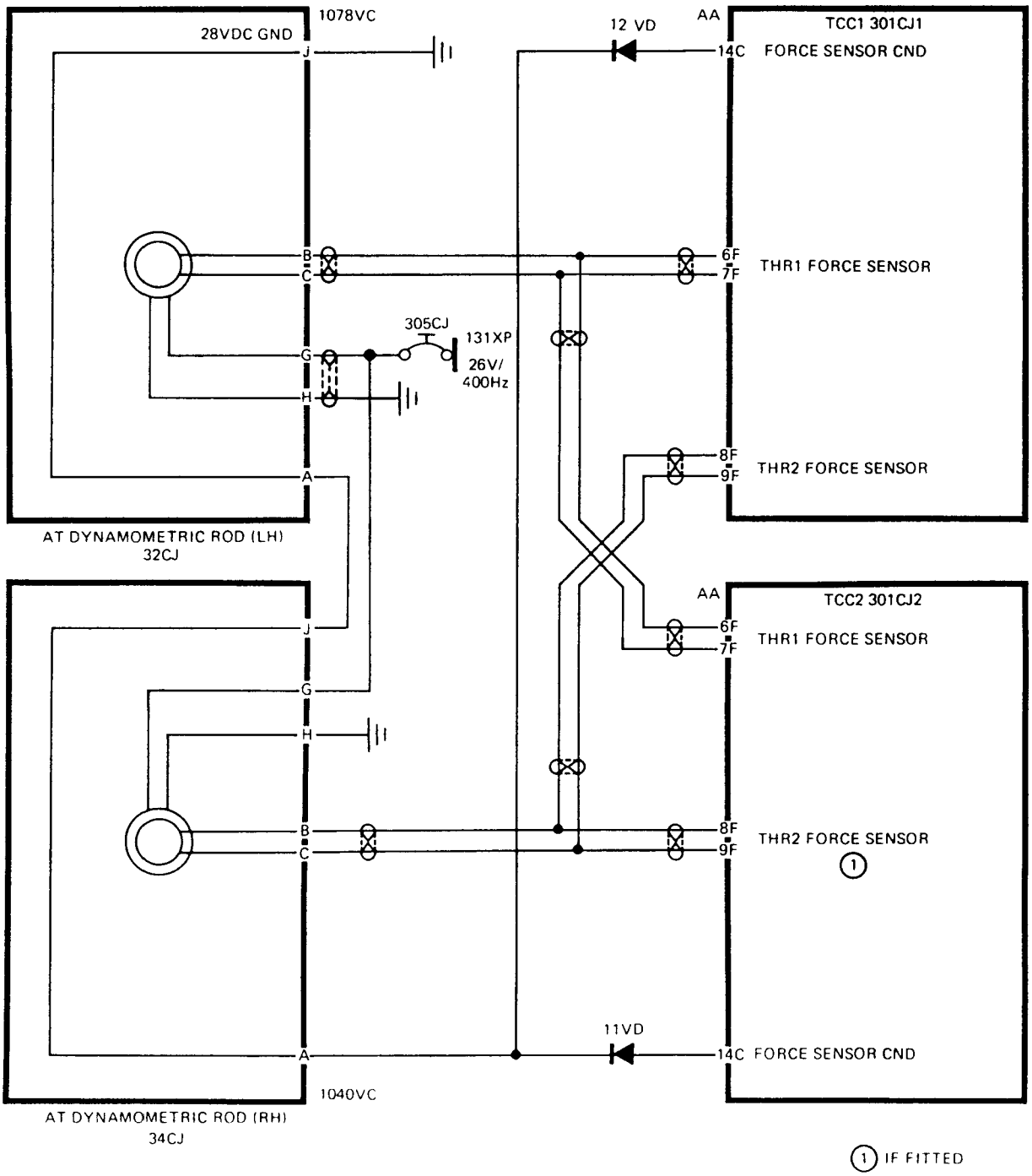
(3)Electrical connections between TCC, autothrottle and coupling units

Actuator order	H/C signal	THR increase	10V = 8°/s



BM5 22 34 00 0 APM2 01

Throttle Position Detectors - TCC Interconnection Schematic
Figure 003



BM5 22 34 00 0 ARM2 01

AT Dynamometric Rods - TCC Interconnection Shematic
Figure 004

	(7b, 8b/9B, 10B)		
Tachometer generator	H/C signal (14A/15A)	THR increase	2.14VDC/DEG/s THR

(Ref. Fig. 005)

B. Discrete Signals

(1)FAC-TCC interface via discrete signals

Each FAC provides a TCC with :

- two YAW DAMPER ENGAGE signals
- two TRIM ENGAGE signals.

These signals are used in the TCC by the data bus selection logic of FAC1 or FAC2 depending on the lever engagenent status.

FAC data bus selection logic

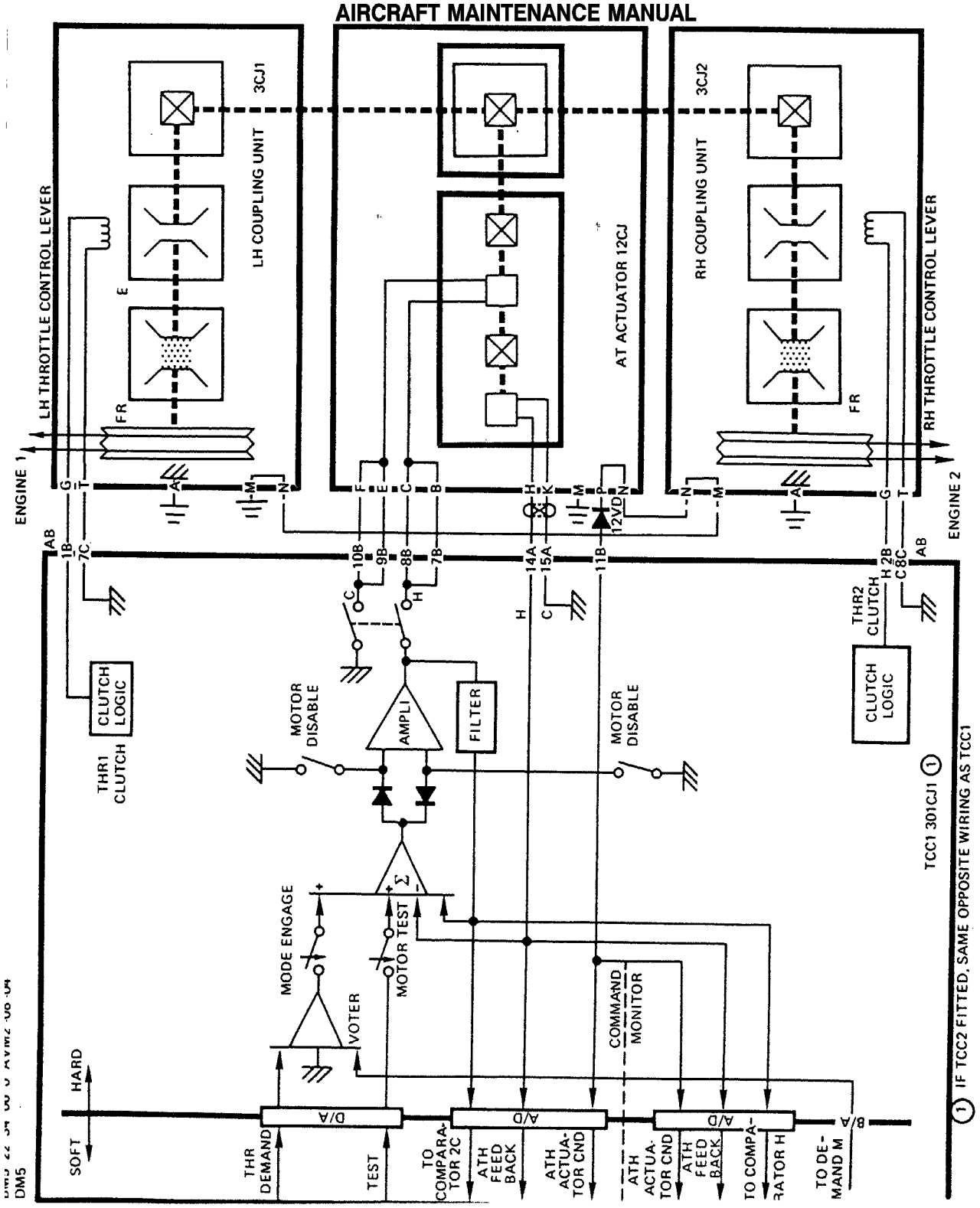
The type and characteristics of the signal are shown in the following table. Signals and data are duplicated.

	LEVEL		SIGNIFICANCE STATE 1	FUNCTION	NOTES
	1	0			
TRIM 1 ENGD	28 V	OC	TRIM 1 ENGAGED	FAC bus validity	FAC1 to TCC
YAW DAMPER 1 ENGD	28 V	OC	YAW DAMPER 1 ENGAGED	FAC bus validity	
TRIM 2 ENGD	28 V	OC	TRIM 2 ENGAGED	FAC bus validity	FAC2 to TCC
YAW DAMPER 2	28 V	OC	YAW DAMPER 2 ENGAGED	FAC bus validity	

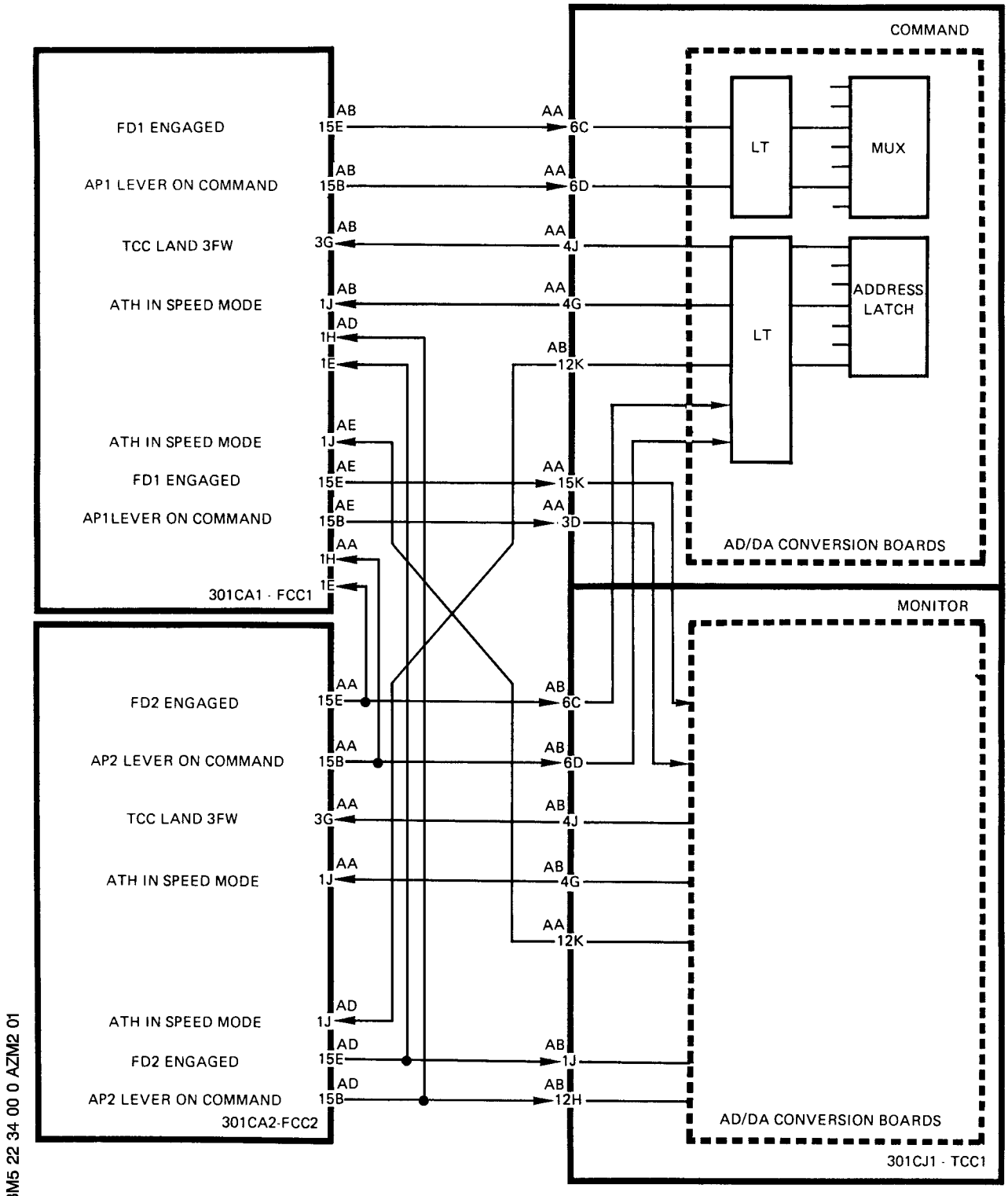
(2)TCC - FCC interface

(a)The TCC logic circuits take into account AP1 (AP2) and FD1 (FD2) engagement status which is delivered by the engage levers.
In the event of AP/FD disengagement, SPD selection is obtained automatically by pressing the A/THR pushbutton switch.

(b)The TCC sends the SPD mode engagement signal (ATS IN SPEED MODE) and the TCC failure singal (TCC LAND 3 F/W) in LAND phase to each FCC.
(Ref. Fig. 006)



TCC, AT Actuator and Coupling Units - Interconnection Schematic
Figure 005



FCC - TCC Discrete Signals - Schematic
Figure 006

Discrete Signal : TCC to FCC and FCC to TCC

	LEVEL		SIGNIFICANCE STATE 1	FUNCTION	NOTES
	1	0			
FLIGHT DIRECTOR OWN ENGD (CMD)	28 V	OC	FD1 ENGAGED	FAC bus validity logic	
A/P1 OWN CMD (CMD)	28 V	OC	FCC1 ENGAGED IN COMMAND	FCC bus validity logic	
ATH IN SPEED MODE (CMD)	28 V	OC	ATH IN SPEED MODE		
ATH IN SPEED MODE (CMD)	28 V	OC	ATH IN SPEED MODE		
FLIGHT DIRECTOR OPP ENGD (CMD)	28 V	OC	FD2 ENGAGED	FCC bus validity logic	
A/P1 LEVER ON CMD (CMD)	28 V	OC	FCC2 ENGAGED	FCC bus validity logic	
ATH IN SPEED MODE	28 V	OC	ATH IN SPEED MODE	FCC bus validity	
TCC LAND 3 FW	28 V	OC	TCC FAILURE	LAND 3 capabi- lity	
FD OWN ENG (MON)	28 V	OC	FD1 ENGAGED	FCC bus valididyt logic	
AP OWN CMD (MON)	28 V	OC	AP1 ENGAGED IN COMMAND	FCC bus validity logic	
FD OPP ENG (MON)	28 V	OC	FD2 ENGAGED	FCC bus validity logic	
AP OPP CMD (MON)	28 V	OC	AP2 ENGAGED IN COMMAND	FCC bus validity logic	
ATH IN SPEED TO OPP	28 V	OC	ATH IN SPEED MODE		

(3)Identification of engine type through aircraft wiring.
A wire on the rack connector informs the TCC of the type of engine which is installed on the aircraft.
According to this coding the TCC extracts from memory the parameters corresponding to the engine in order to generate the N1 limit value.

(4)Engine failure detection
Each flight warning computer generates a failure signal for each engine from oil pressure and N2 data.
An engine is detected faulty if N2 is lower than a given threshold and if the oil pressure drop is detected.
The flight warning computer generates a ground signal in the event of an engine failure and an open circuit signal in normal operating conditions.

(5)Wing anti-icing and air conditioning

The TCC takes into account the wing anti-ice, nacelle and air conditioning valve signals.

The valve signals are characterized as follows :

- open valve : aircraft ground
- closed valve : open circuit.

The wing anti ice control signals are taken into account for correction of the N1 limit at takeoff.

The valves provide the TCC with the following parameters :

TCC DISCRETES INPUTS BLEED CORRECTION

SIGNAL	LEVEL		SIGNIFICANCE	FUNCTION
	OPEN	CLOSED		
ENG1 BLEED	GND	O/C	PT2 PROBE HT SPLY CO, ENG1 (29 KC)	Thrust computation
ENG2 BLEED	GND	O/C	PT2 PROBE HT SPLY CO, ENG2 (28 KC)	Thrust computation
CROSS BLEED	GND	O/C	X FEED & monitoring valve opened (14HV & 16HV)	Thrust computation
AIRCOND 1	GND	O/C	PACK VALVE 1 opened (69HB)	Thrust computation
AIRCOND 2	GND	O/C	PACK VALVE 2 opened (68HB)	Thrust computation
WING ANTI ICE 1	GND	O/C	L normal wing anti ice or alternate valve open (37DM & 87DM)	Thrust computation
WING ANTI ICE 2	GND	O/C	R normal wing anti ice or alternate valve open (38DM & 88DM)	Thrust computation
WING ANTI ICE SELECTION	+28VDC	O/C	Wing anti ice function selected (P/BSW 119DM)	Thrust computation
ENG1 ANTI ICE	GND	O/C	ENG1 anti ice selected	Thrust computation
ENG2 ANTI ICE	GND	O/C	ENG2 anti ice selected	Thrust computation

TCC Code Pin Program

PINS	CODE	LEVEL	ENGINE	FUNCTION
AA10A	PIN PROG ENG FAMILY	+28VDC	GE	
AA11A	PIN PROG ENG FAMILY	GND	GE	Monitoring and
AA12A	PIN PROG ENG MODEL	+28VDC	Selection	selection of
AA13A	PIN PROG ENG MODEL	or O/C	of	data in memory
AA14A	PIN PROG ENG MODEL	(at least one+28VDC)	engine type	

Engine Failure

SIGNAL	LEVEL	SIGNIFICANCE	FUNCTION
ENGINE 1 FAILED	GND	Engine failure	THR LATCH
ENGINE 2 FAILED	GND	Information from FWC1	Logic
ENGINE 1 FAILED	GND	Engine failure	THR LATCH
ENGINE 2 FAILED	GND	Information from FWC2	Logic

(6)Instinctive disconnect pushbutton switches
Each throttle control lever is equipped with an instinctive disconnect pushbutton switch
Instinctive disconnect pushbutton switches provide the following function :
- if A/THR or N1 LATCH mode is engaged, mode is disengaged by pressing one instinctive disconnect pushbutton switch without the arming lever returning to the OFF position.

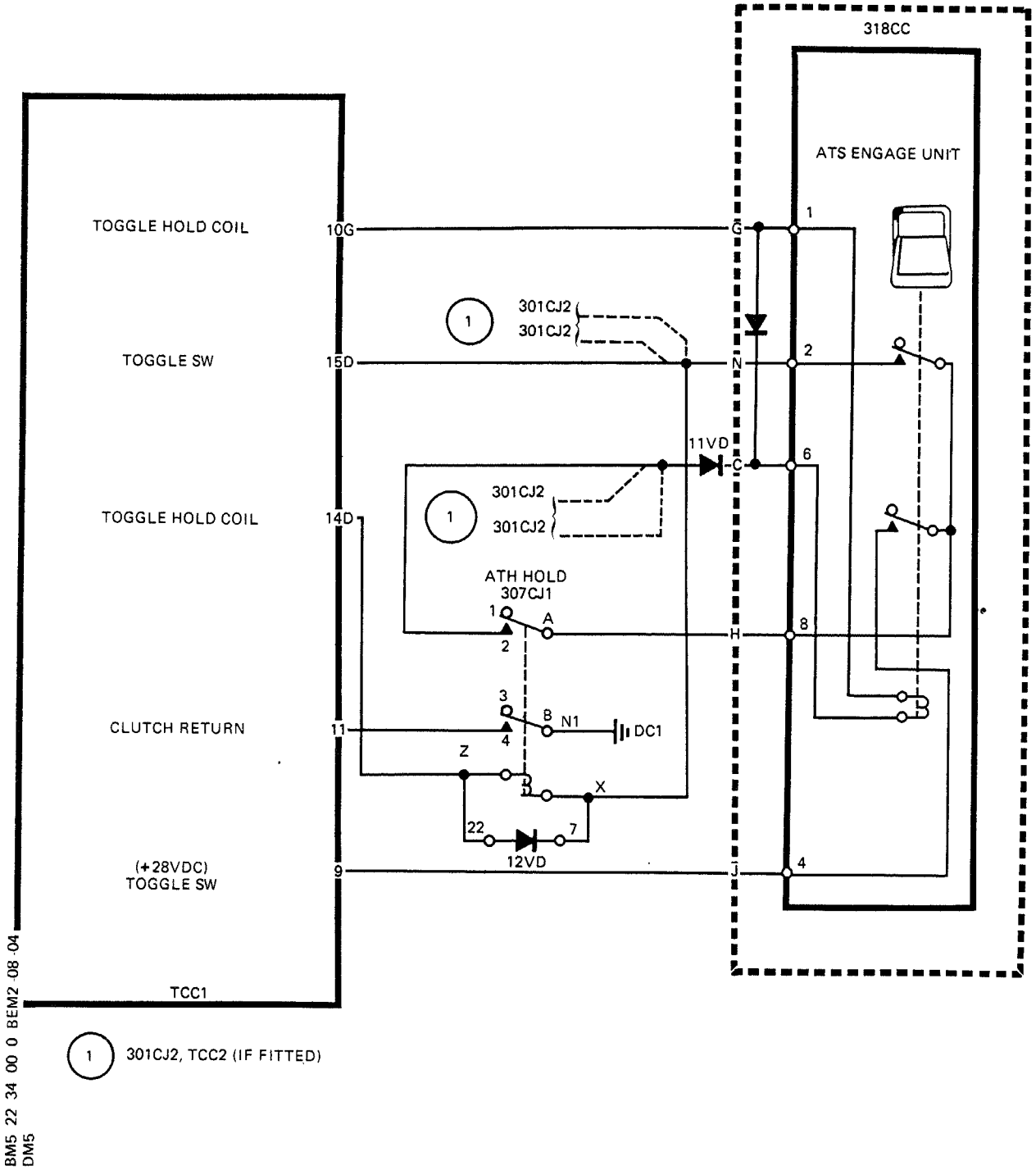
(7)TO/GA levers
Each throttle control lever is equipped with a TO/GA lever.
TO/GA levers enable engagement of A/THR mode or N1 LATCH mode if A/THR cannot be selected.
On the ground, after selection of the TO or FLX TO mode on the TRP, A/THR mode is automatically engaged by pulling one TO/GA lever.

(8)N1 indicator microswitches
In flight and in the event of N1 limit computation failure the pilot can set the bugs manually.
If in SPD mode, the engine rating exceeds the value indicated by the bug in either N1 indicator, the throttle control levers stop as long as this detection is present.

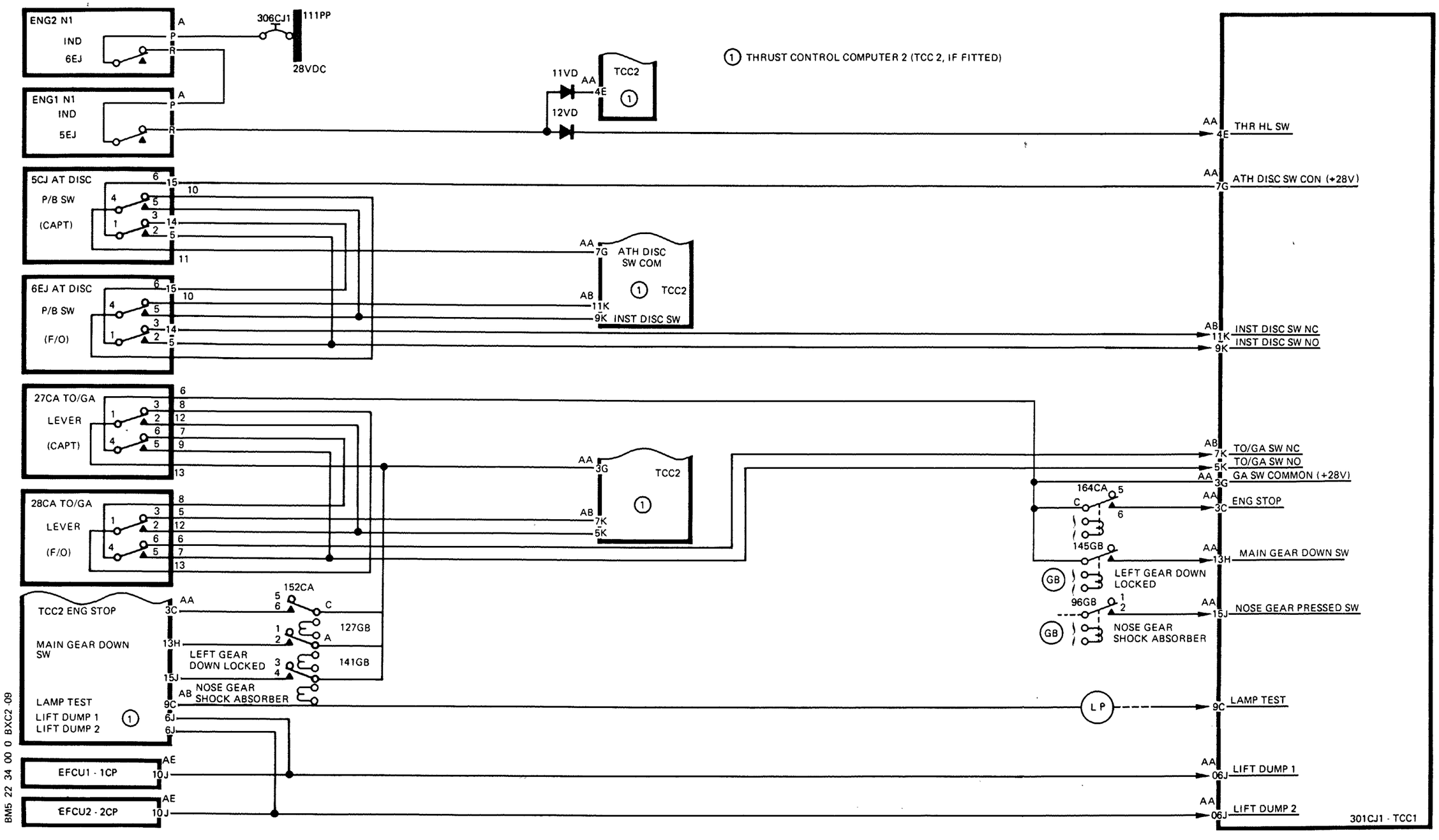
(9)FAC/ATS engage unit and TCC interface
Each TCC receives the signals required for system arming from FAC/ATS engage unit 318CC.
- TCC1 is associated with ATS1 arming lever.
- TCC2 is associated with ATS2 arming lever:

Whether the lever is held in the ON position is contingent on the command and monitoring logic validity (see arming logic).
Each self-latching relay power supply goes through the associated arming lever. System disconnection is indicated by the lever returning to the OFF position.
Both TCCs are permanently armed with priority being given to TCC1.
TCC1 disarming activates TCC2.

(Ref. Fig. 007)
(Ref. Fig. 008)



TCCS Arming - Schematic
Figure 007



TCC Discrete Inputs - Schematic
Figure 008

(10)PMC - TCC and FAC/ATS engage unit interface

(a)ENG TRIM pushbutton switch - function

The FAULT legend on the upper part of the ENG TRIM pushbutton switch (located on the face of the FAC/ATS engage unit) comes on to indicate that a PMC failure has occurred.

The pilot must reduce the engine speed and then press the ENG TRIM pushbutton switch to disengage both PMCs.

Further to this action the FAULT legend goes off and the OFF legend comes on.

(b)Signals processed by TCC

The OFF signal is used to inhibit any new failure detection after PMC disengagement thus allowing re-engagement of the TCC.

(c)Ground configuration with engines shut down

Failure detection is interrupted by the oil low pressure switches so that the FAULT legend is not continuously on when the aircraft is on the ground (the PMCs are not supplied since the engines are shut down).

(d)A test pushbutton switch located on the lateral panel provides a means of checking correct operation of the PMCs on the ground with engines stopped.

(11)Dual TCC configuration

A wire is installed to inform TCC1 of the single or dual TCC configuration.

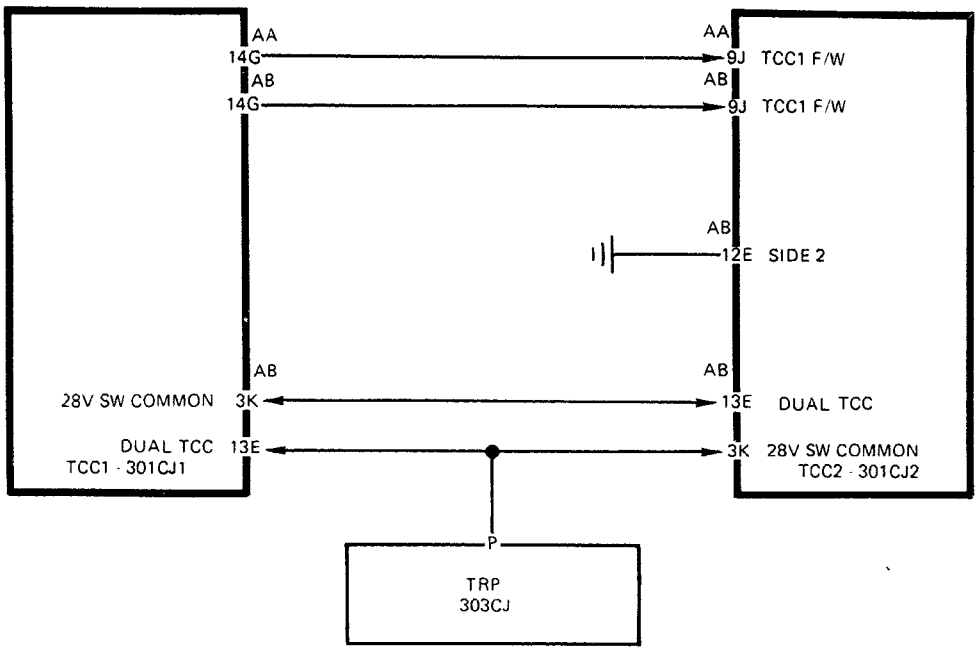
(Ref. Fig. 009).

Input Discretes (Definition)

SIGNAL	LEVEL		NOTES
	ACTIVE	NOT ACTIVE	
MAIN GEAR UP	28 V	OC	
NOSE GEAR PRESSED	28 V	OC	
ACTUATOR CONNECTED	Ground	OC	
FORCE SENSOR CONNECTED	Ground	OC	
INSTINCT. DISC NO SW	28 V	OC	
INSTINCT. DISC NC SW	OC	28 V	
LIFT DUMP 1 SW	Ground	OC	
LIFT DUMP 2 SW	Ground	OC	
TO/GA NO SW	28 V	OC	
TO/GA NC SW	OC	28 V	

SIGNAL	LEVEL		NOTES
	ACTIVE	NOT ACTIVE	
HIGH LIMIT NC SW	OC	28 V	
ATH TOGGLE SW (COM)	28 V	OC	
ATH TOGGLE SW (MON)	28 V	OC	

NOTE : TCC bus selection in computers :
TCC SELECT and TCC SEL TO OPP signals are generated by each TCC so that the other computers select TCC1 or TCC2 bus according to the state of each signal.



BM5 22 34 00 0 BJM2 08 04
DM5

TCC1 and 2 Discrete Signals - Shematic
Figure 009

C. ARINC Signals

(1)FAC - TCC, FCC - TCC interface

ARINC 429 Word Characteristics - FAC to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
Discrete data 1 (FAC DW1)	274			18							AT mode logic
Minimum airspeed (MIN SPD)	245	kt	512	13	0.0625	125		X		X	Flight envelope protection
Placard speed (MAX SPD)	207	kt	512	11	0.25	200		X		X	Flight envelope protection
Average angle-of-attack (AVERAGE AOA)	241	deg/180°	±180°	12	0.05	65.5		X		X	SPD control laws

ARINC 429 Word Characteristics - FCC to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	RESOLUTION	SIG BITS	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
AFS discretes	146	FCC DW4			18			X			SPEED or MACH automatic engagement LAND TRACK
AFS discretes	270	FCC DWS			18						"
AFS discretes	273	FCC DWS									
AFS discretes	274	DWS									

FCC - ARINC Output Signals - FCC Discretes 4

BIT No.	FUNCTION	ENCODING		NOTES
		ZERO	ONE	
1 - 8	Label 146			
9	Instinctive Disconnect			
10	Alt Alert Inhibition			FWC
11	AP CWS ENGD			
12	AP CMD ENGD			
13	FD ENGD			
14	Land Track			
15	F/D - F/W COM			
16				
17	LAND 2			For EFIS
18	LAND 3			For EFIS
19	F/D - F/W MON			
20		Not Requested	Requested	
21				
22				
23				
24				
25				
26	LAND 3 F/W			EFIS
27				
28				
29				
30	Matrix			
31	"			
32	Parity			

ARINC Output Signals (continued) - FCC Discretes 5

BIT No.	FUNCTION	ENCODING		NOTES
		ZERO	ONE	
1 - 8	Label 270			
9	IRS OPP VALIDITY		Valid	
10	LD 54			
11	G/S CPT LD COM			
12	LOC CPT COND			
13	LOC TRK LD COM			
14	LOC TRK LD MON			
15				
16				
17	FLARE COND			
18	LD 12			
19	PITCH CWS LD			
20	ROLL CWS LD	Not Requested	Requested	
21				
22	LAND 2 Capability			
23	LAND 3 Capability			
24	FCC F/W			
25	IRS OWN Validity		Valid	
26	THRUST DEMAND		Requested	
27				
28			Requested	
29	G/S CPT LD MON			
30	Matrix			
31	"			
32	Parity			

ARINC Output Signals - FCC Discretes 3

BIT No.	FUNCTION	ENCODING		NOTES
		ZERO	ONE	
1 - 8	Label 273			
9				Coded Data
10				PROFILE CLB ARM
11				15 and 24
12	ALT ACQ ARM			PROFILE DES ARM
13				15 and 25
14	NAV ARM			
15	PROFILE ARM			
16	LOC ARM			
17				
18				
19				
20	LAND ARM	Not Requested	Requested	
21				
22	GLIDE ARM			
23	VOR ARM			
24	- - CLB			
25	- - DES			
26				
27				
28				
29	V/S Limit			
30	Matrix			
31	"			
32	Parity			

ARINC Output Signals (continued) - FCC Discretes 1

BIT No.	FUNCTION	ENCODING ZERO	ENCODING ONE	NOTES
1 - 8	Label 274			
9	- - DES			Coded Data
10	- - CLB			ALT ACQ 13, 29
11	SPD			ALT 13, 22
12	PROFILE			G/S CPT 19, 29
13	ALT			G/S TRK 19, 22
14	V/S			SPD CLB 10, 11, $\overline{12}$
15	FLARE			SPD DES 9, 11, $\overline{12}$
16	PITCH GA			MACH CLB 10, 18, $\overline{12}$
17	PITCH TO			MACH DES 9, 18, $\overline{12}$
18	MACH			PROFILE ALT $\overline{9}$, $\overline{10}$, 12
19	G/S			
20		Not Requested	Requested	PROFILE CLB 12, 10, $\overline{9}$
21				
22	- - TRK			PROFILE DES 12, 9, $\overline{10}$
23				PSPD PHASE 11, 12
24	FD Bar Flash			PMACH PHASE 18, 12
25				
26				
27				
28				
29	- - CPT			
30	Matrix			
31	"			
32	Parity			

(3)FCU - TCC interface

ARINC 429 Word Characteristics - FCU to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE B C D	CODE B N R	FUNCTIONS
Selected Mach (SEL MACH)	106	mMACH	0 to 4096	12	1	62.5		X		X	MACH hold mode
Discrete word 1 (FCU DW1)	270			18		56					MACH mode, SPD mode, EPR mode, LVL CHANGE logic (see DTD FCU)
Selected airspeed (SEL SPD)	103	kt	512	11	0.25	200		X		X	SPEED hold mode

FCU Discrete Word

BIT No.	FUNCTION	LEVEL 0	LEVEL 1	NOTES
1-8	LABEL 270			
9				
10				
11				
12	FCC1 SELECT		SELECT	
13	FCC2 SELECT		"	
14				
15				
16				
17				
18				
19				
20				
21				
22				
23				
24				
25	A/THR PUSHBUTTON SWITCH	NOT PRESSED	PRESSED	
26				
27				
28				
29	DFA PUSHBUTTON SWITCH	NOT PRESSED	PRESSED	(If fitted)
30				
31				
32	PARITY			

(4)MPT - TCC interface

ARINC 429 Word Characteristics - MTP to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	SIG BITS		TRANSMIT INTERVAL (ms)	SPEED		CODE BC D	CODE BN R	FUNCTIONS
Maintenance data 1	350			21							Fault location task and LAND TEST
Maintenance data 2	351			21							"
Maintenance data 3	353			21							"

(5)TCC ARINC Output (See Figure para. 1.)

ARINC Outputs - 12 KHz Line - TCC to PMC2 and other systems.

LABEL (octal)	PARAMETER NAME	CYCLE	TRANSMIT INTERVAL (ms)	SIG BITS	UNITS	RANGE	NOTES
211	Total Air Temperature		438	11	°C	512	ADC to PMC
203	Altitude STD		62.5	17	feet	131072	"
205	Mach		125	13	mMach	4096	"
242	Total Pressure		125	11	mb	2048	"
215	Impact Pressure		125	9	mb	512	"
214	Derate Temperature		125	7	°C	99	TRP
342	Limit Thrust		125	14	%RPM	256	"
343	Target Thrust		125	14	%RPM	256	"
021 BCD	Nominal Thrust		125	4	%RPM	256	"
302	Limit Thrust Test		125	14	%RPM	104.8	"
55	Actual Thrust-PMC1		62.5	14	%RPM	256	TCC/M
52	Actual Thrust-PMC2		62.5	14	%RPM	256	"
305	Thrust Error		125	15	%RPM	150	TCC/M
306	Thrust Control Signal		62.5	15	TLA	128	"
51	CMD Thrust-PMC1		125	14	%RPM	256	TCC/M
54	CMD Thrust-PMC2		125	14	%RPM	256	"
307	Radio Height		125	16	feet	8192	"
103	Target Speed (FMC)		125	11	Kt	512	"
164	Target Height		125	16	feet	8192	
174	DFA speed (if fitted)		62.5	11	Kts	512	
272	TCC Discretes 2		62.5	21			
273	TCC Discretes 3		62.5	21			
274	TCC Discretes 4		62.5	21			
275	TCC Discretes 5		62.5	21			
350	Maintenance Data			21			
351	Maintenance Data		62.5	21			
353	Maintenance Data		62.5	21			
346	Actual Thrust		62.5	14	%RPM	256	Bit 9 / Bit 10

Bit 9	Bit 10
0	1 left
1	0 right

ARINC Outputs - 12 KHz - TCC to PMC1

LABEL (octal)	PARAMETER NAME	CYCLE	TRANSMIT INTERVAL (ms)	SIG BITS	UNITS	RANGE	NOTES
203	Altitude standard		62.5	17	feet		
205	Mach		62.5	13	mMach	4096	
211	Total Air Temperature		62.5	11	°C	512	
215	Impact Pressure		62.5	9	mb	512	
242	Total Pressure		62.5	11	mb	2048	
273	TCC discretes word 3		62.5	21			
341	N1 Command		62.5	14	%RPM	256	

LABEL (octal)	PARAMETER NAME	CYCLE	TRANSMIT INTERVAL (ms)	SIG BITS	UNITS	RANGE	NOTES
342	N1 limit		62.5	14	%RPM	256	
346	N1 actual		62.5	14	%RPM	256	

ARINC Outputs - 12 KHz Line (continued) - TCCDW2 Definition

BIT No.	FUNCTION	ENCODING		NOTES
		BIT = 1	BIT = 0	
1-8	Label 272			
9	INST. DISC. SW (Normally closed)	Not Active	Active	
10	TO/GA lever (Normally closed)	Not Active		
11	N1 LIM-N1 CMD < 20%	Active	Not Active	
12	Ambient Temp ≤8°	Active	Not Active	
13	N1 LIM-N1 CMD max max < 20%	Active	Not Active	
14	N1 LIM-N1 CMD R ≤0.5%	Active	Not Active	
15	N1 LIM-N1 CMD L ≤0.5%	Active	Not Active	
16	N1 LIM-N1 CMD_max max < 20%	Active	Not Active	
17	TO/GA P/BSW	Active	Not Active	3-Cycle Transmission
18	FLX TO P/BSW	Active	Not Active	"
19	CL P/BSW TRP	Active	Not Active	"
20	CR P/BSW	Active	Not Active	"
21	MCT P/BSW	Active	Not Active	"
22	AUTO P/BSW	Active	Not Active	"
23	PMC Engine Code Valid	Valid	Not Valid	
24	N1 LIM	Valid	F/W	
25	TCC Valid	Active	Not Active	
26	Dynamometric Rod	Activated	Not Activated	
27	Land 3 - Valid	Valid	F/W	
28	RA/OWN Retard	Active	Not Active	
29	TCC Select/CMD	Selected	Not Selected	
30-31	SSM			
32	Parity			

ARINC Outputs - 12 KHz Line (continued)
Label 273 - TCCDW3 Definition

BIT No.	FUNCTION	BIT STATUS 1	BIT STATUS 0
1 to 8	Label 273		
9	SDI		
10	SDI		
11	Bleed Air Correction: Air Conditioning 1	See NOTE	
12	Bleed Air Correction: Air Conditioning 2	See NOTE	
13	Bleed Air Correction: Wing Anti-Icing 1	See NOTE	
14	Bleed Air Correction: Wing Anti-Icing 2	See NOTE	
15	Bleed Air Correction: Nacelle Anti-Icing	See NOTE	
16	Engine Rating Code		
17	Engine Rating Code		
18	Engine Rating Code		
19	Fast	1	0 DFA (if
20	Slow	1	0 fitted)
21	Descent Idle	1	0
22	Weight on Wheel	1	0
23	Slats Retracted	1	0 (Slats >0°)
24	VAPP	1	0
25	FLAPS	1	0 DFA (if
26	GEAR	1	0 fitted)
27	PMC1 On-Off	On	Off
28	PMC2 On-Off	On	Off
29	ADC Comparison (C2 engine only)	ADC1 ≠ ADC2	ADC1 = ADC2
30			0
31			0
32	Parity		

NOTE : Air Cond Correction (K1)

	Bit 11	Bit 12
0	0	0
+1	0	1
+2	1	1

Wing Anti-Ice 1 Correction(K2)	Bit 13	Wing Anti-Ice 2 Correction(K3)	Bit14
0	0	0	0
+1	1	+1	1

Nacelle Anti-Ice Correction(K4)	Bit 15
0	0
+1	1

ARINC Outputs - 12 KHz Line (continued)
Label 274 - TCCDW4 Definition

BIT No.	FUNCTION	BIT STATUS 1	BIT STATUS 0
1 to 8	Label 274		
9	DFA - Armed -| (if fitted)	Armed	Not Armed
10	DFA - Active _|	Engaged	Not Engaged
11	ATS Arming Lever (OFF position)	OFF	Armed
12	MAN THR	Armed	OFF
13	ATHR Mode	Engaged	Not Engaged
14	FLAPS Position 4	Active	Not Active
15	FLAPS Position 5	Active	Not Active
16	ATS Clutches - off (Takeoff)	Off	On
17	VNAV	Engaged	Not Engaged
18	A/THR THRUST N1	Engaged	Not Engaged
19			0
20	Mode |- MACH	Engaged	Not Engaged
21	Status |_ SPEED	Engaged	Not Engaged
22	RETARD	Engaged	Not Engaged
23	Engine |- Engine 1 F/W	Failed	Not Failed
24	Fail |_ Engine 2 F/W	Failed	Not Failed
25			0
26			0
27			0
28	Throttle Pusher	Activated	Not Activated
29			0
30	O---| SSM		0
31	O---| "		0
32	Parity		

ARINC Outputs - 12 KHz Line (continued)
Definition
Label 275 - TCCDW5 Definition

BIT No.	FUNCTION	BIT STATUS 1	0
1 to 8	Label 275		
9	Target Display Code		
10	" " "		
11			
12			
13	Engine Type Code	See Below	
14			
15			
16	Limited Autothrottle Control	Limited	Not Limited
17	TO		
18	FLX TO		
19	CL — Selected		
20	CR — Thrust Limit	Engaged	Not Engaged
21	MCT — Mode		
22	GA		
23			0
24	Limit Thrust Computation F/W	F/W	No F/W
25	ADC1 Selection	1	0
	(C2 engines only)		
26	ADC2 Selection	1	0
27			0
28	Automatic Thrust Limit Mode Selection	Auto	Manual
29	Autothrottle in N1 Control	Engaged	Not Engaged
30	SSM		0
31	"		0
32	Parity		

TARGET	DISPLAY CODE
BIT No. 9	DISPLAYED 10
0	0	FLX TO
1	0	FMC (021)
0	1	LIMIT (342)
1	1	IDLE

ENGINE TYPE CODING

PINS AA	10A	11A	12A	13A	14A	
BIT No.	11	12	13	14	15	
GE	1	0	1	0	0	80C2A8

1. Interface with Peripheral Systems

- TCC ARINC Input Signals

  Data together with their characteristics are given in figure for :

  TRP
  FCU
  FCC1(2)
  FAC1(2)
  ADC1(2)
  MTP
  Associated R/A1(2)
  FMC1(2)
  EEC1(2) (channel A and B)
  opposite TCC (if fitted)

- TCC ARINC Output Signals

  TRP
  FCU
  FMC1(2)
  MTP
  opposite TCC (if fitted)
  TCC own (MON via CMD)
  EFIS SGU1(2)(3)
  EEC1(2) (channel A and B)
  FCC1(2)
  Engine 1 EPR indicator
  Engine 2 EPR indicator
R DFDAU

R (Ref. Fig. 001)

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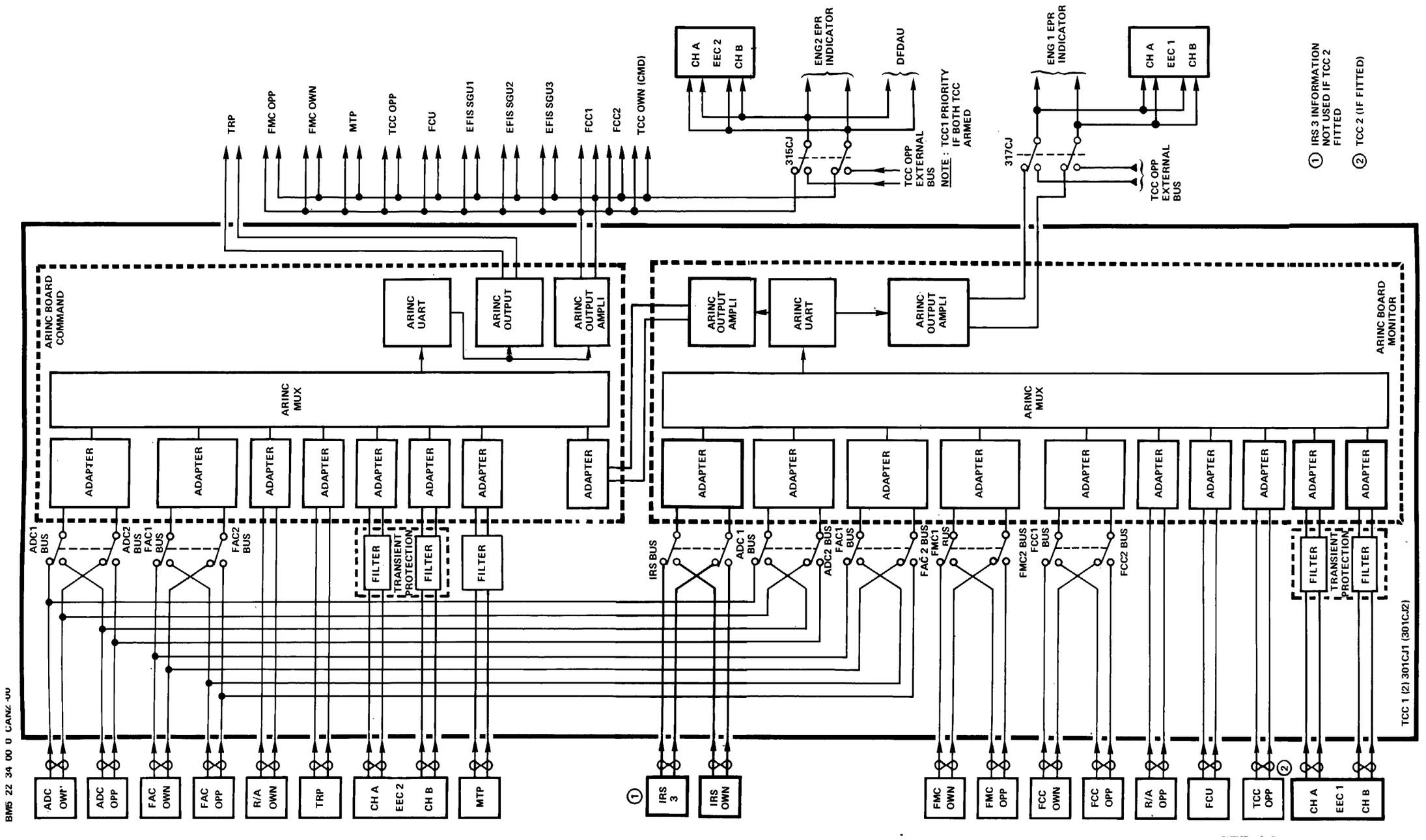


TCC - ARINC Inputs and Outputs
Figure 001

R

A. Air Data Computer 1(2)

Switching of one ADC to the other is automatic when the arming lever returns to the OFF position : as soon as the arming lever is in the OFF position, after detection of ADC failure, ADC2 parameters are transmitted automatically to the EECs and TRP.

(1)ADC1 failure

When the pilot operates the arming lever, the system checks for correct status of ADC1. In the event of a fault, the ADC data transmitted via the TCC bus are the labels found valid on ADC1. The faulty labels are replaced by the latest refreshed labels, i.e. ADC2 labels.

If ADC1 is found faulty and after failure detection has been checked, the system is reconfigured on ADC2 and generates all the labels using ADC2.

(2)ADC2 failure

If ADC2 failure occurs when acquisition is made by the TCC, the arming lever returns to the OFF position and the system checks for correct status of ADC1.

- If the latter operates correctly, ADC1 parameters are transmitted through the data bus and ADC1 bus selection will be confirmed when the system is re-armed by the pilot.
- If ADC1 is faulty, the system cannot be re-armed.

ARINC 429 Word Characteristics - ADC to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BC D	CODE BN R	FUNCTIONS
Altitude (ALTSTD)	203	ft.	131072	17	1	62.5		X		X	EPR limit computation
Computed airspeed (CMPSPD)	206	kt	1024	14	0.0625	125		X		X	EPR limit computation and SPEED mode
Mach (MACH)	205	m Mach	4096	16	0.0625	125		X		X	EPR limit computation and MACH mode
Total air temperature	211	°C	512	11	0.25	500		X		X	EPR limit computation Transmission of TAT to TRP
Total pressure	242	mb	2045	16	0.03	125		X		X	EPR limit computation

B. Inertial Reference Unit
The TCC is connected to IRU1 and IRU3.
Switching to IRU3 is performed when ATT HDG pushbutton switch on L instrument panel is pressed (in).

ARINC 429 Word Characteristics - IRU to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE NR	FUNCTIONS
Body long acceleration (ACC)	331	g	4 g	12	0.001	20	X			X	Computed acceleration SPD/MACH mode (Gamma-t)
Pitch angle (PITCH)	324	Deg	180°	14	0.01	20	X			X	Computed acceleration SPD/MACH mode Gamma-a)
Vertical acceleration (Z ACC)	364	g	4 g	12	0.001	20	X			X	Computed acceleration SPD/MACH mode (Gamma-t)
Roll angle	325	Deg	±180	14	0.01	20	X			X	Computed acceleration SPD/MACH mode Gamma-a)

C. Radio Altimeter Transceivers

The radio altimeter signal is used to detect altitude of 30 ft. to initiate the throttle retard phase for the final flare configuration.

The TCC receives this information signal from the two radio altimeter transceivers and computed altitudes are used in the retard logic through an OR circuit.

ARINC 429 Word Characteristics - R/A to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
Radio height (RH)	164	ft.	8192	16	0.125	50		X		X	Retract

The radio altimeter signals (400 ft.) are used to validate mode request engagement performed by the FMC.

D. Flight Management Computer 1(2)
The TCC receives the data buses from the two FMCs. The selection logic of FMC1 or FMC2 data bus depends on the FCC1 or FCC2 engagement status.

ARINC 429 Word Characteristics - FMC to TCC

PARAMETER NAME (LABEL)	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BC D	CODE BN R	FUNCTIONS
Target thrust -TRP	021	EPR	0-3	4	0.001	200		X	X		TARGET display on TRP
Target thrust -ATS	343	EPR	4	12	0.001	200		X		X	Reference value for TCC
Target Speed	103	kt	512	11	0.25	200		X		X	Reference value for TCC
Target Mach	106	mMach	4096	12	1	200		X		X	Reference value for TCC
FMC DW	271			21		100					Profile status

FMC STATUS WORD LABEL 271

BIT No.	FUNCTION	BIT STATUS 1	BIT STATUS 0
1 to 8	Label 271		
9 to 10	SDI		
11	PCLB Armed	Armed	Not armed
12	PDES Armed	Armed	Not armed
13	PCLB Engaged	Engaged	Not engaged
14	PALT Engaged	Engaged	Not engaged
15	PDES Engaged	Engaged	Not engaged
16	PSPD	Active	Not active
17	PMACH	Active	Not active
18	PTHR	Active	Not active
19	Throttle Idle	Idle	Not idle
20	CL	Active	Not active
21	CR	Active	Not active
22	MCT	Active	Not active
23 to 26	Not Used		

BIT No.	FUNCTION	BIT STATUS	
		1	0
27	THR Control	Speed	Thrust
28	Altitude Alert	ON	OFF

E. Electronic Engine Controls
The TCC receives the data busses from each channel of the two EECS (status word 270 to 274 and maintenance word 350 to 354).
Each EEC has two independent calculation channels one of which (channel A) has priority over the other (channel B). In the event of a failure detected on one channel, the other channel is selected automatically in the EEC.
Bus inputs connected to the EECs are protected from electrical discharges caused by lightning.

ARINC 429 word characteristics - EEC to TCC.

PARAMETER NAME	LABEL (Octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE H	CODE L
EPR Idle	301		128	14	0.001	100		X		X
EPR Actual	340		4	13	0.0005	50		X		X
EPR Command	341		4	13	0.0005	50		X		X
Max EPR LIM	342		4	13	0.0005	200		X		X

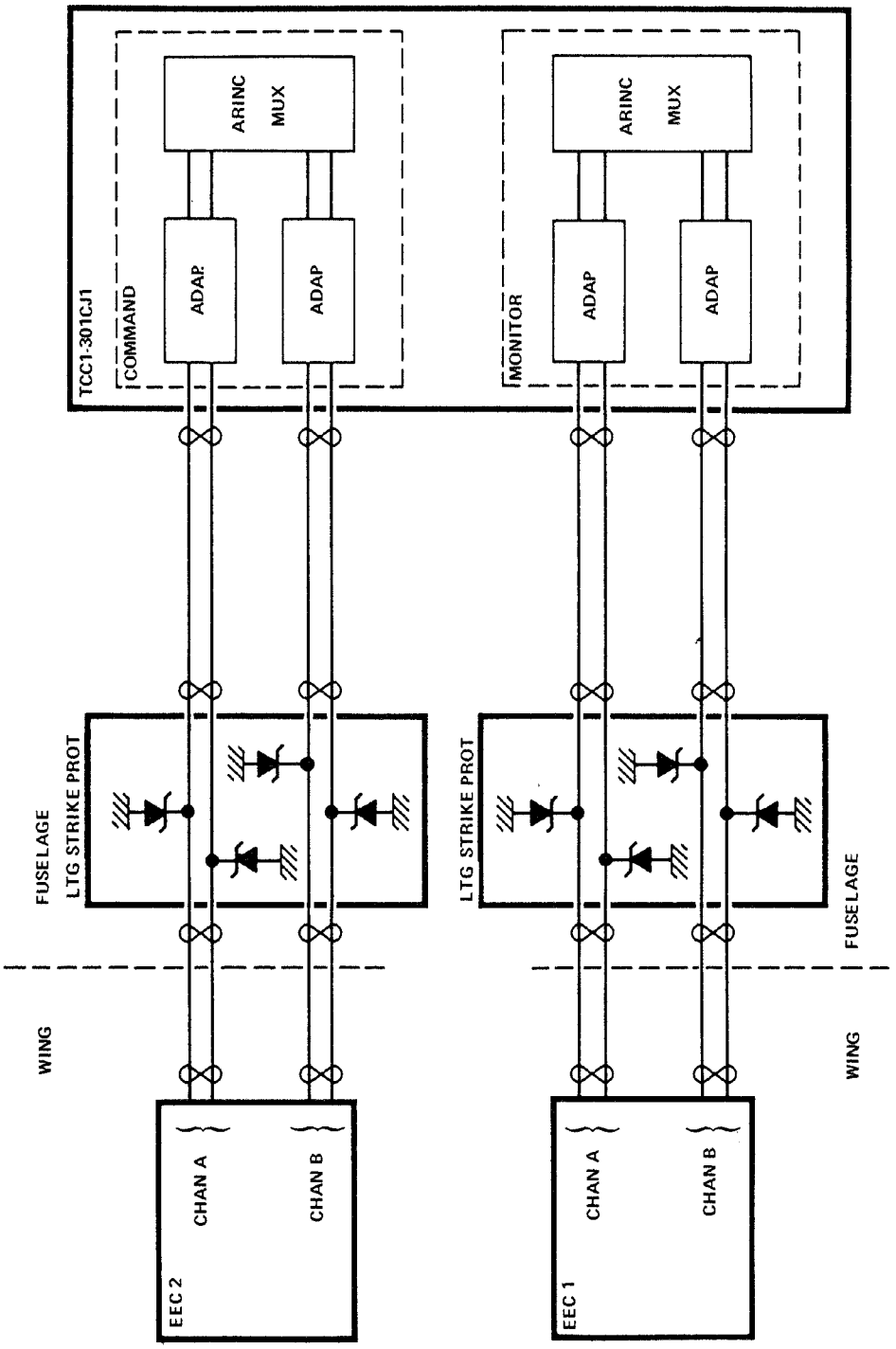
Engine Model Code
Status word LABEL 271

bit	24	23	22	21
Model 4152	0	0	1	0

(Ref. Fig. 002)

2. Signals Exchanged with AFS Sub-Systems and Components

A. Analog Signals
(1)Throttle position detectors
Depending on throttle position detector data the computers define the following specific positions :
(a)5° position (full idle)
This position is detected to inhibit any throttle back command in automatic flight.
The 5° value corresponds to the autothrottle idle position throughout the flight phase. The 5° position is active in the level change, descent or FMC IDLE phase and in the final flare configuration (RETARD) where the throttle control levers move to the 5° position.
At touchdown disengagement of the A/THR mode is effective when both throttle control levers are in the 5° position and when LH or RH gear microswitches are activated.
(b)17° throttle position
With the SPEED mode engaged and following a failure of the EPR limit computation the TCCS is disarmed if :
- one throttle control lever is in a position lower than 17°,
- either microswitch of the EPR indicators has tripped ;
in other words, if one engine has reached the EPR limit value while the throttle control lever is under 17°, this ensures protection against indicator microswitch failures (data not monitored).
(d)Upper limit



BM5 22 34 00 0 CCM2-21

EEC - TCC ARINC Interconnection Schematic
Figure 002

The upper limit stop is provided by the EPR limit mode or in case of failure of the latter by the EPR indicator microswitches.

(2)Dynamometric rods

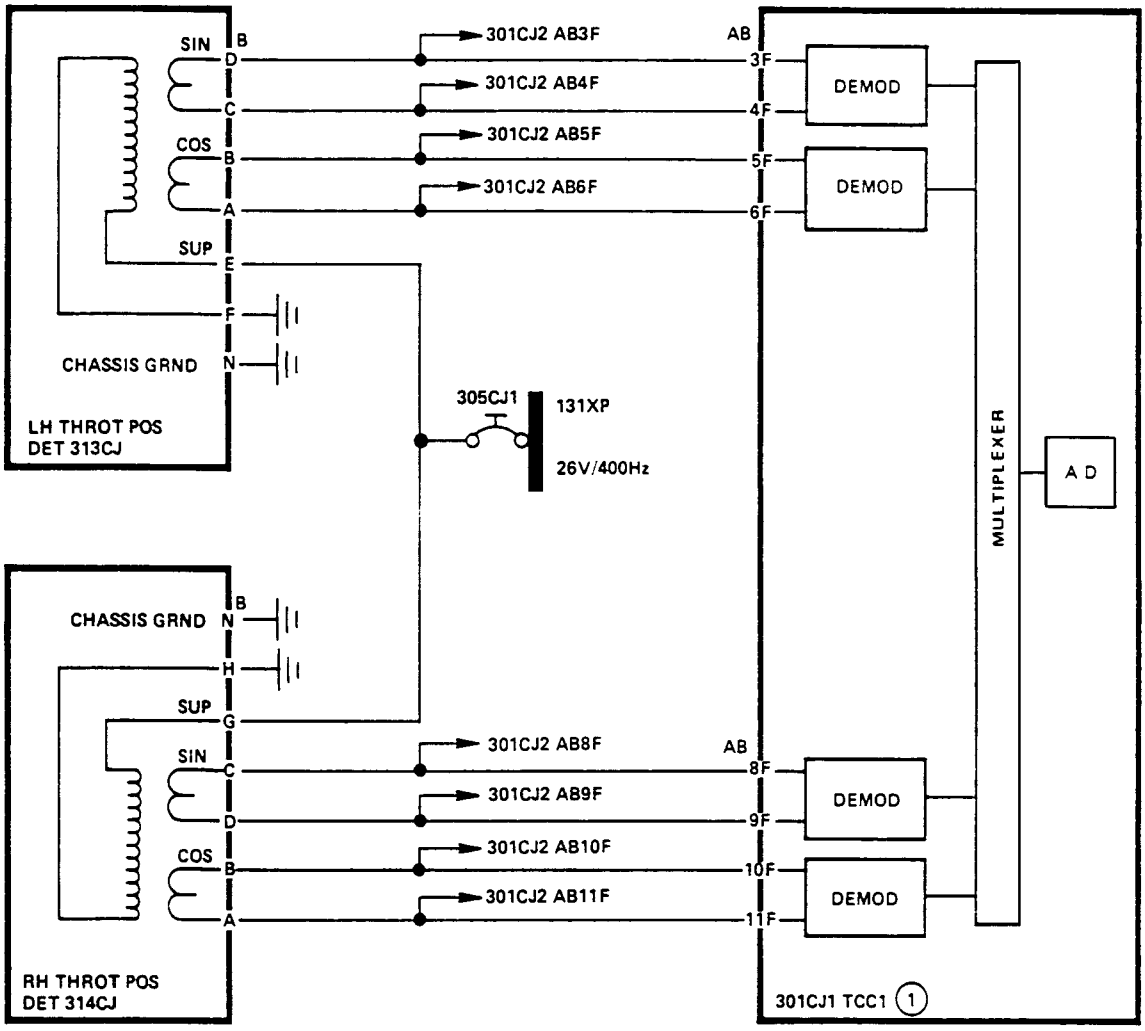
Each dynamometric rod is connected to theTCC in which the alternating signal is rectified and becomes active through an hysteresis on the clutching logic.

When the pilot applies a load to a throttle control lever, connection between the latter and the actuator is interrupted by the clutch during load application.

Particular case : If the level change mode is active and if the pilot applies a load to one throttle control lever during the retard phase, both levers declutch and remain declutched until another mode is selected.

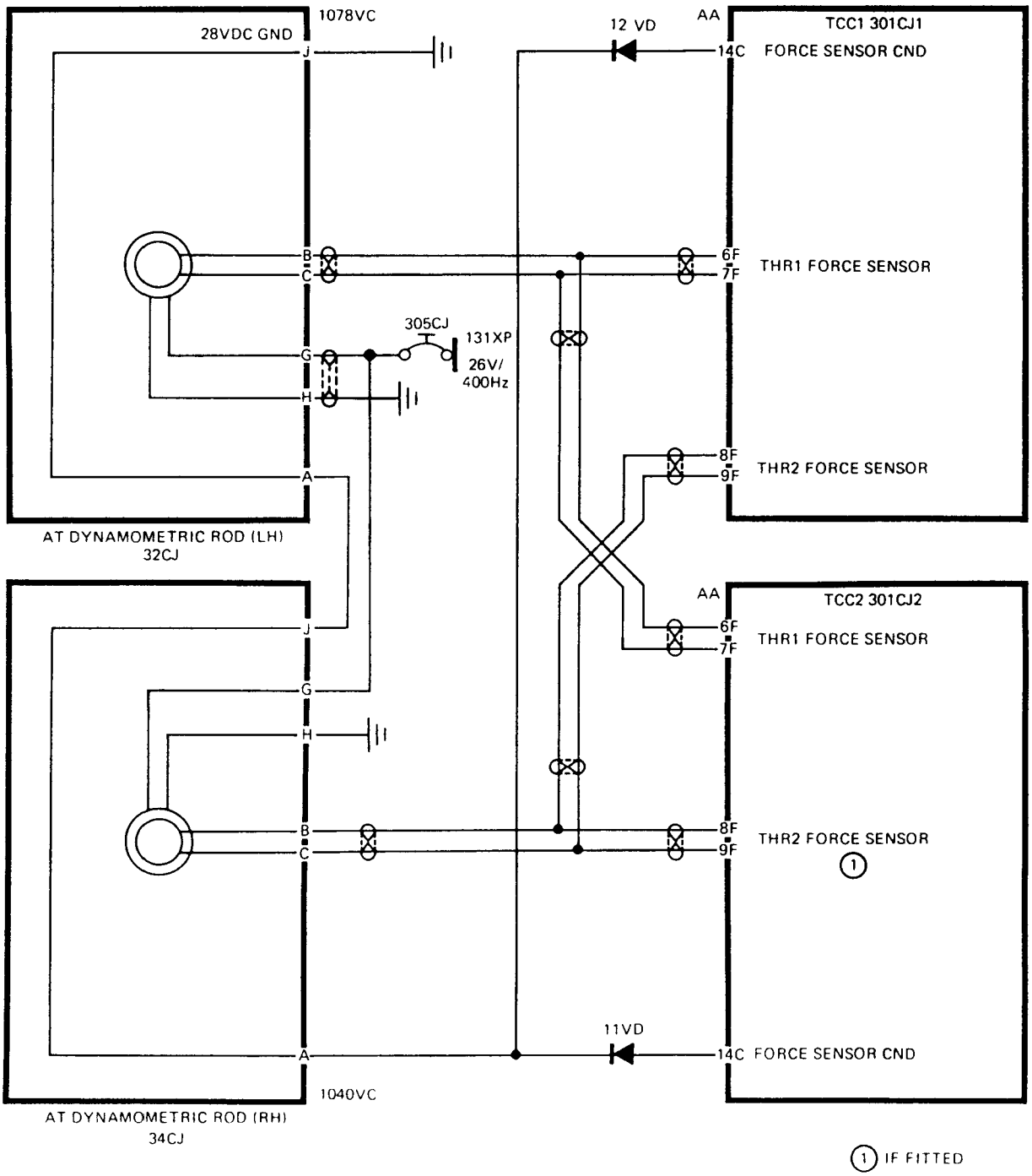
(3)Electrical connections between TCC, autothrottle actuator and coupling units.

(Ref. Fig. 003)
(Ref. Fig. 004)
(Ref. Fig. 005)



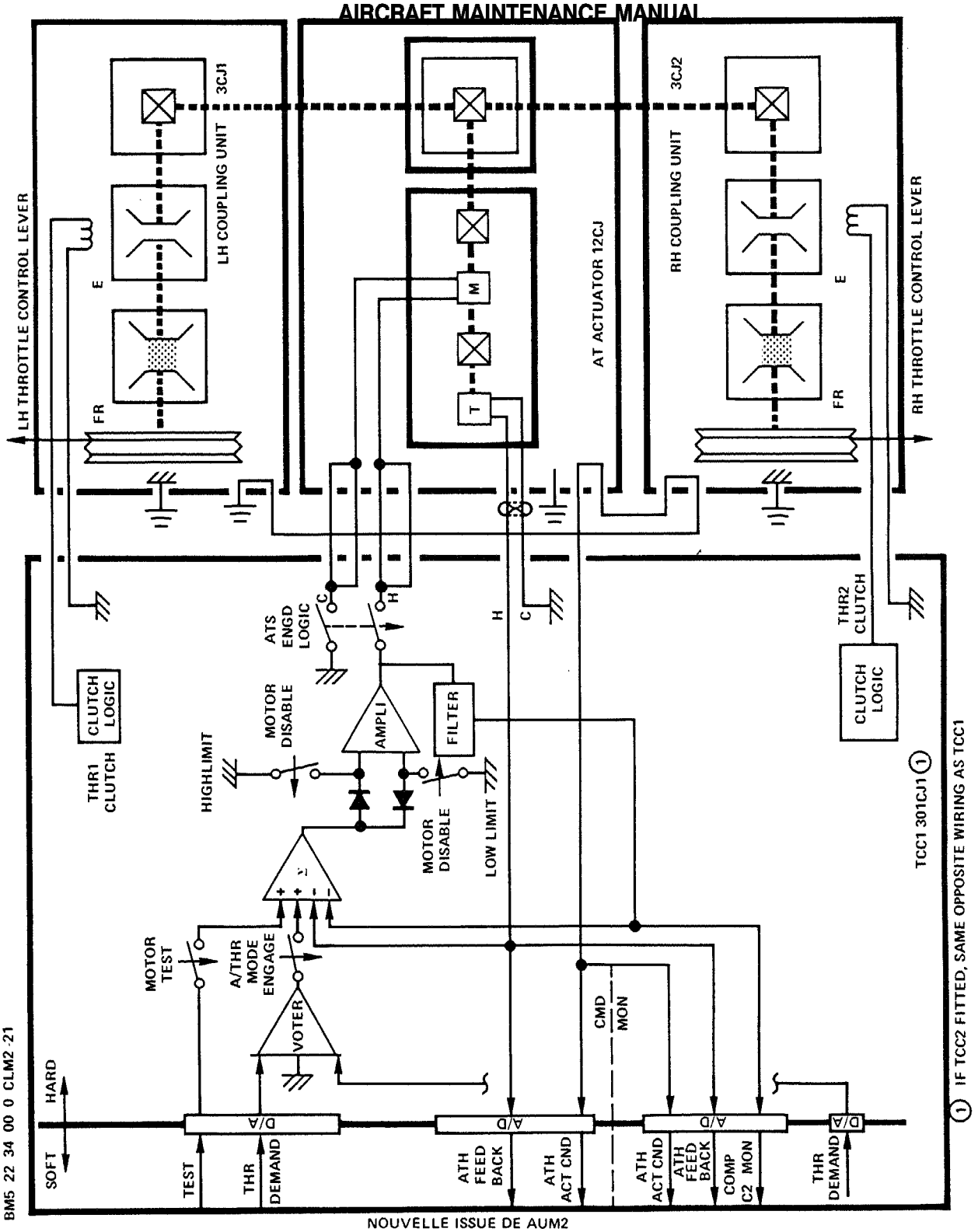
BM5 22 34 00 0 APM2 01

Throttle Position Detectors - TCC Interconnection Schematic
Figure 003



BM5 22 34 00 0 ARM2 01

AT Dynamometric Rods - TCC Interconnection Schematic
Figure 004



TCC, AT Actuator and Coupling Units - Interconnection Schematic
Figure 005

B. Discrete Signals

(1)FAC-TCC Interface via Discrete Signals

Each FAC provides a TCC with :

- two YAW DAMPER ENGAGE signals
- two TRIM ENGAGE signals.

These signals are used in the TCC by the data bus selection logic of FAC1 or FAC2 depending on the lever engagement status.

FAC data bus selection logic

The type and characteristics of the signals is shown in the following table. Signals and data are duplicated.

SIGNAL	LEVEL		SIGNIFICANCE	FUNCTION	NOTES
			STATE 1		
	1	0			
TRIM 1 ENGD	28 V	OC	TRIM 1 ENGAGED	FAC bus validity	FAC1 to TCC
YAW DAMPER 1 ENGD	28 V	OC	YAW DAMPER 1 ENGAGED	FAC bus validity	
TRIM 2 ENGD	28 V	OC	TRIM 2 ENGAGED	FAC bus validity	FAC2 to TCC
YAW DAMPER 2	28 V	OC	YAW DAMPER 2 ENGAGED	FAC bus validity	

(2)TCC - FCC Interface

The TCC logic circuits take into account AP1 (AP2) and FD1 (FD2) engagement status which is delivered by the engage levers.

In the event of AP/FD disengagement, SPD selection is obtained automatically by pressing the A/THR pushbutton switch.

The TCC sends the SPD mode engagement signal (ATS IN SPEED MODE) and the TCC failure signal (TCC LAND 3 F/W) in LAND phase to each FCC.

Discrete Signal : TCC to FCC and FCC to TCC

SIGNAL	LEVEL		SIGNIFICANCE STATE 1	FUNCTION	NOTES
	1	0			
FD1 ENGD (CMD)	28 V	OC	FD1 ENGAGE	FCC bus validity logic	
AP1 IN CMD (CMD)	28 V	OC	FCC1 ENGAGE IN COMMAND	FCC bus validity logic	
ATH IN SPEED MODE (CMD)	28 V	OC	ATH IN SPEED MODE		
ATH IN SPEED MODE (MON)	28 V	OC	ATH IN SPEED MODE		
FD2 ENGD (CMD)	28 V	OC	FD2 ENGAGE	FCC bus validity logic	
AP2 IN CMD (CMD)	28 V	OC	FCC2 ENGAGE	FCC bus validity logic	
ATH IN SPEED MODE (CMD)	28 V	OC	ATH IN SPEED MODE		
TCC LAND 3 FW	28 V	OC	TCC FAILURE	LAND 3 capability	
FD1 ENGD (MON)	28 V	OC	FD1 ENGAGE	FCC bus validity logic	
AP1 CMD (MON)	28 V	OC	AP1 ENGAGE IN COMMAND	FCC bus validity logic	
FD2 ENGD (MON)	28 V	OC	FD2 ENGAGE	FCC bus validity logic	
AP2 CMD (MON)	28 V	OC	AP2 ENGAGE IN COMMAND	FCC bus validity logic	
ATH IN SPEED MODE (MON)	28 V	OC	ATH IN SPEED MODE		

(3)Identification of engine type through aircraft wiring.
A wire on the rack connector informs the TCC of the type of engine which is installed on the aircraft.
According to this coding the TCC extracts from memory the parameters corresponding to the engine in order to generate the EPR limit value.
Coding of the engine type is defined as follows :

LABEL 275 BIT	11	12	13	14	15	
Coding	1	1	0	1	0	P.W 4152

(4)Engine failure detection
Each flight warning computer generates a failure signal for each engine.
For an engine failure detection, the two channels A and B from EEC must detect it (LABEL 270 BIT 20 : Engine running) and that the HP fuel shut-off valve control levers are in ON position.
The flight warning computer generates a ground signal in the event of an engine failure and an open circuit signal in normal operating conditions.
The TCC takes directly the engine failure information on the EEC busses for the selection logics of the engine trim condition parameters and the resolver failures detection linked with the throttle levers.
(5)Wing anti-icing and air conditioning
The TCC takes into account the wing antiice, nacelle and air conditioning valve signals.
The valve signals are characterized as follows :
- open valve : aircraft ground
- closed valve : open circuit.

TCC DISCRETES INPUTS BLEED CORRECTION

SIGNAL	LEVEL		SIGNIFICANCE	FUNCTION
	OPEN	CLOSED		
ENG1 BLEED	GND	O/C	PT2 PROBE HT SPLY CO, ENG1 (35 KC)	Thrust computation
ENG2 BLEED	GND	O/C	PT2 PROBE HT SPLY CO, ENG2 (34 KC)	Thrust computation
CROSS BLEED	GND	O/C	X FEED & monitoring valve opened (14HV & 16HV)	Thrust computation
AIRCOND 1	GND	O/C	PACK VALVE 1 opened (69HB)	Thrust computation
AIRCOND 2	GND	O/C	PACK VALVE 2 opened (68HB)	Thrust computation
WING ANTI ICE 1	GND	O/C	L normal wing anti ice or alternate valve open (37DM & 87DM)	Thrust computation
WING ANTI ICE 2	GND	O/C	R normal wing anti ice or alternate valve open (38DM & 88DM)	Thrust computation
WING ANTI ICE SELECTION	+28VDC	O/C	Wing anti ice function selected (P/BSW 112DM)	Thrust computation
ENG1 ANTI ICE	GND	O/C	ENG1 anti ice selected (43DN and 19KC)	Thrust computation
ENG2 ANTI ICE	GND	O/C	ENG2 anti ice selected (44DN and 18KC)	Thrust computation

TCC Code Pin Program

PINS	CODE	LEVEL	ENGINE	FUNCTION
AA10A	PIN PROG ENG FAMILY	+28VDC	PW	
AA11A	PIN PROG ENG FAMILY	+28VDC	PW	Monitoring and
AA12A	PIN PROG ENG MODEL	+28VDC or O/C (at least one+28VDC)	Selection of engine type	selection of data in memory
AA13A	PIN PROG ENG MODEL			
AA14A	PIN PROG ENG MODEL			
AA 5H	- Pin	O/C	-	
AA 6H	Prog.	+28VDC	Selection	Select of control
AB11C	A/C Type	O/C	of A/C Type	laws
AB12C	-	+28VDC	-	

Engine Failure

SIGNAL	LEVEL	SIGNIFICANCE	FUNCTION
ENGINE 1 FAILED	GND	Engine failure	THR LATCH
ENGINE 2 FAILED	GND	Information from FWC1	Logic
ENGINE 1 FAILED	GND	Engine failure	THR LATCH
ENGINE 2 FAILED	GND	Information from FWC2	Logic

(6)Instinctive disconnect pushbutton switches
Each throttle control lever is equipped with an instinctive disconnect pushbutton switch.
Instinctive disconnect pushbutton switches provide the following function :
- if **A/THR** or **EPR LATCH** mode is engaged, mode is disengaged by pressing one instinctive disconnect pushbutton switch without the arming lever returning to the **OFF** position.

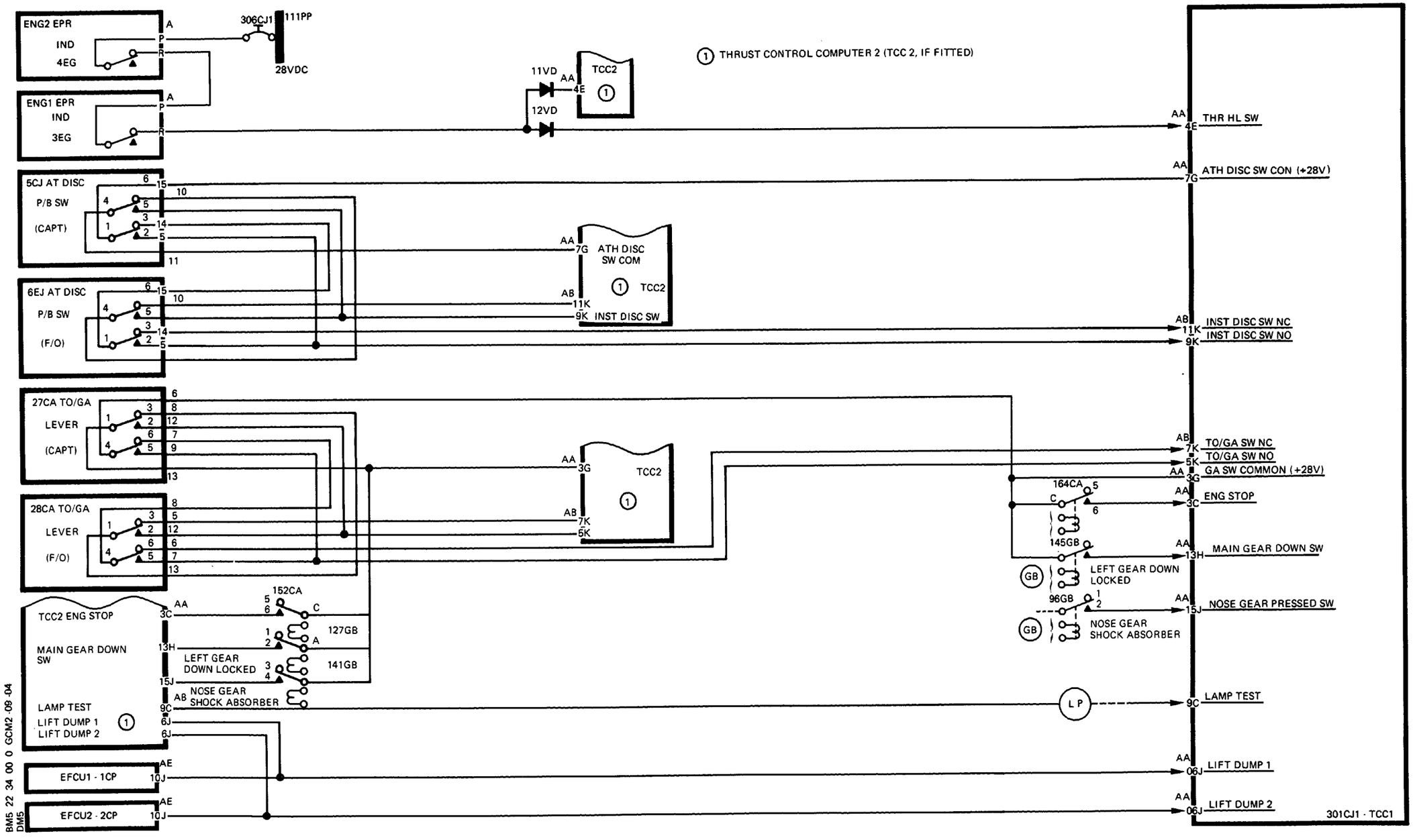
(7)**TO/GA** levers.
Each throttle control lever is equipped with a **TO/GA** lever.
TO/GA levers enable engagement of **A/THR** mode or **EPR LATCH** mode if **A/THR** cannot be selected.
On the ground, after selection of the **TO** or **FLX TO** mode on the **TRP**, **A/THR** mode is automatically engaged by pulling one **TO/GA** lever.

(8)**EPR** indicator microswitches.
In flight and in the event of **EPR** limit computation failure the pilot can set the bugs manually.
If in **SPD** mode, the engine rating exceeds the value indicated by the bug in either **EPR** indicator, the throttle control levers stop as long as this detection is present.

(9)**FAC/ATS** engage unit and **TCC** interface.
The **TCC** receives the signals required for system arming from **FAC/ATS** engage unit **318CC**.
Whether the lever is held in the **ON** position is contingent on the command and monitoring logic validity (see arming logic). Self-latching relay power supply goes through the associated arming lever. System disconnection is indicated by the lever returning to the **OFF** position.
(Ref. Fig. 006)

(10)Input discretes (definition)

SIGNAL	LEVEL		NOTES
	ACTIVE	NOT ACTIVE	
MAIN GEAR UP	28 V	OC	
NOSE GEAR PRESSED	28 V	OC	
ACTUATOR CONNECTED	Ground	OC	
FORCE SENSOR CONNECTED	Ground	OC	
INSTINCT. DISC NO SW	28 V	OC	
INSTINCT. DISC NC SW	OC	28 V	
LIFT DUMP 1 SW	Ground	OC	
LIFT DUMP 2 SW	Ground	OC	
TO/GA NO SW	28 V	OC	
TO/GA NC SW	OC	28 V	
HIGH LIMIT NC SW	OC	28 V	
ATH TOGGLE SW (COM)	28 V	OC	
ATH TOGGLE SW (MON)	28 V	OC	



TCC Discrete Inputs - Schematic
Figure 006

NOTE : TCC SELECT and TCC SEL TO OPP signals are generated by each TCC so that the other computers select TCC1 or TCC2 bus according to the state of each signal.

C. ARINC Signals

(1)FAC - TCC, FCC - TCC interface

ARINC 429 Word Characteristics - FAC to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
Discrete data 1 (FAC DW1)	274			18							AT mode logic
Minimum airspeed (MIN SPD)	245	kt	512	13	0.0625	125		X		X	Flight envelope protection
Placard speed (MAX SPD)	207	kt	512	11	0.25	200		X		X	Flight envelope protection
Average angle-of-attack (AVERAGE AOA)	241	deg/ 180°	±180°	12	0.05	62,5		X		X	SPD control laws

FAC ARINC Output Signals - FAC Discrete 2

BIT No.	FUNCTION	ENCODING ZERO	ENCODING ONE	NOTES
1 to 8	Label 274			
9 - 10				
11 to 15	S/F SELECTOR POSITION 1 to 5	ONLY ONE BIT AND ONE AT LEAST SET TO THE ONE STATE		
16				
17				
18	BOTH SFCC VALIDITY	NOT VALID	VALID	
19	LEFT GEAR PRESSED	NOT PRESSED	PRESSED	
20	RIGHT GEAR PRESSED	NOT PRESSED	PRESSED	
21	GEAR SW VALIDITY	NOT VALID	VALID	
22	OWN EFCU VALIDITY	NOT VALID	VALID	
23	BOTH EFCU VALIDITY	NOT VALID	VALID	
24 - 25	-			
26	THROTTLE PUSHER	Alpha < Alpha 0	Alpha ≥ Alpha 0	
27 - 29	-			
30 - 31	SSM			

BIT No.	FUNCTION	ENCODING		NOTES
		ZERO	ONE	
32	PARITY			

ARINC 429 Word Characteristics - FCC to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	RESOLUTION	SIG BITS	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
AFS discretes	146	FCC DW4			18			X			SPEED or MACH automatic engagement LAND TRACK
AFS discretes	270	FCC DW5			18						"
AFS discretes	273	FCC DW3									
AFS discretes	274	DW1									

FCC - ARINC Output Signals - FCC Discretes 4

BIT No.	FUNCTION	ENCODING ZERO	ENCODING ONE	NOTES
1 - 8	Label 146			
9 to 13	Not Used			
14	Land Track	NOT REQUESTED	REQUESTED	
15 to 29	Not Used			
30	Matrix			
31	"			
32	Parity			

ARINC Output Signals (continued) - FCC Discretes 5

BIT No.	FUNCTION	ENCODING ZERO	ENCODING ONE	NOTES
1 - 8	Label 270			
9	IRS 2 Validity			
10 to 24	Not Used			
25	IRS 1 Validity		Valid	
26 to 29	Not Used			
30	Matrix			
31	"			
32	Parity			

ARINC Output Signals - FCC Discretes 3

BIT No.	FUNCTION	ENCODING		NOTES
		ZERO	ONE	
1 - 8	Label 273			
9 to 14	Not Used			
15	PROFILE ARM	Not Armed	Armed	
16 to 19	Not Used			
20	LAND ARM	Not Armed	Armed	
21 to 29	Not Used			
30	Matrix			
31	"			
32	Parity			

ARINC Output Signals (continued) - FCC Discretes 1

BIT No.	FUNCTION	ENCODING		NOTES
		ZERO	ONE	
1 - 8	Label 274			
9	- - DES			Coded Data
10	- - CLIMB			ALT ACQ 13, 29
11	SPD			ALT 13, 22
12	PROFILE			G/S CPT 19, 29
13	ALT			G/S TRK 19, 22
14	V/S			SPD CLB
15				10, 11, 12
16	PITCH GA			SPD DES
17	PITCH TO			9, 11, 12
18	MACH			MACH CLB
19	G/S			10, 18, 12
20		Not Requested	Not Requested	MACH DES
21				9, 18, 12
22	- - TRK			PROFILE ALT
23				12, 9, 10
24				
25				PROFILE CLB
26				12, 10
27				
28				PROFILE DES
29	- - CPT			12, 9
30	Matrix			PSPD PHASE
31	"			11, 2
32	Parity			PMACH PHASE
				18, 12

(3)FCU - TCC interface

ARINC 429 Word Characteristics - FCU to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	SIG BITS	RESOLUTION	TRANSMIT INTERVAL (ms)	SPEED H	SPEED L	CODE BCD	CODE BNR	FUNCTIONS
Selected Mach (SEL MACH)	106	mMACH	0 to 4096	12	1	62,5		X		X	MACH hold mode
Discrete word 1 (FCU DW1)	270			18		56					MACH mode, SPD mode, EPR mode, LVL CHANGE logic (see DTD FCU)
Selected airspeed (SEL SPD)	103	kt	512	11	0.25	200		X		X	SPEED hold mode

FCU Discrete Word

BIT	FUNCTION	LEVEL 0	LEVEL 1	NOTES
1 - 8	LABEL 270			
9 to 10	SDI			
11				
12	FCC1 SELECT		SELECT	
13	FCC2 SELECT		"	
14 to 24	Not Used			
25	A/THR PUSHBUTTON SWITCH	NOT PRESSED	PRESSED	
26 to 29	Not Used			
30	MATRIX			
31	"			
32	PARITY			

(4)MTP - TCC interface

ARINC 429 Word Characteristics - MTP to TCC

PARAMETER NAME	LABEL (octal)	UNITS	RANGE	SIG BITS		TRANSMIT INTERVAL (ms)	SPEED		CODE B C D	CODE B N R	FUNCTIONS
Maintenance data 1	350			21							Fault location task and LAND TEST
Maintenance data	351			21							"
Maintenance data	353			21							"

(5)TCC ARINC output

ARINC Outputs - 12 KHz Line to EEC2 and other systems.

LABEL (octal)	PARAMETER NAME	CYCLE	TRANSMIT INTERVAL (ms)	SIG BITS	UNITS	RANGE	NOTES
211	Total Air Temperature		438	11	°C	512	ADC (1 or 2) to EEC
203	Altitude 1013		62.5	17	feet	131072	"
205	Mach		125	16	mMach	4096	"
242	Total Pressure		125	16	mb	2048	"
214	Derate Temperature		125	7	°C	99	TRP
342	Limit Thrust (**)		125	12	EPR	4	"
343	Target Thrust		125	12	EPR	4	"
021 BCD	Nominal Thrust		125	8	EPR	3	"
302	Limit Thrust Test		125	12	EPR	1.42	"
340	Actual Thrust-EEC1 (*)		62.5	12	EPR	4	TCC/M
340	Actual Thrust-EEC2 (*)		62.5	12	EPR	4	"
341	CMD Thrust-EEC2 (*)		125	12	EPR	4	"
164	Target Height		125	16	ft.	8192	
103	Target Speed (FMC)		125	11	Kt	512	
273	TCC Discretes 3		62.5	21			
274	TCC Discretes 4		62.5	21			
275	TCC Discretes 5		62.5	21			
350	Maintenance Data		360	21			
351	Maintenance Data		360	21			
353	Maintenance Data		360	21			
354	Maintenance Data		360	21			
	(Context Table)						

(*)Engine 1 or 2 signals are transmitted by the same label actual thrust EPR, label 340.

(**) SDI = 01 (TCC1)
= 10 (TCC2 if fitted)

Bit 10	Bit 9	
0	1	left
1	0	right

ARINC Outputs - 12 KHz Line - TCC to EEC1

LABEL (octal)	PARAMETER NAME	CYCLE	TRANSMIT INTERVAL (ms)	SIG BITS	UNITS	RANGE	NOTES
203	Altitude Standard		62.5 ms	17	ft	131072	SDI ADC 2
205	Mach		62.5 ms	13	mMach	4096	
211	Total Air Temperature		62.5 ms	11	°C	512	
242	Total Pressure		62.5 ms	11	mb	2048	

LABEL (octal)	PARAMETER NAME	CYCLE	TRANSMIT INTERVAL (ms)	SIG BITS	UNITS	RANGE	NOTES
273	TCC Discrete Word 3		62.5 ms	21			
340	THR Actual		62.5 ms	14		4	
341	THR CMD		62.5 ms	14		4	
342	THR LIMIT		62.5 ms	14		4	

ARINC Outputs - 12 KHz Line (continued)
Label 273 - TCCDW3 Definition

BIT No.	FUNCTION		BIT STATUS 1	0
1 to 8	Label 273			
9	SDI			
10	SDI			
11	Bleed Air Correction	Air Conditioning 1	See NOTE	
12		Air Conditioning 2		
13		Wing Anti-Icing 1		
14		Wing Anti-Icing 2		
15		Nacelle Anti-Icing		
16				
17				
18				
19				
20				
21	Descent Idle		1	0
22	Weight on Wheels		1	0
23	Slats retracted		1	0 (slats >0)
24	VAPP		1	0 -|
25	FLAPS		1	0 | DFA (If
26	GEAR		1	0 _| fitted)
27	EEC1 On-Off		On	Off
28	EEC2 On-Off		On	Off
29	ADC Comparison		ADC1 ≠ ADC2	ADC1 = ADC2
30				
31				
32	Parity			

NOTE : Air Cond Correction(KΔEPR1) | Bit 11 | Bit 12

0	0	0
+1	0	1
+2	1	1

Wing Anti-Ice 1 Correction (KΔEPR2)	Bit 13	Wing Anti-Ice 2 Correction	Bit 14
0	0	0	0
+1	1	+1	1

Nacelle Anti-Ice Correction (KΔEPR4)	Bit 15
0	0
+1	1

ARINC Outputs - 12 KHz Line (continued)
Label 274 - TCCDW4 Definition

BIT No.	FUNCTION		BIT STATUS 1	BIT STATUS 0
1 to 8	Label 274			
9				0
10				0
11	ATS Arming Lever (OFF position)		OFF	Armed
12		MAN THR	Armed	OFF
13		ATHR Mode	Engaged	Not Engaged
14				0
15				0
16	ATS Clutches - off (Takeoff)		Off	On
17		VNAV	Engaged	Not Engaged
18				0
19	A/THR	THRUST EPR	Engaged	Not Engaged
20	Mode	MACH	Engaged	Not Engaged
21	Status	SPEED	Engaged	Not Engaged
22		RETARD	Engaged	Not Engaged
23	Engine	Engine 1 F/W	Failed	Not Failed
24	Fail	Engine 2 F/W	Failed	Not Failed
25				0
26				0
27				0
28	Throttle Pusher		Activated	Not Activated
29				0
30	O--- SSM			0
31	O--- "			0
32	Parity			

ARINC Outputs - 12 KHz Line (continued)
Definition
Label 275 - TCCDW5 Definition

BIT No.	FUNCTION	BIT STATUS 1	BIT STATUS 0
1 to 8	Label 275		
9	Target Display Code	See coding below	
10	" " "	See coding below	
11			
12			
13	Engine Type Code	See Coding Below	
14			
15			
16	Limited Autothrottle Control	Limited	Not Limited
17	TO		
18	FLX TO		
19	CL	Selected	
20	CR	Thrust Limit	Not Engaged
21	MCT	Mode	
22	GA		
23			0
24	Limit Thrust Computation F/W	F/W	No F/W
25	ADC1 Selection	1	0
26	ADC2 Selection	1	0
27			0
28	Automatic Thrust Limit Mode Selection	Auto	Manual
29	Autothrottle in EPR Control	Engaged	Not Engaged
30	SSM		0
31	"		0
32	Parity		

TARGET DISPLAY CODE

BIT No. 9	BIT No. 10	DISPLAYED TARGET
0	0	FLX TO
1	0	FMC (021)
0	1	LIMIT (342)
1	1	IDLE

ENGINE TYPE CODING

PINS AA	10A	11A	12A	13A	14A
BIT No.	11	12	13	14	15
	1	1	0	1	0
	PW		4152		

1. Thrust Rating Panel (TRP)

A. General

The thrust rating panel (TRP) enables communication between the flight crew and the TCC.

The main functions of the TRP are :

- display of the THR LIMIT value computed by the TCC
- display of the TAT value which is normally transmitted by the armed TCC In the event of TCC failure the TAT value is directly taken from ADC1 data bus
- display of the TARGET thrust value computed by the FMC and transmitted by the TCC. In the event of TCC failure the TARGET thrust value is directly taken from FMC1 data bus
- display of the mode selected by the pilot or by the FMC when in AUTO mode
- transmission to the TCC of the mode selected by the pilot and of the fictitious temperature set for the flexible takeoff mode.

TRP general characteristics are :

- dimensions : 81 mm x 126 mm x 228 mm
- weight : 3 kg approximately
- dissipated power : 25 W approximately in normal operating condition.

B. Description

(1)The face of the TRP includes :

- a keyboard with six mode selection pushbutton switches
- a selector knob equipped with an optical incremental coder for fictitious temperature selection in flexible takeoff mode
- a THR LIMIT display
- a TARGET thrust display
- a TAT display
- an FLXTO TEMP display (selected fictitious temperature)
- a LIM MODE display (selected limit mode).

(a)Mode selection keyboard

The mode selection keyboard comprises six pushbutton switches. These pushbutton switches are integrated in the face.
They are provided with two 5 V bulbs used for mode confirmation.
No mode logic is accomplished in the TRP. Any action on a pushbutton switch delivers a signal which is transmitted directly via TRP ARINC output data bus (TRPDW1 label 270) after anti-bounce conditioning. When a pushbutton switch is pressed three stripes whose signals are taken into account by the TCC transmitted data come on.
The pushbutton switches are :

TO GA : takeoff/go around (green)
FLX TO : flexible takeoff (green)
CL : climb (green)
CR : cruise (green)
MCT : maximum continuous (green)
AUTO : automatic selection (green).

(b)Displays

Displays are three or four-digit seven-segment built-in modules. Their control is latched. However updating frequency is 56 ms.
Display is of transflective type with white-colored information on a black background. Night lighting is provided by two lamps located behind each display.
They can be seen clearly from an angle of ±60°.

1 Three-digit displays

This type of display is used for TAT and FLX TO :

- the first digit only shows + or -
- the second and third digits show a digital value between 0 and 99
- in lamp test mode the first digit shows +, the second and third digits show 88.

2 Three-digit display

This type of display is used for LIM MODE :

- this module can show : TO, GA, CL, CR and MCT
- in lamp test mode all segments are visible.

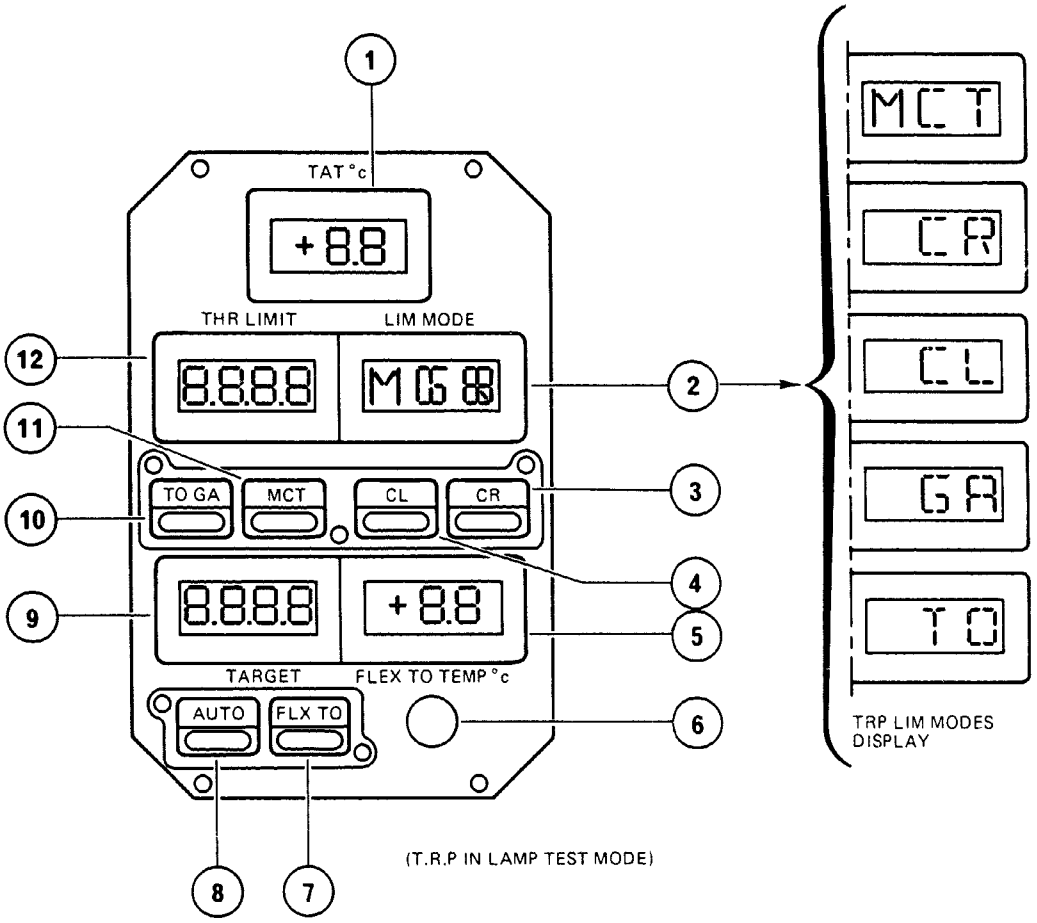
3 Four-digit display

This type of display is used for THR LIMIT and TARGET :

- each digit can show a figure from 0 to 9
- a decimal point enables EPR display
- in lamp test mode all segments are visible.

(Ref. Fig. 001)

C. Flexible Takeoff Temperature Selection



1. TAT (TOTAL AIR TEMPERATURE) DISPLAY
2. LIM MODE DISPLAY
3. CR (CRUISE) MODE SELECTION P/B SW
4. CL (CLIMB) MODE SELECTION P/B SW
5. FLX TO TEMP DISPLAY
6. FLEXIBLE TAKE-OFF TEMPERATURE SELECTION KNOB
7. FLEXIBLE TAKE-OFF MODE SELECTION P/B SW
8. AUTO MODE SELECTION P/B SW
9. THRUST TARGET DISPLAY
10. MCT (MAXIMUM CONTINUOUS) MODE SELECTION P/B SW
11. TO GA (TAKE-OFF/GO AROUND) MODE SELECTION P/B SW
12. THR LIMIT DISPLAY

BM5 22 35 00 0 AKM2 —08—03
DM5

Thrust Rating Panel
Figure 001

- A selector knob equipped with an optical incremental coder enables selection of the reference value which appears in the FLX TO TEMP display.
  The selector knob is equipped with a notch mechanism with 32 clicks per turn.
- Each click corresponds to 1°C increment or decrement depending on the direction of rotation of the selector knob.
- Selection of the FLX TO mode synchronizes the display value to the TAT value.
- Any selection of a FLX TO value lower than the TAT value results in "---" in the FLX TO display (processing performed by the TCC).

D. Operation

(1)Electrical power supply

The 28VDC provides primary power supply to the TRP ; power consumption is less than 25 W.

The TRP is also connected to the 8VDC power supply which is used by the mode confirmation lighting ; current consumption is 0.6 A approximately during test and 0.12 A in normal operation (in fact it is contingent on the DIM value).

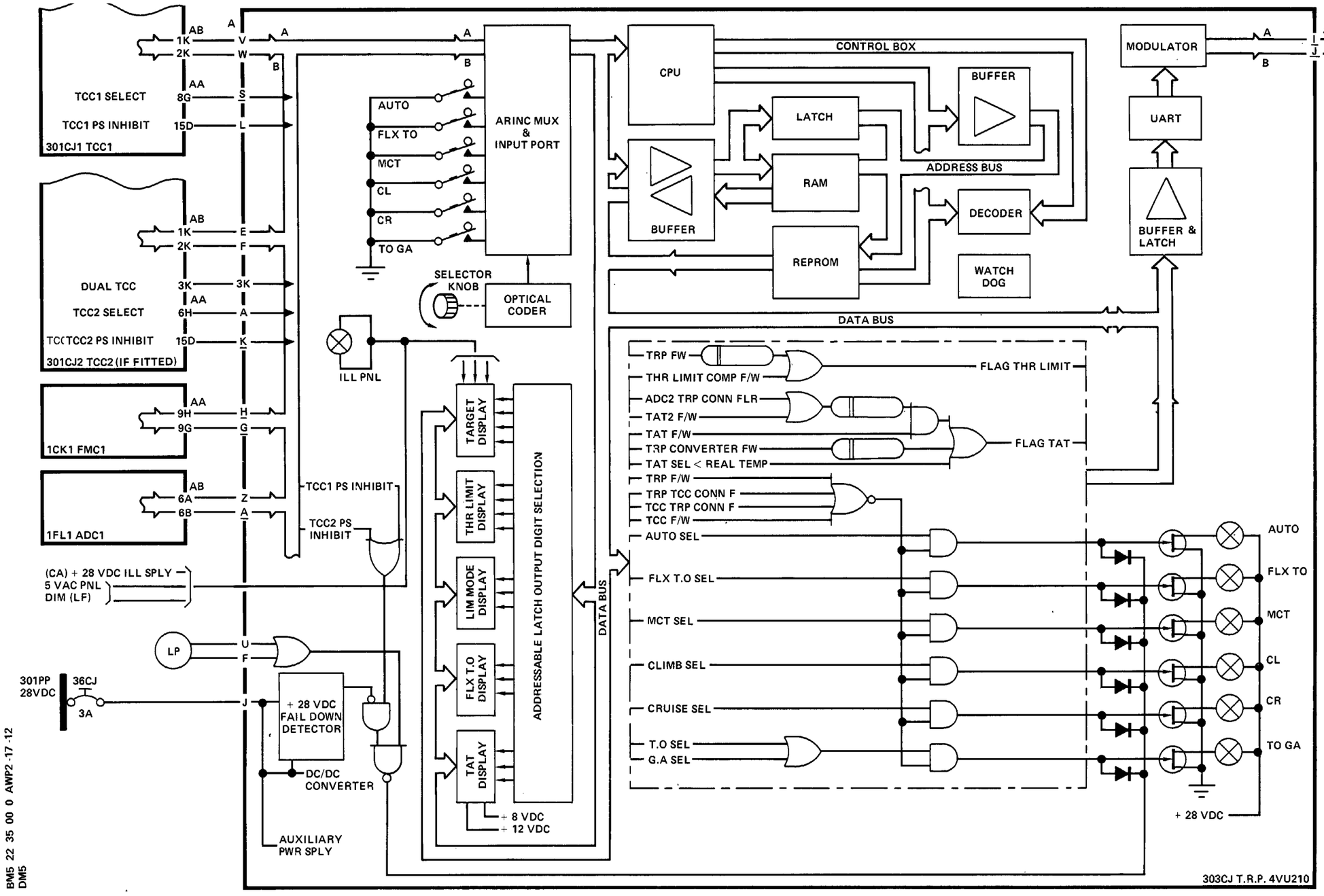
For night lighting the TRP receives a variable 5VAC signal ; current consumption on this signal is less than 1.4 A.

(2)Interface

(Ref. Fig. 002)

The TRP is connected to the TCCs, FMC1 and ADC1.

(Ref. Fig. 002)



TRP - Interconnection and Functional Block Diagram
Figure 002

(3)Functional aspect
The following pages give a description of ARINC input/output and logic and analogic signal definition.

TRP - Inputs/Outputs
ARINC Inputs

SOURCE	PARAMETER	OCTAL LABEL	FUNCTION	NOTES
	Limit Thrust	342	Display (THR LIMIT) (TARGET)	
TCC1	Total Air Temperature	211	Display (TAT)	
or	Derate Temperature	214	SELTAT LOOP TEST	Selected TAT value feedback
TCC2 (if fitted)	TCCDW2	275	TCC Mode + Display confirm (LIM MODE)	
(ARINC 703)	Derate Target Thrust	343	Display (TARGET)	
	Nominal Target Thrust	021	Display (TARGET)	
ADS (ARINC 706)	Total Air Temperature	211	Display (TAT)	Only read in the absence of TCC1 or TCC2 (if fitted) results
FMC1	EPR/ Target	021	Display (TARGET)	
	FMCDW1	270	EPR Display CMD	
	FMCDW2	271	Idle Display CMD	

ARINC Output

LABEL	PARAMETER NAME	TRANSMIT INTERVAL (ms)	SIG BITS (NOT INC SIGN)	UNIT	RANGE	NOTES
214	Derate Temp (SELTAT)	56	7	°C	99°	
270	TRP Discrete ≠1 (TRPDW1)	56	-	-	-	Ref. following detailed table
342	Limit Thrust Echo (THRUMFB)	56	14/12	EPR	256/4	THR LIMIT recopy for TRP to TCC interface test
350	Software Ident	-	-	-	-	Transmitted once at power rise or after interruption ≥ 200 ms

TRPDW1 ARINC Word Format

BIT No.	LABEL FUNCTION	ENCODING		NOTES
		ZERO	ONE	
1	-			
2				
3				
4	Label 270			
5				
6				
7				
8	_			
9	-			
10				
11				
12	Space			
13				
14				
15				
16	-			
17	TO P/B SW	NO SEL	SEL	* Bit Nos 17 and 22 provide the same information TO/GA
18	FLX TO P/B SW	"	"	
19	CL P/B SW	"		
20	CR P/B SW	"	"	
21	MCT P/B SW	"	"	
22	GA P/B SW	"	"	
23	-			
24				
25	Spare			

BIT No.	LABEL FUNCTION	ENCODING		NOTES
		ZERO	ONE	
26				
27	-			
28	AUTO P/B SW	NO SEL	SEL	
29	-			
30	SSM			
31	-			
32	Parity			

TRP - Analog Inputs

SIGNAL	TYPE	GRADIENT	RANGE	POSITIVE SENSE	FUNCTION	NOTES
5 V PANEL DIM	AC		0 V -5Vrms		Night lighting brightness adjustment - markings - pushbutton switches - displays	

SIGNAL	STATUS		FUNCTION	NOTES
	ACTIVE	NOT ACTIVE		
LAMP TEST	+28 V	OC	Lighting of all mode selection pushbutton switches and display segments	Maximum intensity
TCC PS INH 1	GND	OC	TRP power supply cut off	TCC PS INH 1 is required to cut off power
TCC SELECT	+28 V	OC	TCC selected	Conditions ARINC line reading switching
TEST PIN 1	GND	OC	Specific for bench maintenance	Spare
SPARE TCC 1 TO TRP	GND	OC	Spare	
DIM DAY/NIGHT	GND	OC	Mode confirmation light level	GND maximum intensity
		OC = Open Circuit		

TRP - Output Discrete Signals

SIGNAL	STATUS		FUNCTION	NOTES
	ACTIVE	NOT ACTIVE		
Spare TRP TO TCC1	28 V	OC	Spare	

E. Lighting

(1)Night lighting : markings

- the markings identifying the modes are illuminated by lamps integrated in the panel face.

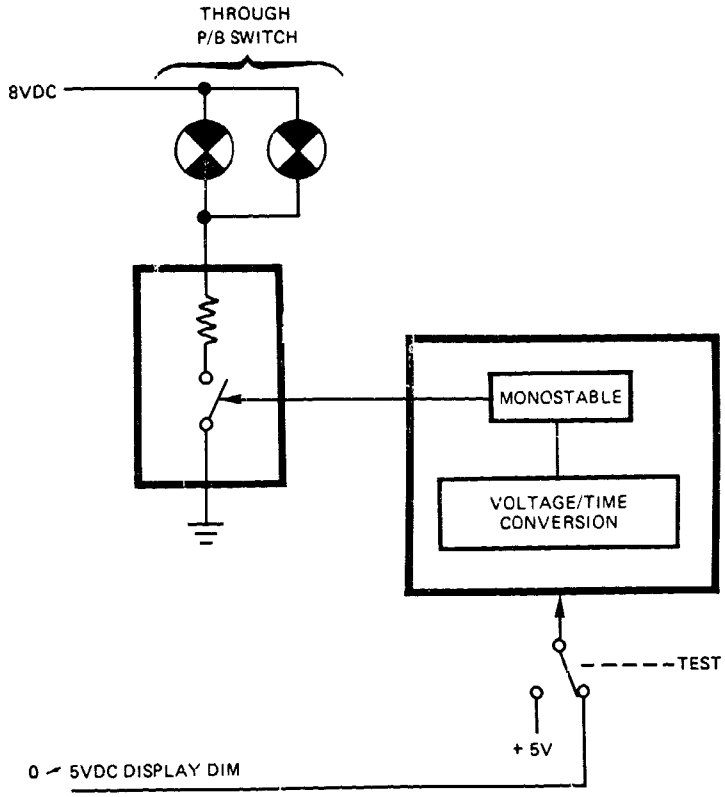
(2)Lighting control

- variation in the lighting of the mode selection pushbutton switch bulbs is obtained by varying the power application time.
- the night lighting intensity (markings and displays) is controlled using the 5VAC power signal amplitude serving for panel lighting. (Ref. Fig. 003).

F. Integrity

- The TRP circuitry monitoring is mainly achieved by a watchdog circuit : each untimely deviation of the program causes a monostable to trigger which results in display interruption and generation of ARINC lines (TRP interruption : the TRP inhibits its power supply).
- TCC - TRP ARINC interface is checked by the TRP by comparing the flexible takeoff temperature value transmitted by the TRP and the same value sent back by the TCC.
- TRP - TCC ARINC interface is checked by the TCC by comparing the EPR limit value transmitted by the TCC and the same value (limit thrust) sent back by the TRP.

(Ref. Fig. 004)



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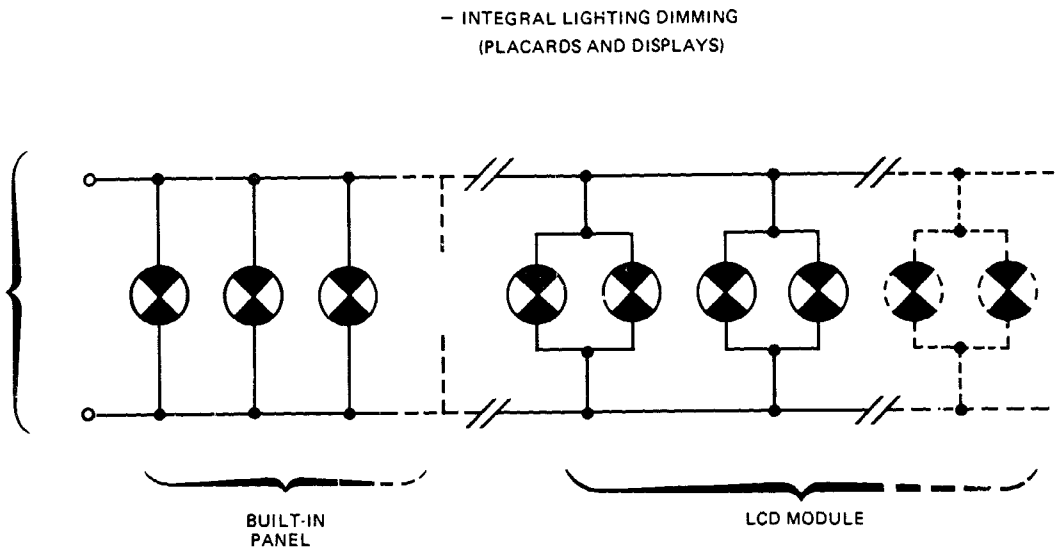
Pushbutton Switch Lighting (Mode Confirmation)
Figure 003



BMS 22 35 00 0 AZM2

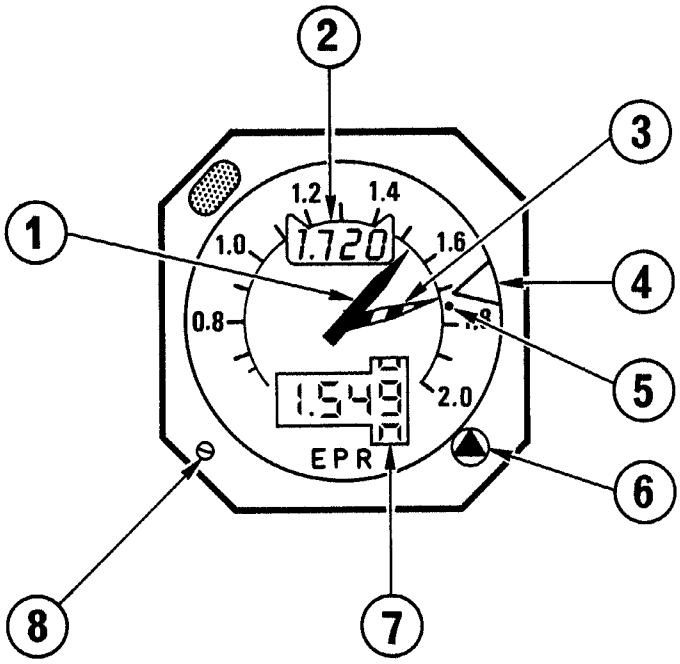
Integral Lighting Dimming (Placards and Displays)
Figure 004

2. EPR Limit Recopy on EPR Indicators - EPR Limit Manual Setting

A. EPR Indicator
(Ref. Fig. 005)

B. EPR Signal Processing
The EPR command and the EPR actual signals are computed in the EECs and then sent to the indicators.

C. EPR Limit Information
(1)Automatic mode
The automatic mode is selected by pushing knob (6) on the EPR indicator. The upper counter (2) is covered by a flag. Each TCC supplies a label 342 on its bus and in ARINC 429 form which serves for bug (4) control. The bug is positioned by a servo device.
(2)Manual mode
The manual mode is selected by pulling knob (6) on the EPR indicator. The flag in upper counter (2) goes out of view. The EPR limit signal delivered by the TCC is replaced by a signal generated in the indicator which represents the EPR limit value manually selected by the pilot.
When the EPR actual signal amplitude (with phase advance to avoid exceeding values) reaches the reference value selected by the pilot a comparator indicates that engine speed is exceeded and that the throttle control levers are stopped.
The throttle control levers are stopped as long as the EPR actual signal value is greater than the EPR limit.
A BITE test can be performed from the indicator by push action (8). If the indicator is operational, this action causes positioning of EPR actual (1) and EPR command (3) pointers, and bug (4) at 1.75 EPR. This value is displayed in EPR actual digital counter (7) (test value 1.75 EPR corresponding to blue dot (5) on the indicator scale).



1. EPR ACTUAL POINTER (WHITE)
   INDICATES ACTUAL ENGINE EPR CALCULATED BY THE EEC
2. DIGITAL COUNTER
   INDICATES EPR LIMIT (MANUAL MODE ONLY)
3. EPR COMMAND POINTER (BLACK AND WHITE)
   INDICATES THE EPR CALCULATED BY THE EEC ACCORDING TO THE THROTTLE SETTING
4. BUG
   INDICATES THE EPR LIMIT
5. BLUE DOT
   AT 1.75 EPR FOR INDICATOR BITE TEST
6. KNOB
   USED TO SELECT MANUAL OR AUTOMATIC EPR LIMIT CALCULATION MODE
7. DIGITAL COUNTER
   INDICATES ACTUAL ENGINE EPR CALCULATED BY THE EEC
8. BITE SCREW
   USED TO TEST INDICATOR OPERATION (WHEN PUSHED)

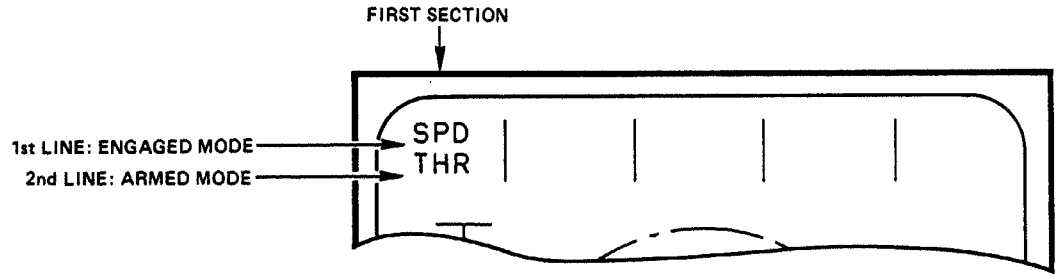
BM5 22 35 00 0 BAP2 24

EPR Indicator
Figure 005

3. TCCS Indicating and Warnings on EFIS Displays
Indicating and warnings are generated by the EFIS SGUs (Symbol Generator Units) which receive the ARINC 429 output data bus from the TCC.

A. TCCS Indicating on EFIS Displays
Information concerning armed or engaged mode appears in the upper part of the Primary Flight Display (PFD) in the first section.
A detailed description of the displays appearing on the PFD is given in the following figure.

B. Warnings
A warning display associated with ATS operation can appear on the PFD :
MAN THR : ATS declutching (loss of A/THR or THR LATCH). MAN THR disappears as soon as A/THR or THR LATCH mode is engaged.
(Ref. Fig. 006).



	DISPLAY	COLOR	MEANING
1st LINE	SPD	GREEN	SPEED SELECTION MODE
	MACH	GREEN	MACH SELECTION MODE
	THR	GREEN	EPR MODE ENGAGED
	RETARD	GREEN	THROTTLE RETARD
	P.THR	GREEN	EPR MODE ENGAGED THROUGH FMC (TARGET MODE)
	P.SPD	GREEN	SPEED MODE ENGAGED THROUGH FMC
	P.MACH	GREEN	MACH MODE ENGAGED THROUGH FMC
	THR/L (FLASHING)	GREEN	EPR LATCH MODE ENGAGED (NO ALPHA FLOOR CONDITION)
	THR/L	GREEN	EPR LATCH MODE ENGAGED (THROUGH ALPHA FLOOR CONDITION)
2nd LINE	A/THR	CYAN	A/THR MODE ARMED
	THR	CYAN	AUTOTHROTTLE DECLUTCH AFTER TO
	A/THR	CYAN	EPR MODE ARMED (LIMIT MODE)
	THR	CYAN	EPR DESCENT MODE
	MAN THR	AMBER	AUTOTHROTTLE DECLUTCHING

BM5 22 35 00 0 BCM2 - 24 - 19
DM5

ATS Mode Indicating on the PFD
Figure 006

4. ATS OFF Warning

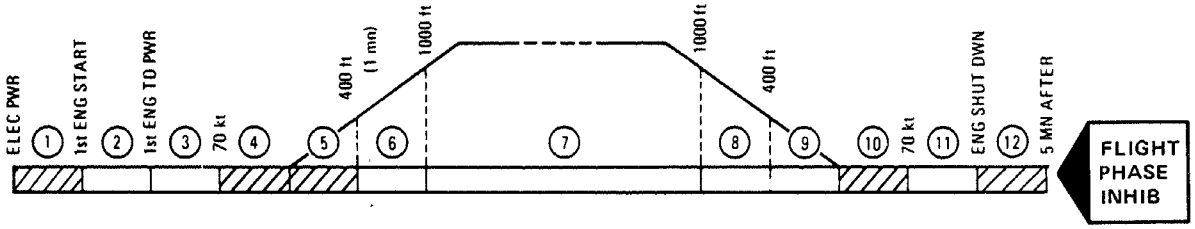
The loss of the ATS system is indicated by :
- the ATS lever returning to the OFF position
- the ATS OFF message on the left ECAM display unit and the MASTER CAUTION lights coming on on the main instrument panels 3VU and 5VU

The following figure show the flight phases during which these warnings are active and give the message displayed on the left ECAM display unit.
(Ref. Fig. 007)

The warnings are cancelled :
- after successful rearming of the system
- by pressing the CLR pushbutton switch on the ECAM control panel
- by pressing the MASTER CAUTION pushbutton switch.

(Ref. Fig. 008)

FAILURE : ATS	DETECTOR :	AURAL WARNING
ATS DISARMED	ATS LEVER	SC
R ECAM DU SYS PAGE CALLED :		MASTER CAUTION
NIL		MASTER CAUTION A



ACTIONS		LOCAL WARNING	WARNING LIGHTS	L ECAM DISPLAY UNIT	
		ATS LEVER IN OFF POSITION	MASTER CAUTION	ATS ATS 1 OFF .....RESET	A A-C
If unsuccessful to reset : Cancel	CLR	Id	nil	STATUS LAND 3 INOP ATS ALPHA FLOOR INOP	 A A
If recall	RCL	Id		ATS ATS 1 OFF .....RESET	A A-C

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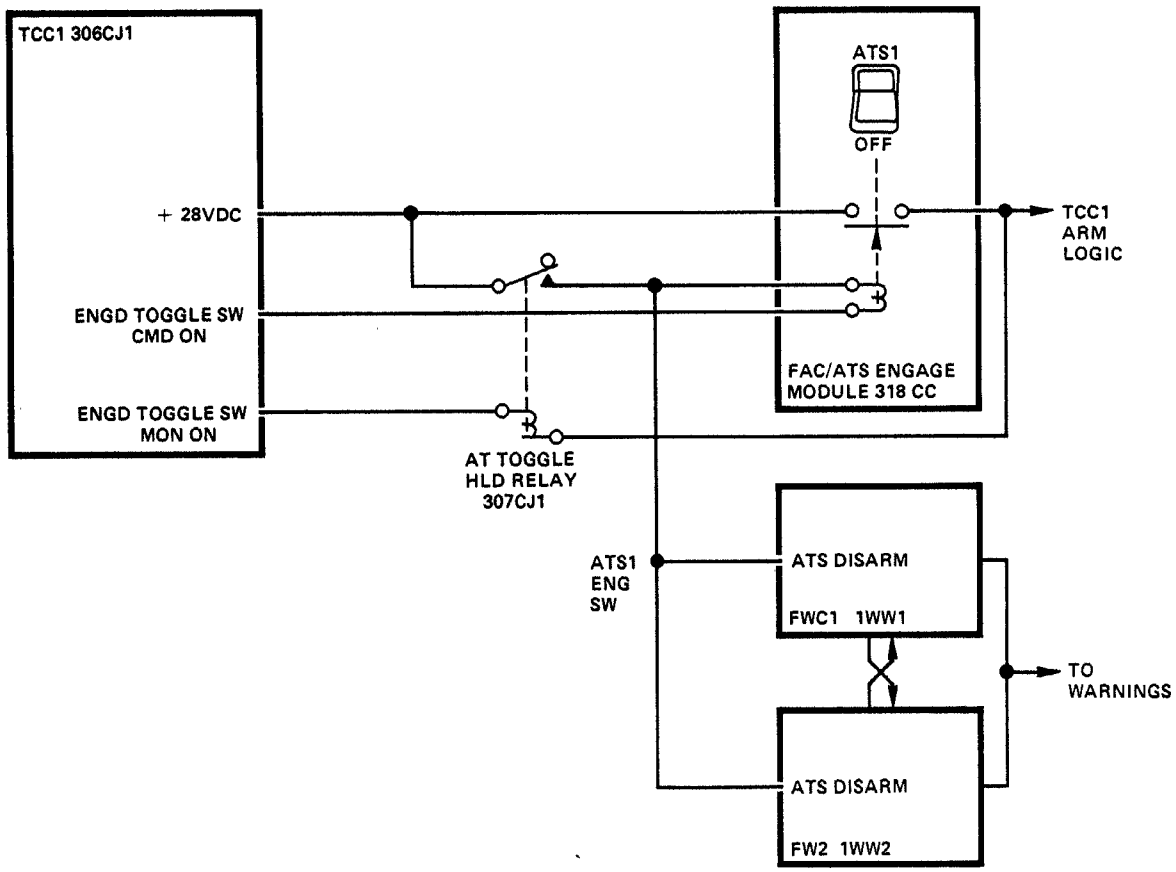
DM5

ATS Disarmed
Figure 007



BM5 22 35 00 0 BEM2 - 24 - 19
DM5

Discretes Used in ATS OFF Warning Generation
Figure 008

1. Thrust Rating Panel (TRP)

A. General

The thrust rating panel (TRP) enables communication between the flight crew and the TCC.

The main functions of the TRP are :

- display of the THR LIMIT value computed by the TCC
- display of the TAT value which is normally transmitted by the armed TCC. In the event of TCC failure the TAT value is directly taken from ADC2 data bus
- display of the TARGET thrust value computed by the FMC and transmitted by the TCC. In the event of TCC failure the TARGET thrust value is directly taken from FMC1 data bus
- display of the mode selected by the pilot or by the FMC when in AUTO mode
- transmission to the TCC of the mode selected by the pilot and of the fictitious temperature set for the flexible takeoff mode.

TRP general characteristics are :

- dimensions : 81 mm x 126 mm x 228 mm
- weight : 3 kg approximately
- dissipated power : 25 W approximately in normal operating condition.

B. Description

(1)The face of the TRP includes :

- a keyboard with six mode selection pushbutton switches
- a selector knob equipped with an optical incremental coder for fictitious temperature selection in flexible takeoff mode
- a THR LIMIT display
- a TARGET thrust display
- a TAT display
- an FLXTO TEMP display (selected fictitious temperature)
- a LIM MODE display (selected limit mode).

(a)Mode selection keyboard

The mode selection keyboard comprises six pushbutton switches. These pushbutton switches are integrated in the face.

They are provided with two 5 V bulbs used for mode confirmation.

No mode logic is accomplished in the TRP. Any action on a pushbutton switch delivers a signal which is transmitted directly via TRP ARINC output data bus (TRPDW1 label 270) after anti-bounce conditioning. When a pushbutton switch is pressed three stripes whose signals are taken into account by the TCC transmitted data come on.

The pushbutton switches are :

TO GA : takeoff/go around (green)
FLX TO : flexible takeoff (green)
CL : climb (green)
CR : cruise (green)
MCT : maximum continuous (green)
AUTO : automatic selection (green).

(b)Displays

Displays are three or four-digit seven-segment built-in modules. Their control is latched. However updating frequency is 56 ms.
Display is of transflective type with white-colored information on a black background. Night lighting is provided by two lamps located behind each display.
They can be seen clearly from an angle of ±60°.

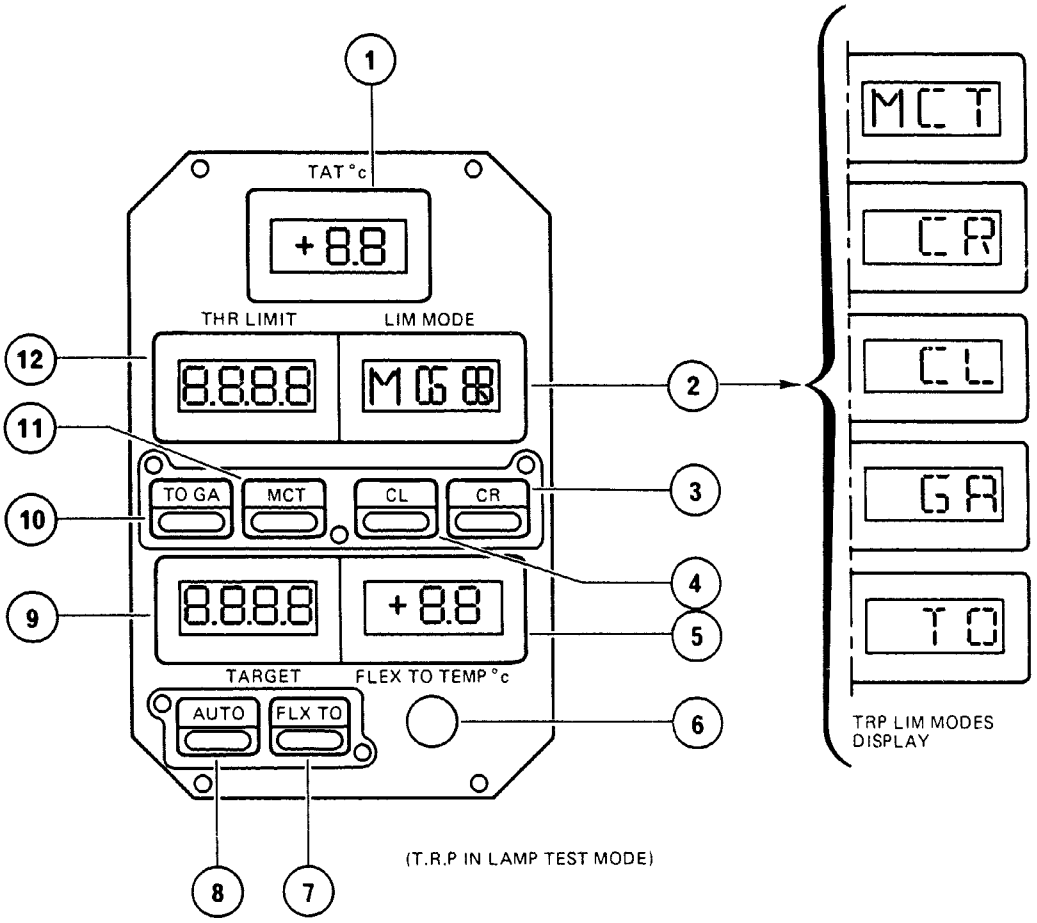
1 Three-digit displays
This type of display is used for TAT and FLX TO :
- the first digit only shows + or -
- the second and third digits show a digital value between 0 and 99
- in lamp test mode the first digit shows +, the second and third digits show 88.

2 Three-digit display
This type of display is used for LIM MODE :
- this module can show : TO, GA, CL, CR and MCT
- in lamp test mode all segments are visible.

3 Four-digit display
This type of display is used for THR LIMIT and TARGET :
- each digit can show a figure from 0 to 9
- a decimal point enables N1 or EPR display
- in lamp test mode all segments are visible.

C. Flexible Takeoff Temperature Selection
- A selector knob equipped with an optical incremental coder enables selection of the reference value which appears in the FLX TO TEMP display.
  The selector knob is equipped with a notch mechanism with 32 clicks per turn.
- Each click corresponds to 1°C increment or decrement depending on the direction of rotation of the selector knob.
- Selection of the FLX TO mode synchronizes the display value to the TAT value.
- Any selection of a FLX TO value lower than the TAT value results in "---" in the FLX TO display (processing performed by the TCC).

(Ref. Fig. 001)



1. TAT (TOTAL AIR TEMPERATURE) DISPLAY
2. LIM MODE DISPLAY
3. CR (CRUISE) MODE SELECTION P/B SW
4. CL (CLIMB) MODE SELECTION P/B SW
5. FLX TO TEMP DISPLAY
6. FLEXIBLE TAKE-OFF TEMPERATURE SELECTION KNOB
7. FLEXIBLE TAKE-OFF MODE SELECTION P/B SW
8. AUTO MODE SELECTION P/B SW
9. THRUST TARGET DISPLAY
10. MCT (MAXIMUM CONTINUOUS) MODE SELECTION P/B SW
11. TO GA (TAKE-OFF/GO AROUND) MODE SELECTION P/B SW
12. THR LIMIT DISPLAY

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DM5

Thrust Rating Panel
Figure 001

D. Operation

(1)Electrical power supply

The 28VDC provides primary power supply to the TRP ; power consumption is less than 25 W.

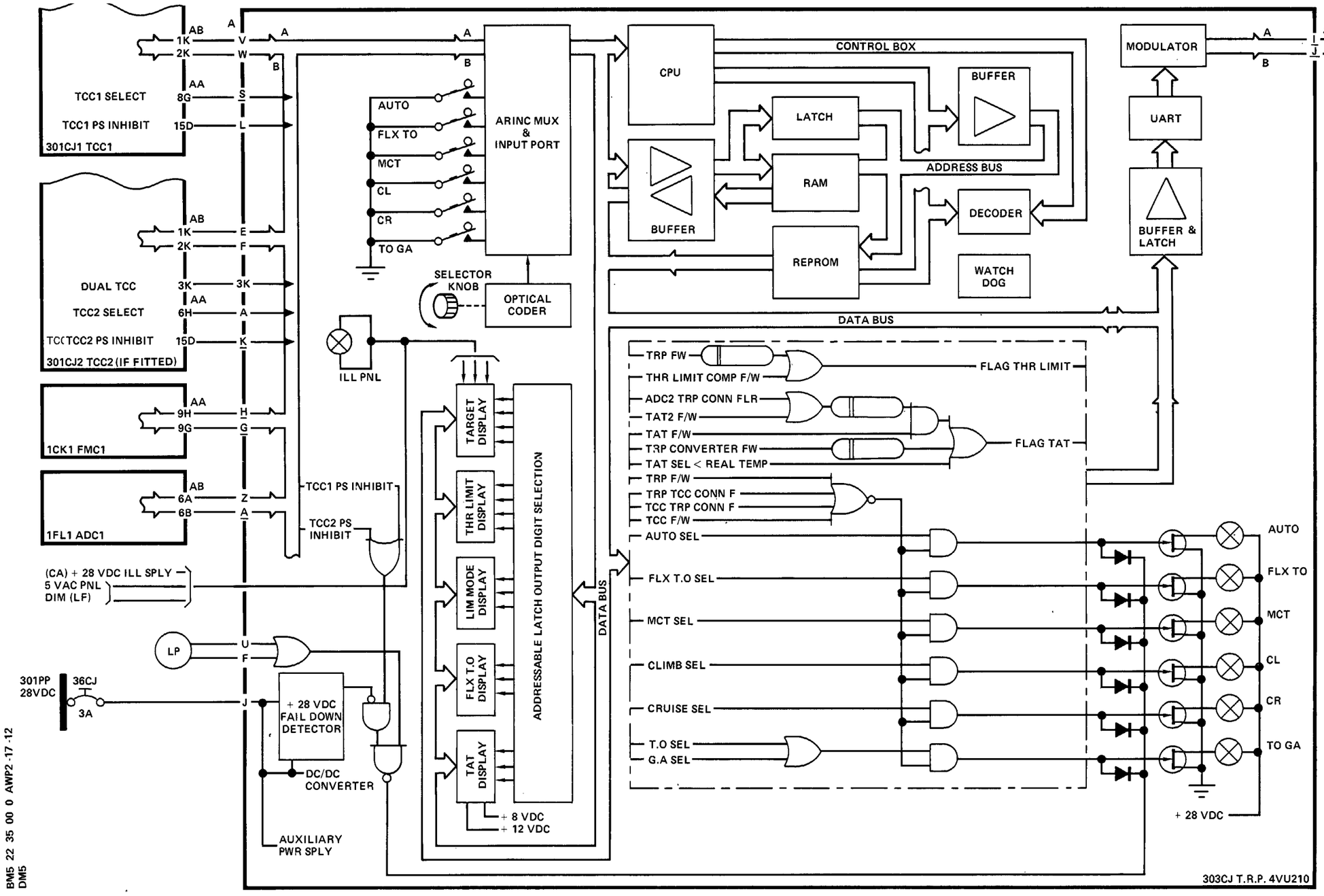
The TRP is also connected to the 8VDC power supply which is used by the mode confirmation lighting ; current consumption is 0.6 A approximately during test and 0.12 A in normal operation (in fact it is contingent on the DIM value).

For night lighting the TRP receives a variable 5VAC signal ; current consumption on this signal is less than 1.4 A.

(2)Interface

(Ref. Fig. 002)

The TRP is connected to the TCCs, FMC1 and ADC1.



TRP - Interconnection and Functional Block Diagram
Figure 002

The following pages give a description of ARINC input/output and logic and analog signal definition.

TRP - Inputs/Outputs

ARINC Inputs

SOURCE	PARAMETER	OCTAL LABEL	FUNCTION	NOTES
	Limit Thrust	342	Display (THR LIMIT) (TARGET)	
TCC1	Total Air Temperature	211	Display (TAT)	
or	Derate Temperature	214	SELTAT LOOP TEST	Selected TAT value feedback
TCC2 (if fitted)	TCCDW2	275	TCC Mode + Display confirm (LIM MODE)	
(ARINC 703)	Derate Target Thrust	343	Display (TARGET)	
	Nominal Target Thrust	021	Display (TARGET)	
ADS (ARINC 706)	Total Air Temperature	211	Display (TAT)	Only read in the absence of TCC1 or TCC2 results
FMC1	N1 Target	021	Display (TARGET)	
	FMCDW1	270	N1 Display CMD	
	FMCDW2	271	Idle Display CMD	

ARINC Output

LABEL	PARAMETER NAME	TRANSMIT INTERVAL (ms)	SIG BITS (NOT INC SIGN)	UNIT	RANGE	NOTES
214	Derate Temp (SELTAT)	56	7	°C	99°	
270	TRP Discrete ≠1 (TRPDW1)	56	-	-	-	Ref. following detailed table
342	Limit Thrust Echo (THRUMFB)	56	14/12		256/4	THR LIMIT recopy for TRP to TCC interface test
350	Software Ident	-	-	-	-	Transmitted once at power rise or after interruption ≥ 200 ms

TRPDW1 ARINC Word Format

BIT No.	LABEL FUNCTION	ENCODING		NOTES
		ZERO	ONE	
1	-			
2	|			
3	|			
4	| Label 270			
5	|			
6	|			
7	|			
8	_|			
9	-			
10	|			
11	|			
12	| Space			
13	|			
14	|			
15	|			
16	-|			
17	TO P/B SW	NO SEL	SEL	* Bit Nos 17 and 22 provide the same information TO/GA
18	FLX TO P/B SW	"	"	
19	CL P/B SW	"		
20	CR P/B SW	"	"	
21	MCT P/B SW	"	"	
22	GA P/B SW	"	"	
23	-			
24	|			
25	| Spare			
26	|			
27	-|			
28	AUTO P/B SW	NO SEL	SEL	
29	-			
30	| SSM			
31	-|			
32	Parity			

TRP - Analog Inputs

SIGNAL	TYPE	GRADIENT	RANGE	POSITIVE SENSE	FUNCTION	NOTES
5 V PANEL DIM	AC		0 V -5Vrms		Night lighting brightness adjustment - markings - pushbutton switches - displays	

TRP - Input Discrete Signals

SIGNAL	STATUS		FUNCTION	NOTES
	ACTIVE	NOT ACTIVE		
LAMP TEST	+28 V	OC	Lighting of all mode selection pushbutton switches and display segments	Maximum intensity
TCC PS INH 1	GND	OC	TRP power supply cut off	If two TCCs are installed TCC PS INH 1 and 2 are required to cut off power
TCC PS INH 2	GND	OC	TRP power supply cut off	
TCC2 CONNECT	+28 V	OC	Detection of TCC2 installation	Generates 1 or 2 TCC condition for power supply cut off
TCC SELECT 1	+28 V	OC	TCC1 selected	Conditions ARINC line reading switching
TCC SELECT 2	+28 V	OC	TCC2 selected	
TEST PIN 1	GND	OC	Specific for bench maintenance	Spare
TEST PIN 2	GND	OC		
SPARE TCC 1 TO TRP	GND	OC	Spare	
SPARE TCC 2 TO TRP	GND	OC	Spare	
DIM DAY/NIGHT	GND	OC	Mode confirmation light level	GND maximum intensity
		OC = Open Circuit		

TRP - Output Discrete Signals

SIGNAL	STATUS		FUNCTION	NOTES
	ACTIVE	NOT ACTIVE		
Spare TRP TO TCC1	28 V	OC	Spare	
Spare TRP TO TCC2	28 V	OC	Spare	

SIGNAL	STATUS		FUNCTION	NOTES
	ACTIVE	NOT ACTIVE		

E. Lighting

(1)Night lighting : markings

- the markings identifying the modes are illuminated by lamps integrated in the panel face.

(2)Lighting control

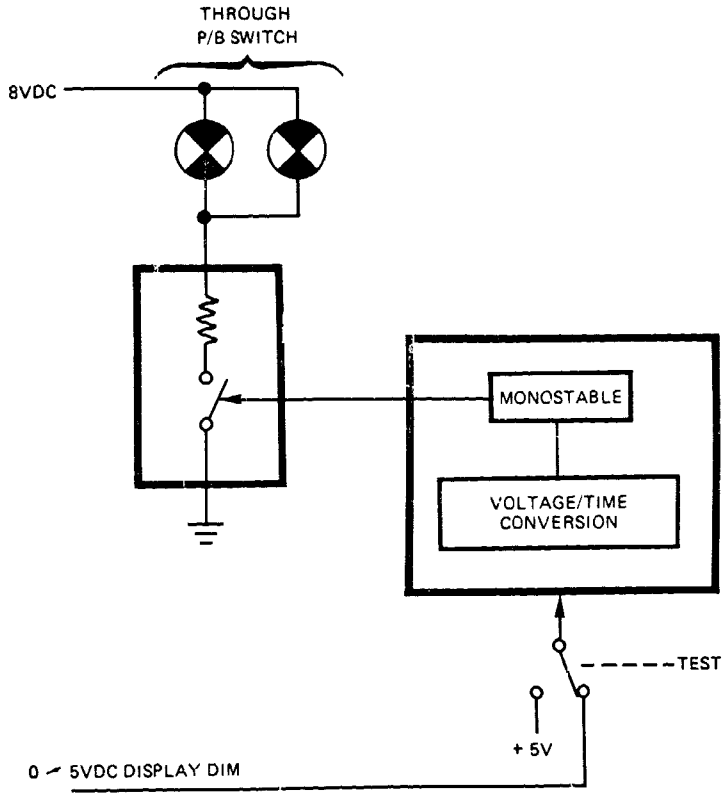
- variation in the lighting of the mode selection pushbutton switch bulbs is obtained by varying the power application time.
- the night lighting intensity (markings and displays) is controlled using the 5VAC power signal amplitude serving for panel lighting.

(Ref. Fig. 003)

F. Integrity

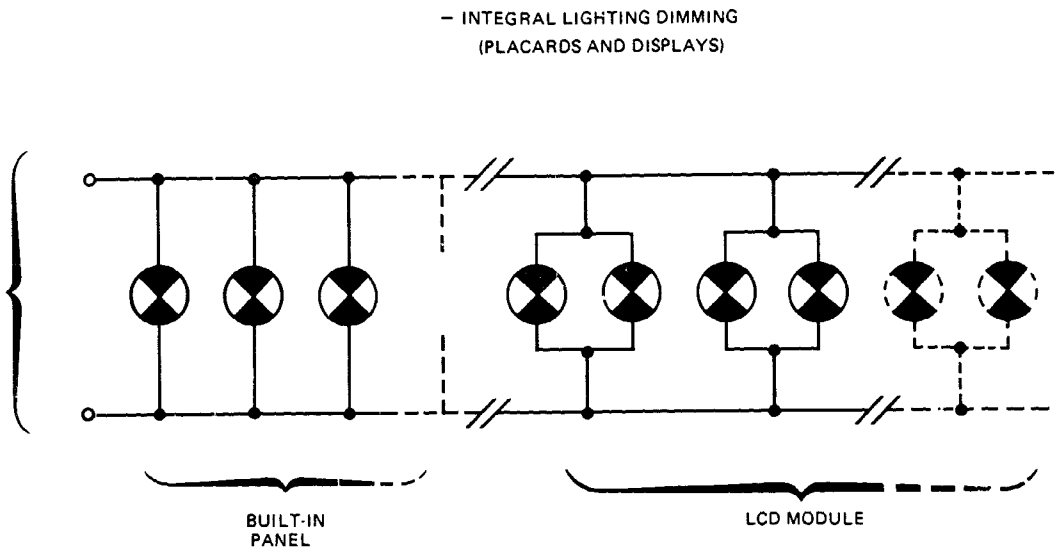
- The TRP circuitry monitoring is mainly achieved by a watchdog circuit : each untimely deviation of the program causes a monostable to trigger which results in display interruption and generation of ARINC lines (TRP interruption : the TRP inhibits its power supply).
- TCC - TRP ARINC interface is checked by the TRP by comparing the flexible takeoff temperature value transmitted by the TRP and the same value sent back by the TCC.
- TRP - TCC
  ARINC interface is checked by the TCC by comparing the N1 limit value transmitted by the TCC and the same value (limit thrust) sent back by the TRP.

(Ref. Fig. 004)



BM5 22 35 00 0 AXM2

Pushbutton Switch Lighting (Mode Confirmation)
Figure 003



BM5 22 35 00 0 AZM2

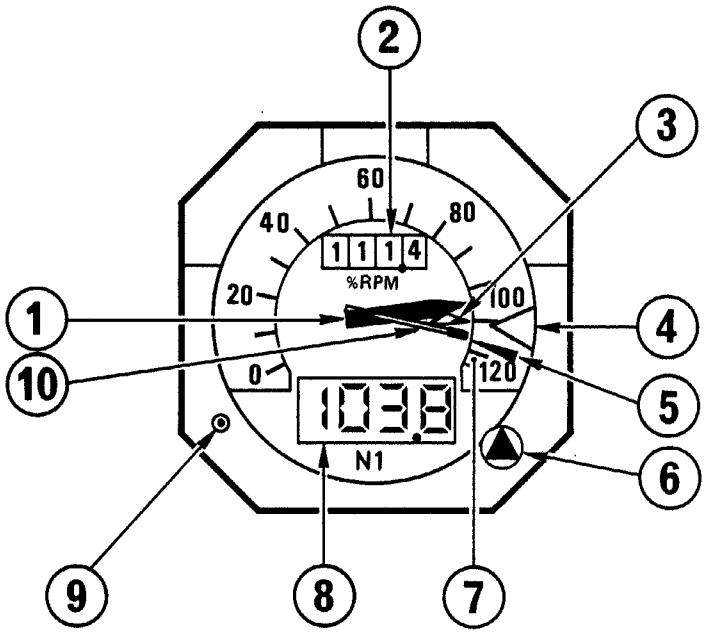
Integral Lighting Dimming (Placards and Displays)
Figure 004

2. N1 Limit Recopy on N1 Indicators - N1 Limit Manual Setting

A. N1 Indicator
(Ref. Fig. 005)

B. N1 Signal Processing
The N1 command and the N1 actual are computed in the PMCs and then sent to the indicators.

C. N1 Limit Information
(1)Automatic mode
The automatic mode is selected by pushing knob (6) on the N1 indicator. The upper counter (2) is covered by a flag. Each TCC supplies a label 342 on its bus and in ARINC 429 form which serves for bug (4) control.
(2)Manual mode
The manual mode is selected by pulling knob (6) on the N1 indicator. The flag in upper counter (2) goes out of view. The N1 limit signal delivered by the TCC is replaced by a signal generated in the indicator which represents the N1 limit value manually selected by the pilot.
When the N1 actual signal amplitude (with phase advance to avoid exceeding values) reaches the reference value selected by the pilot a comparator indicates that engine speed is exceeded and that the throttle control levers are stopped (operating speed does not exceed red line (5)).
The throttle control levers are stopped as long as the N1 actual signal value is greater than the N1 limit.
A BITE test can be performed from the indicator by push action (9). If the indicator is operational, this action causes positioning of N1 actual (1), N1 command (3) and overspeed (10) pointers and bug (4) at 121.4 % N1. This value is also displayed in N1 actual counter (8) (test value 121.4 % N1 corresponding to blue dot (7) on the indicator scale).



1. N1 ACTUAL POINTER (WHITE)
   INDICATES ENGINE N1 CALCULATED BY THE PMC.
2. DIGITAL COUNTER
   INDICATES N1 LIMIT (MANUAL MODE ONLY).
3. N1 COMMAND POINTER (BLACK AND WHITE)
   INDICATES THE N1 CALCULATED BY THE PMC ACCORDING TO THE THROTTLE SETTING.
4. BUG
   INDICATES THE N1 LIMIT.
5. RED LINE
   INDICATES MAXIMUM PERMISSIBLE OPERATING SPEED
6. KNOB
   USED TO SELECT MANUAL OR AUTOMATIC N1 LIMIT CALCULATION MODE
7. BLUE DOT
   AT 121.4 %N1 FOR INDICATOR BITE TEST.
8. DIGITAL COUNTER
   INDICATES ACTUAL ENGINE N1 CALCULATED BY THE PMC
9. BITE PUSHBUTTON SWITCH
   USED TO TEST INDICATOR OPERATION (WHEN PUSHED)
10. OVERSPEED POINTER (BACK AND RED END)
    NORMALLY IN FRONT OF RED LINE IT WILL INDICATE THE MAXIMUM VALVE OBTAINED AND WILL REMAIN IS THIS POSITION.

BM5 22 35 00 0 GAM2 24.19
DM5

N1 Indicator
Figure 005

3. TCCS Indicating and Warnings on EFIS Displays

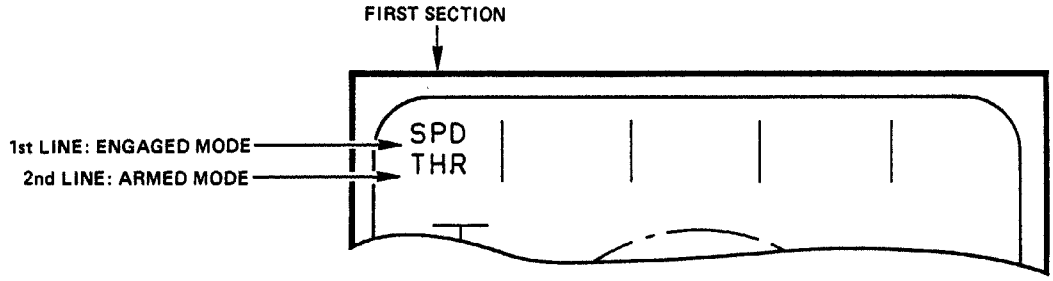
Indicating and warnings are generated by the EFIS SGUs (Symbol Generator Units) which receive the ARINC 429 output data bus from the TCC.

A. TCCS Indicating on EFIS Displays

Information concerning armed or engaged mode appears in the upper part of the Primary Flight Display (PFD) in the first section.
A detailed description of the displays appearing on the PFD is given in the following figure.

B. Warnings

A warning display associated with ATS operation can appear on the PFD : MAN THR : ATS declutching (loss of A/THR or THR LATCH). MAN THR disappears as soon as A/THR or THR LATCH mode is engaged.
(Ref. Fig. 006).



	DISPLAY	COLOR	MEANING
1st LINE	SPD	GREEN	SPEED SELECTION MODE
	MACH	GREEN	MACH SELECTION MODE
	THR	GREEN	N1 MODE ENGAGED
	RETARD	GREEN	THROTTLE RETARD
	P.THR	GREEN	N1 MODE ENGAGED THROUGH FMC (TARGET MODE)
	P.SPD	GREEN	SPEED MODE ENGAGED THROUGH FMC
	P.MACH	GREEN	MACH MODE ENGAGED THROUGH FMC
	THR/L (FLASHING)	GREEN	N1 LATCH MODE ENGAGED (NO ALPHA FLOOR CONDITION)
	THR/L	GREEN	N1 LATCH MODE ENGAGED (THROUGH ALPHA FLOOR CONDITION)
2nd LINE	A/THR	CYAN	A/THR MODE ARMED
	THR	CYAN	AUTOTHROTTLE DECLUTCH AFTER TO
	A/THR	CYAN	N1 MODE ARMED (LIMIT MODE)
	THR	CYAN	N1 DESCENT MODE
	MAN THR	AMBER	AUTOTHROTTLE DECLUTCHING

BM5 22 35 00 0 BCP2 24 19
DM5

ATS Mode Indicating on the PFD
Figure 006

4. ATS OFF Warning

The loss of the ATS1 or ATS2 (or ATS1 and 2) systems is indicated by:
- the ATS lever(s) returning to the OFF position
- the ATS1 or 2 (or ATS1 and 2) OFF message on the left ECAM display unit and the MASTER CAUTION lights coming on on the main instrument panels 3VU and 5VU.

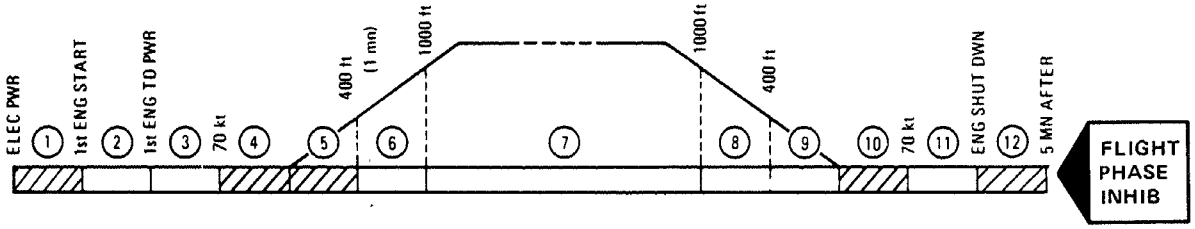
NOTE : The MASTER CAUTION lights come on only when both ATS are lost : ATS 1 +2 OFF message on left ECAM display unit.

The following figures show the flight phases during which warnings are active and give the messages displayed on the left ECAM display unit.
(Ref. Fig. 007)
(Ref. Fig. 008)
(Ref. Fig. 009)

The warnings are cancelled :
- after successful rearming of the system
- by pressing the CLR pushbutton switch on the ECAM control panel
- by pressing the MASTER CAUTION pushbutton switch.

(Ref. Fig. 010)

FAILURE : ATS	DETECTOR :	AURAL WARNING
ATS DISARMED	ATS LEVER	SC
R ECAM DU SYS PAGE CALLED :		MASTER CAUTION
NIL		MASTER CAUTION A

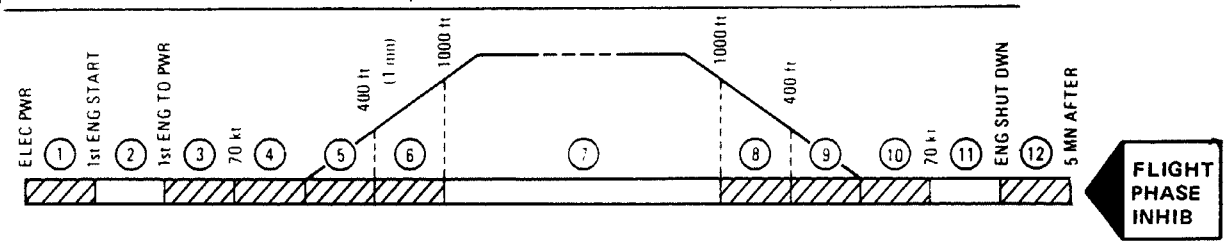


ACTIONS		LOCAL WARNING	WARNING LIGHTS	L ECAM DISPLAY UNIT	
		ATS LEVER IN OFF POSITION	MASTER CAUTION	ATS ATS 1 OFF .....RESET	A A-C
If unsuccessful to reset : Cancel	CLR	Id	nil	STATUS LAND 3 INOP ATS ALPHA FLOOR INOP	A A
If recall	RCL	Id		ATS ATS 1 OFF .....RESET	A A-C

BM5 22 35 00 0 TDP2-24-19
DM5

ATS Disarmed
Figure 007

FAILURE : ATS	DETECTOR :	AURAL WARNING
ATS 1 OR 2 DISARMED	ATS LEVER	NIL
R ECAM DU SYS PAGE CALLED :		WARN LIGHT
NIL		NIL

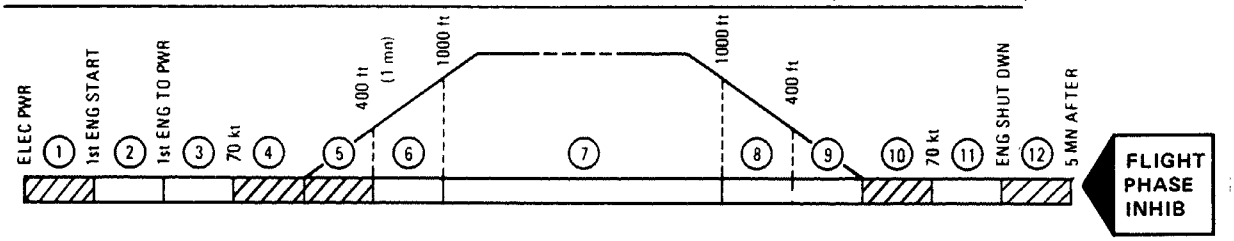


ACTIONS		LOCAL WARNING	WARNING LIGHTS	L ECAM DISPLAY UNIT	
		ATS LEVER IN OFF POSITION	nil	ATS ATS 1 OFF .....RESET (or 2)	A A-C
If unsuccessful to reset : Cancel	CLR	id	nil	nil	
If recall	RCL	id	nil	ATS ATS 1 OFF .....RESET (or 2)	A A-C

BM5 22 35 00 0 TDM2 - 17 - 12
DM5

ATS - ATS 1 or 2 Disarmed
Figure 008

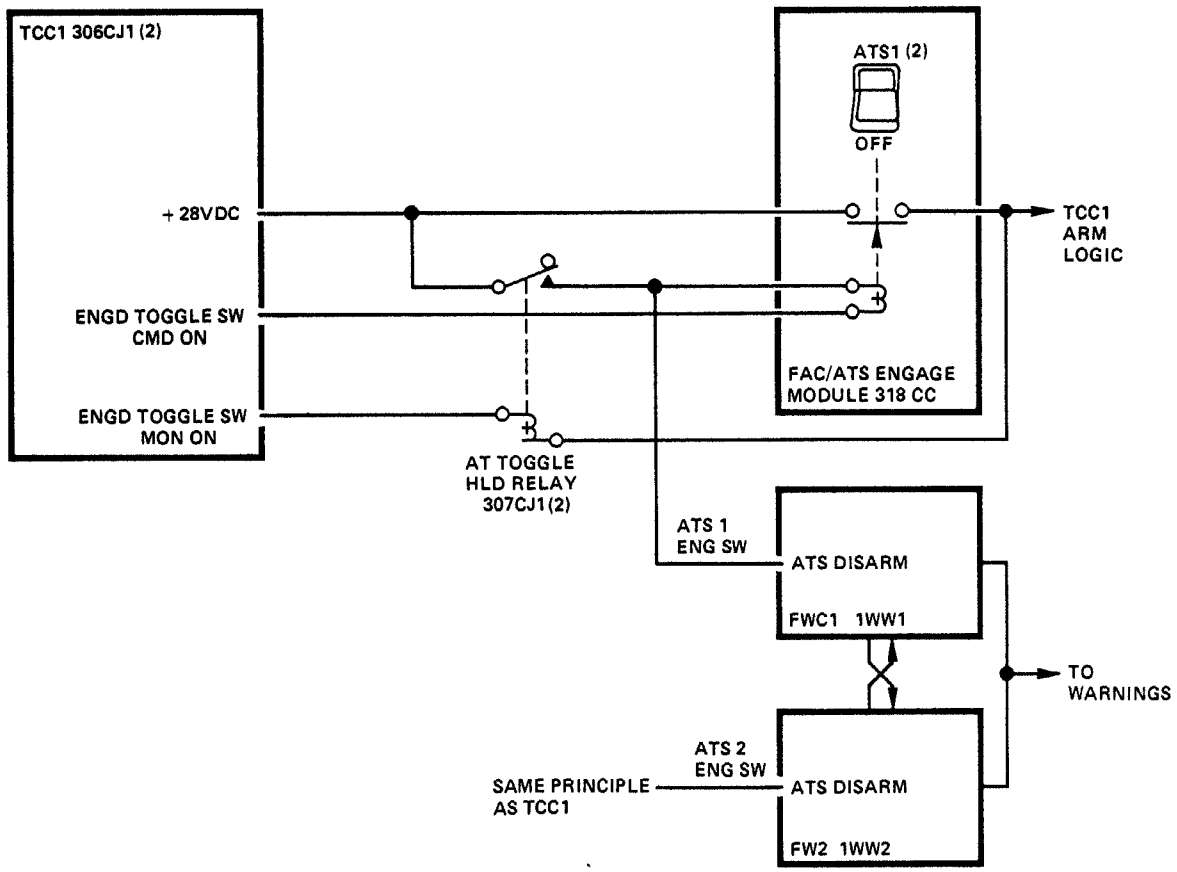
FAILURE : ATS	DETECTOR :	AURAL WARNING
ATS 1 + 2 DISARMED	ATS LEVERS	SC
IR ECAM DU SYS PAGE CALLED :		MASTER CAUTION
		MASTER CAUTION A
NIL		



ACTIONS		LOCAL WARNING	WARNING LIGHTS	L ECAM DISPLAY UNIT	
		BOTH ATS LEVERS IN OFF POSITION	MASTER CAUTION	ATS ATS 1 OFF .....RESET ATS 2 OFF .....RESET	A A-C A-C
If unsuccessful to reset : Cancel	CLR	ld	nil	STATUS LAND 3 INOP ATS ALPHA FLOOR INOP	A A
If recall	RCL	ld		ATS ATS 1 OFF .....RESET ATS 2 OFF .....RESET	A A-C A-C

BM5 22 35 00 0 TDN2-17-12
DM5

ATS - ATS 1 + 2 Disarmed
Figure 009



BM5 22 35 00 0 BEN2 24 19
DM5

Discretes Used in ATS OFF Warning Generation
Figure 010

THRUST RATING PANEL (303CJ) - REMOVAL/INSTALLATION

1. Thrust Rating Panel

A. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Blanking Caps
Referenced Procedure - 22-00-00, P. Block 501 (Ref. Fig. 401)	Auto Flight

B. Procedure

(1)Job set-up

(a)Open, safety and tag the following circuit breakers :

	PANEL	SERVICE	IDENT.	LOCATION
R	**ON A/C 101-200,			
	21VU	AFS/TCC/SYS1/28VDC	306CJ1	102/H8
	21VU	AFS/TRP/PWR/SUPPLY	36CJ	102/H14
R	**ON A/C 001-100,			
	21VU	AFS/TCC/SYS 1/28VDC	306CJ1	102/H8
	21VU	AFS/TCC/SYS 2/28VDC	306CJ2	102/H12
	21VU	AFS/TRP/PWR/SUPPLY	36CJ	102/H14
R	**ON A/C ALL			
	132VU	INTEGRAL LT/MAIN/PNL &/GLARE SHLD	1LF	322/N76
	132VU	AFS CONTROL PANELS/FCU & TRP & MTP	313CA1	325/K77
	132VU	INT LT/ANN LT/TEST/MAN	149LP	324/L82

(2)Removal

(a)Release retaining screws (9) on clamp (2). Do not remove screws.

(b)Pull out TRP (4).

(c)Disconnect electrical connector (3).

(d)Cap electrical connectors (3) and (8).

(e)Remove thrust rating panel (4).

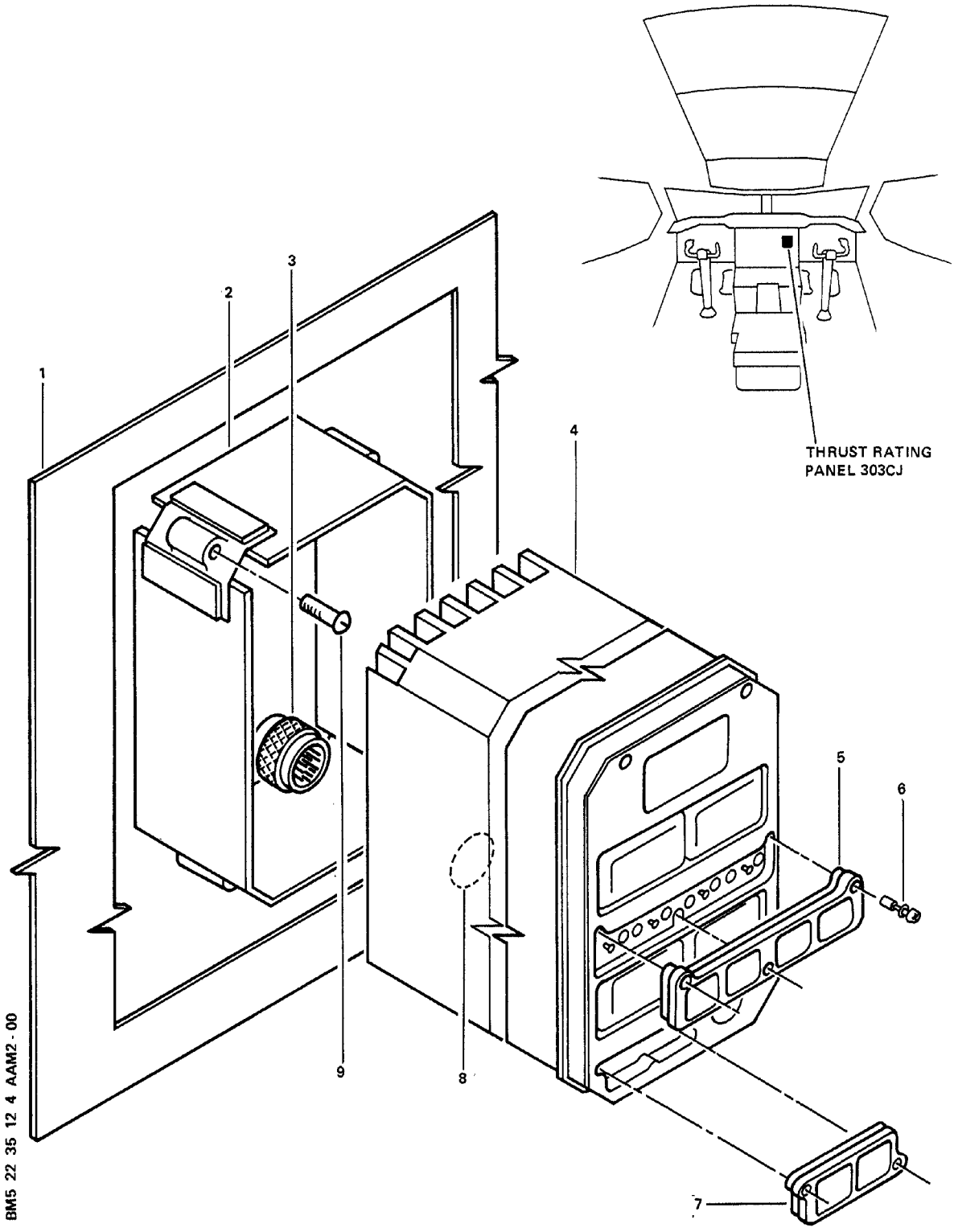
(3)Installation

(a)Clean and inspect thrust rating panel (4) adjacent area or interface.

(b)Remove blanking caps from electrical connectors (3) and (8).

(c)Connect electrical connector (3) on thrust rating panel (4).

(d)Position thrust rating panel (4) in clamp (2) and tighten screws (9).



Thrust Rating Panel
Figure 401

R **ON A/C 101-200,

(e)Remove safety clips and tags and close circuit breakers 36CJ, 306CJ1, 1LF, 313CA1 and 149LP.

R **ON A/C 001-100,

(e)Remove safety clips and tags and close circuit breakers 36CJ, 306CJ1, 306CJ2, 1LF, 313CA1 and 149LP.

R **ON A/C ALL

(4)Test
Carry out AFS light test and display test (Ref. 22-00-00, P. Block 501, paragraph 1. B.)
(5)Close-up
Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

2. Pushbutton Switch Housing

NOTE : There are two pushbutton switch housings on TRP. Each housing must be removed without removing the thrust rating panel.
Removal/Installation procedure is the same for each housing.

A. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Blanking Caps
Referenced Procedure - 22-00-00, P. Block 501	Auto Flight

B. Procedure

(1)Job set-up

(a)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
R **ON A/C 101-200,			
21VU	AFS/TCC/SYS1/28VDC	306CJ1	102/H8
21VU	AFS/TRP/PWR/SUPPLY	36CJ	102/H14
R **ON A/C 001-100,			
21VU	AFS/TCC/SYS1/28VDC	306CJ1	102/H8
21VU	AFS/TCC/SYS2/28VDC	306CJ2	102/H2
21VU	AFS/TRP/PWR/SUPPLY	36CJ	102/H14
R **ON A/C ALL			
132VU	INTEGRAL LT/MAIN/PNL &/GLARE SHLD	1LF	322/N76
132VU	AFS CONTROL PANELS/FCU & TRP & MTP	313CA1	325/K77
132VU	INT LT/ANN LT/TEST/MAN	149LP	324/L82

(2)Removal

(a)On face of TRP remove set screw (6) associated with housing.

(b)Pull out and remove housing (5) or (7).

(3)Installation

(a)Clean and inspect housing (5) or (7).

(b)Position housing on face of TRP and tighten screws (6).

R **ON A/C 101-200,

(c)Remove safety clips and tags and close circuit breakers 36CJ, 306CJ1, 1LF, 313CA1 and 149LP.

R **ON A/C 001-100,

(c)Remove safety clips and tags and close circuit breakers 36CJ, 306CJ1, 306CJ2, 1LF, 313CA1 and 149LP.

R **ON A/C ALL

(4)Test
Carry out AFS light test and display test (Ref. 22-00-00, P. Block 501, paragraph 1. B.).

(5)Close-up
Make certain that working area is clean and clear of tools and miscellaneous items of equipment.

1. Thrust Control Computer (TCC) - Description

A. Characteristics

(1) Dimensions

The computer is 8 MCU size equipped with a size 2 connector, ARINC 600 type.

(2) Weight

Computer weight is less than 12.5 kg.

(3) Current consumption

The permanent consumption of a TCC shall be of approximately 2A with 28VDC supply and of 60 mA with 115VAC/400 Hz supply (power stage for autothrottle actuator).

(4) Cooling

The computers are designed to utilize cooling air as specified in ARINC Specification 600. Operation of the TCC without cooling shall be possible with a subsequent reduction in MTBF.

B. Computer Built-In Safety Devices

R The functions associated with the power loop monitoring (servo image, comparator) are performed by the software.

Each processor monitors separately through analog inputs :

- the computation command in the command channel (voter command input) against the command generated by the monitoring channel
- the motor voltage from the power stage against a software - generated image (monitoring processor).

Each processor provides two booleans representing the comparator status through discrete outputs.

C. Hardware and Software Breakdown

The signals processed in the TCC are given for each component together with the type of processing which is assigned to them.

Hardware and software brekdown is shown for the command and monitoring channels in the following figure.

(Ref. Fig. 001)

2. EPR Limit Computation

The EPR limit computation is shown in figure.

Computation is made by using the following data.

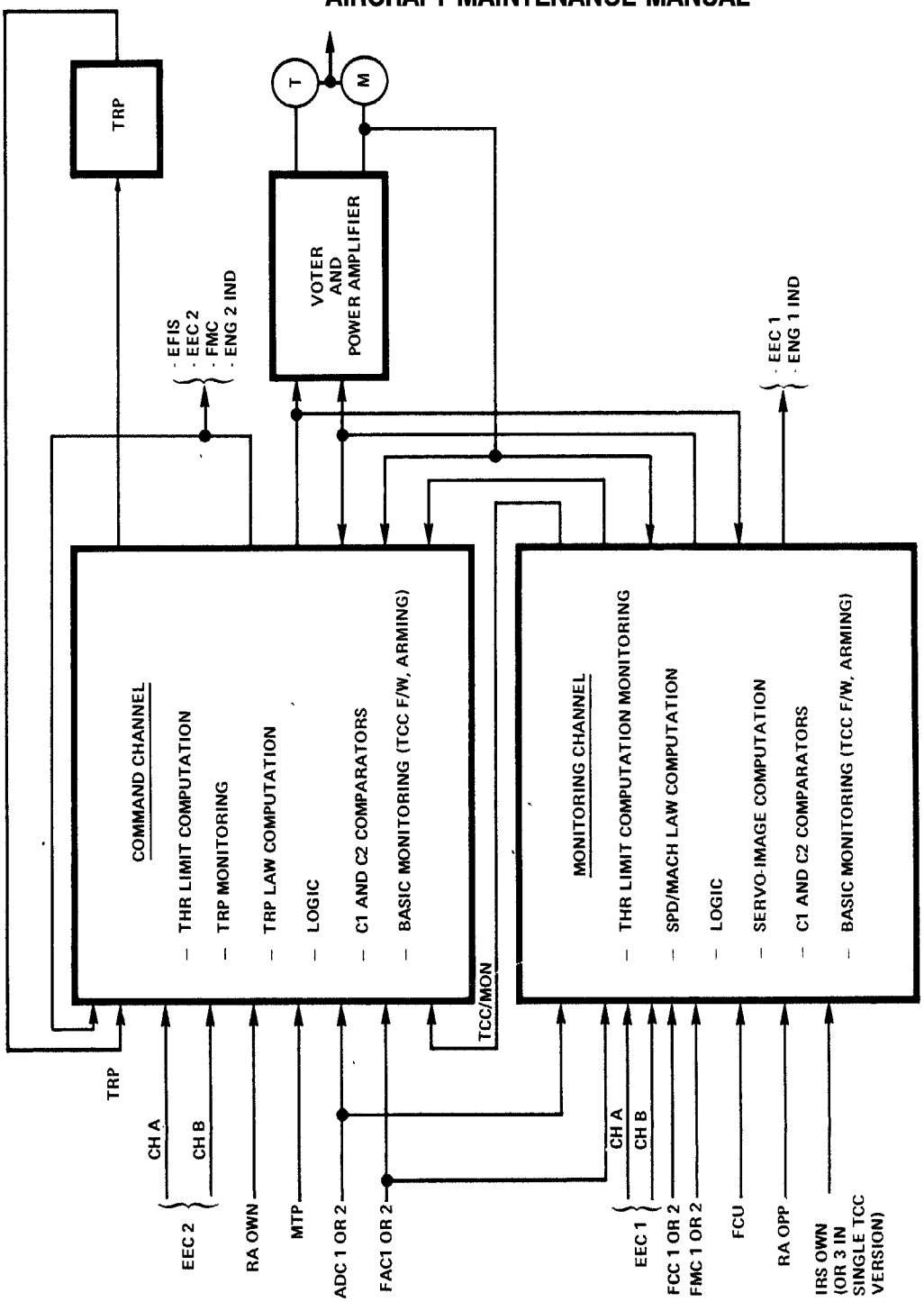
A. ADC1 or ADC2 Data

- Total pressure (TOT PRESS)
- Mach number (M)
- Altitude (ALT)
- Total air temperature (TAT).

B. TRP Data

The TRP gives the mode selected by the crew (THR modes).

C. Bleed Valves



BM5 22 36 00 0 BAN2 24 19
DM5

Principle of TCC Architecture
Figure 001

D. Engine Type Reference
The first computation block acquires the following parameters : ADC, TRP, bleed valves and engine reference.
The processed data are sent to the following blocks :
- computation of T2 breakpoint and standard temperature
- data selection block for takeoff and go around modes
- maximum continuous mode computation
- climb mode computation
- cruise mode computation.

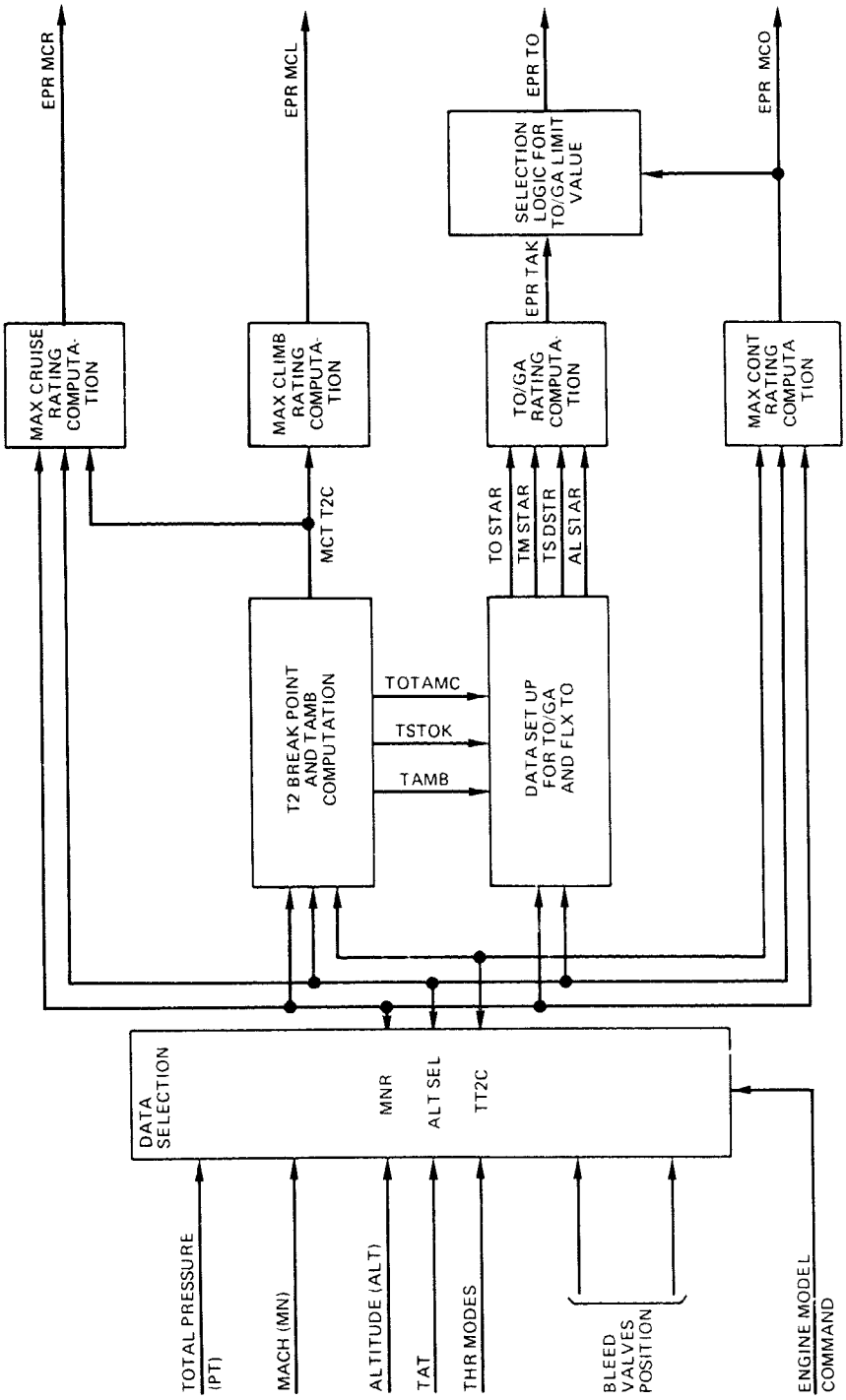
E. Breakpoint Computation Block
This block computes the breakpoint temperature from the following parameters :
- altitude (ADC1 or ADC2) (ALT SEL)
- TAT (ADC1 or ADC2) (Tt2C)
- Mach number (ADC1 or ADC2) (MNR)
- engine reference.

The output parameters are :
- ambient temperature (^{o}C) (TAMB)
- ambient temperature for takeoff (ambient temperature corrected in relation to altitude)
- standard temperature (TSTDK)
- temperature for the computation of CLIMB and CRUISE modes (correction in relation to Mach number)
- computation correction due to air bleeds

F. TO and GA Selection Logic
TO/GA selection is made by means of a single pushbutton switch on the TRP. The TO EPR limit is selected if the landing gear shock absorbers are compressed. If this is not the case, the GA limit appears.
Computation in TO or FLX TO mode utilizes Mach and altitude values from which the EPR limit values are frozen. For GA mode, computation continues evolving.
(Ref. Fig. 002)
(Ref. Fig. 003)
(Ref. Fig. 004)
(Ref. Fig. 005)
(Ref. Fig. 006)

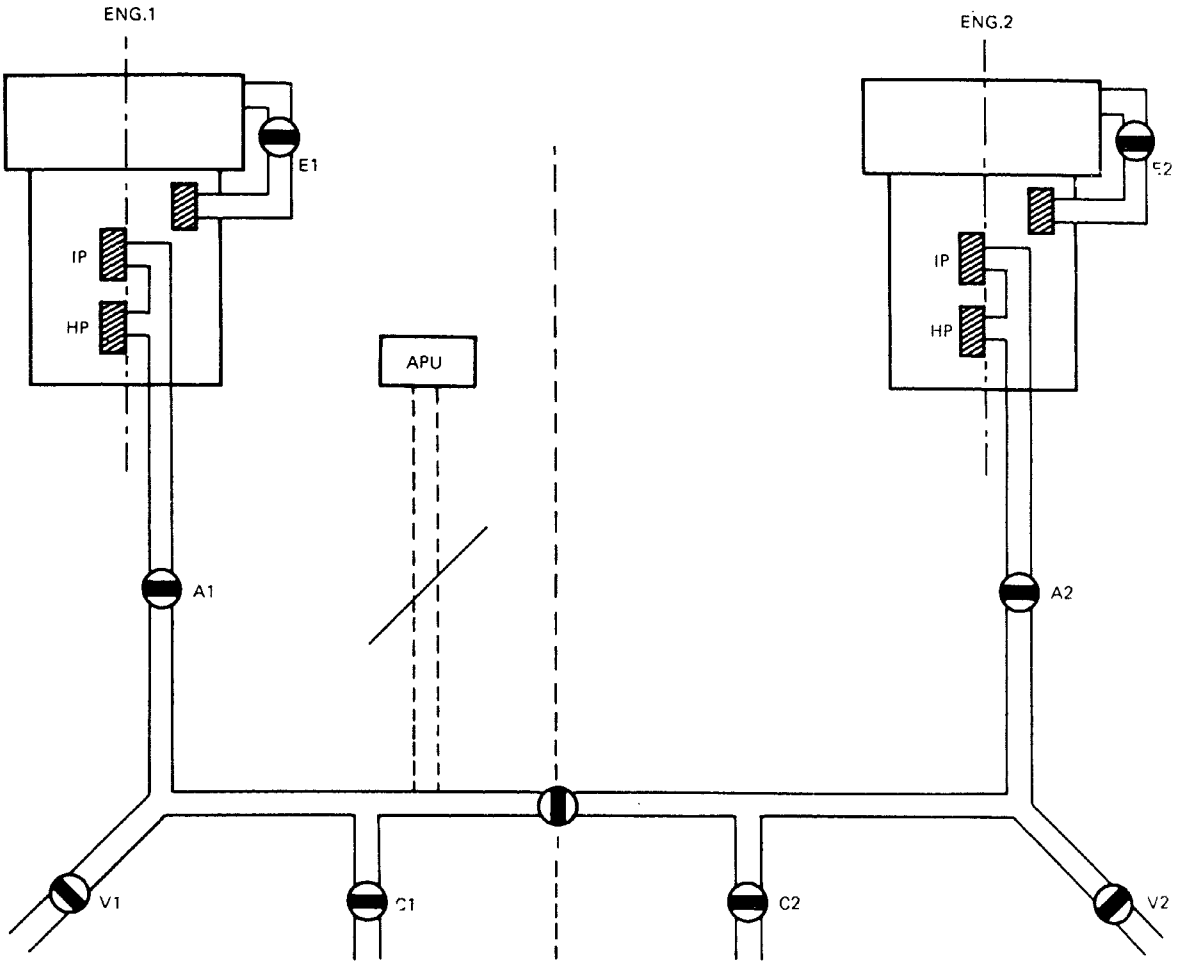
G. Engine Failure Configuration
In an engine failure configuration, the engine remaining valid is overheating according to the anti ice air bleed valve state.
The provided logic ensures a thrust overshoot of 0.008ΔEPR function of temperature and altitude.
The solution consists in taking into account the HP fuel shut-off valve cut-off in series with the air bleed valve cut-off condition to avoid EPR thrust overshoots.

H. Bleed Definition
Only five decrements ΔEPRi are defined. The decrements are defined as polynomials in function of altitude (feet).



DM5 22 36 00 0 FAK2 08 03
BM5

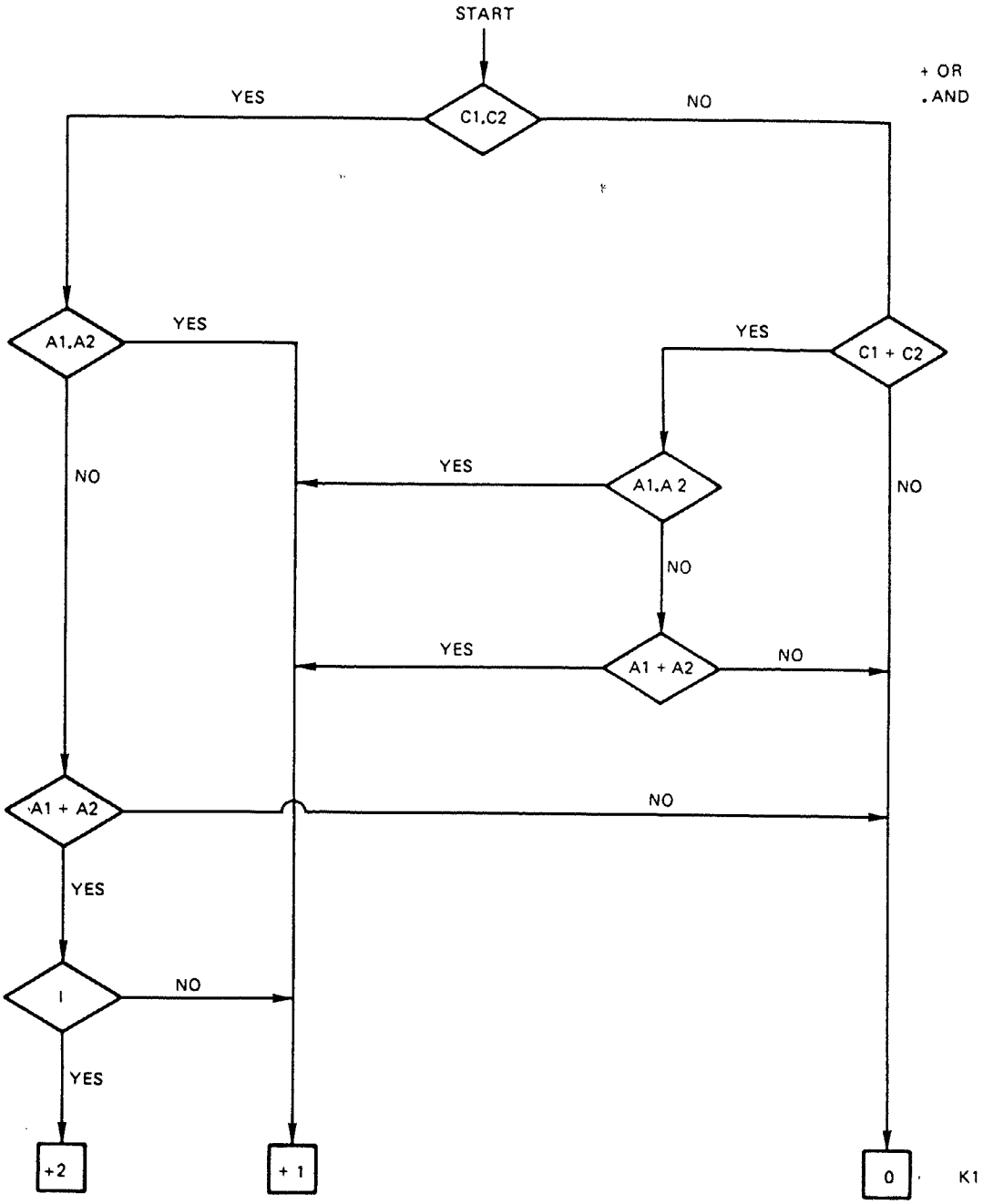
EPR Limit Overview Diagram
Figure 002



A1/A2	ENGINE BLEED VALVE
E1/E2	NACELLE ANTI-ICE (NAI)
V1/V2	WING ANTI-ICE (WAI)
C1/C2	AIR CONDITIONING RACK (A/C)
A	CROSSFEED VALVE

BM5 22 36 00 0 FWM2 08 03
DM5

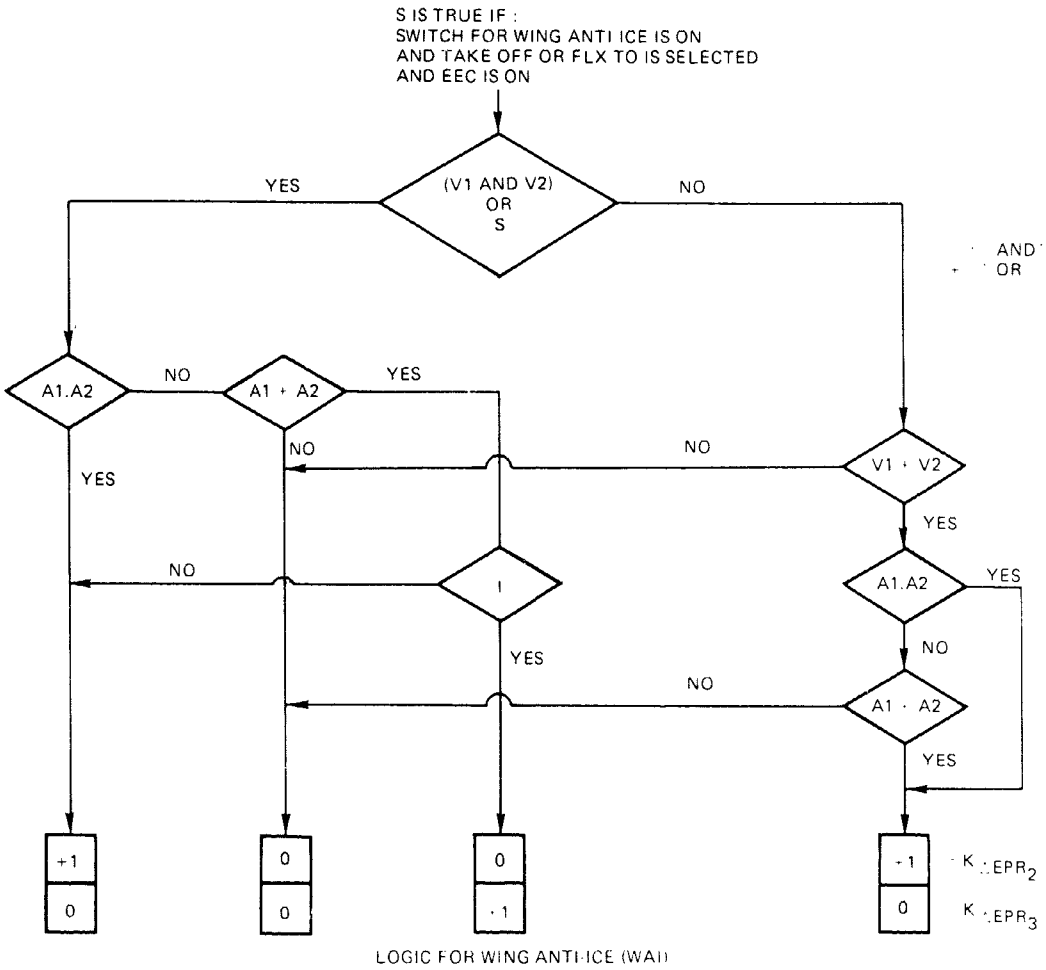
Bleed Valve Configuration
Figure 003



K1 CORRECTION GAIN CORRESPONDING TO AIR BLEED FOR AIR CONDITIONING

BM5 22 36 00 0 FXM2 08 03
DM5

Logic for Air Conditioning
Figure 004



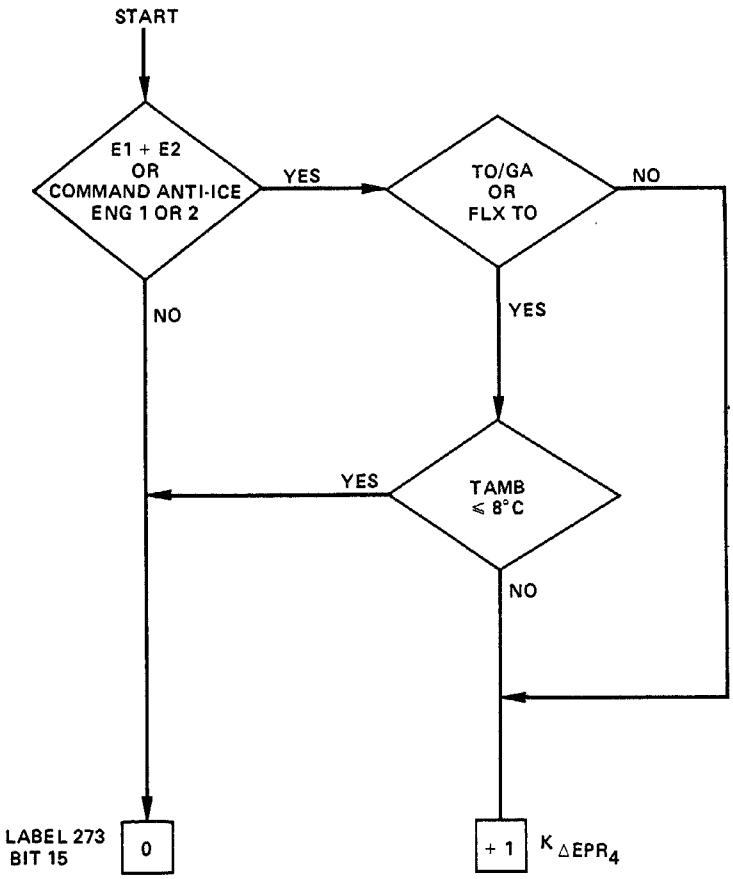
LOGIC FOR WING ANTI-ICE (WAI)

$K_{\Delta EPR_2}$: CORRECTION GAIN CORRESPONDING TO AIR BLEED FOR HALF WING ANTI-ICING

$K_{\Delta EPR_3}$: CORRECTION GAIN CORRESPONDING TO AIR BLEED FOR FULL WING ANTI ICING

BM5 22 36 00 0 FYX2 08 03
DM5

Logic for Wing Anti Ice
Figure 005



$K_{\Delta EPR\,4}$: CORRECTION GAIN CORRESPONDING TO AIR BLEED FOR AIR INTAKE ANTI ICING (COWL)

BM5 22 36 00 0 BCM2 21

Logic for Air Intake Anti Icing
Figure 006

The associated bleed valve logic is given in following figure.

THRUST RATING COMPUTING SYSTEM - DESCRIPTION AND OPERATION

1. Thrust Control Computer (TCC) - Description

A. Characteristics

(1) Dimensions

The computer is 8 MCU size equipped with a size 2 connector, ARINC 600 type.

(2) Weight

Computer weight is less than 12.5 kg.

(3) Current consumption

The permanent consumption of a TCC shall be of approximately 2A with 28VDC supply and of 60 mA with 115VAC/400 Hz supply (power stage for autothrottle actuator).

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Each processor provides two booleans representing the comparator status through discrete outputs

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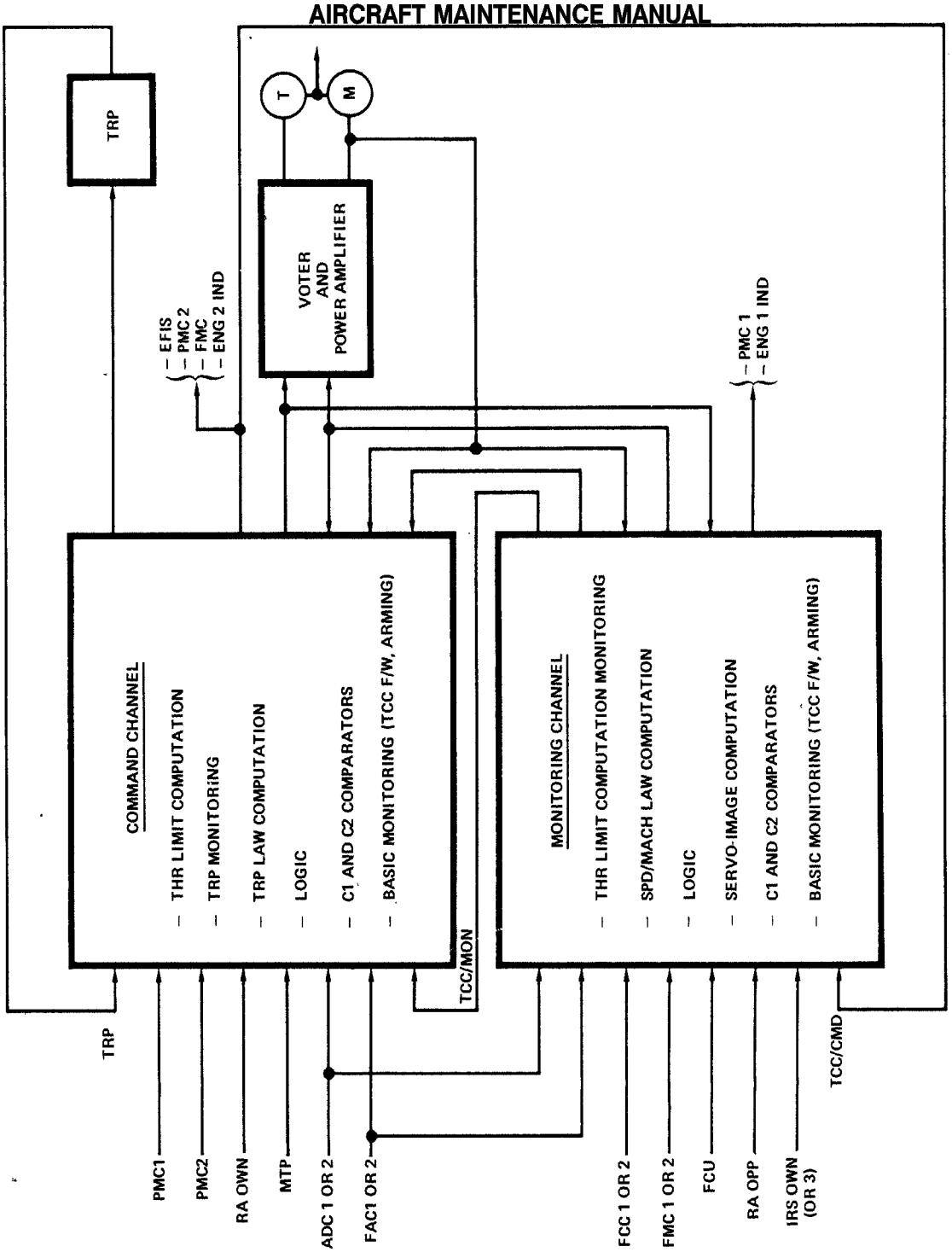
(Ref. Fig. 001).



BM5 22 36 00 0 THL2 22 16

Principle of TCC Architecture
Figure 001

2. N1 Limit Computation

A. General

(1) Introduction

Either GE or PW engine has a control system which incorporates all the hydromechanical elements plus a digital engine control.
The digital engine control system provides simplified engine operation through thrust management and improved communication with the pilot and aircraft systems.

(2) N1 limit computation (TCC)

The N1 limit computation is based on the information given by other systems (ADC, bleed valve positions) and the flight phase either selected on the thrust rating panel (TRP) or automatically selected by other systems. The TCC computes an N1 value corresponding to the given engine limitations.
This value is transmitted to the N1 indicator to drive the N1 limit bug.

(3) Mode selection and display (TRP)

Different modes can be selected and displayed on the TRP :

- TO (Maximum Takeoff)
- FLX TO (Flexible Takeoff)
- CL (Maximum Climb)
- CR (Maximum Cruise)
- MCT (Maximum Continuous)
- GA (Go Around) (The same pushbutton switch is used for TO and GA).

In addition, the TRP is used to display the :

- N1 limit
- Total Air Temperature (TAT)

and to select the fictitious temperature for the FLX TO mode.

B. Rating Computation

For rating computations, the TCC requires the following aircraft measurements :
- MN, Mach Number from ADC1 or ADC2
- TAT, Total Air Temperature from ADC1 or ADC2
- PT, total pressure input | PO (Static Pressure) = PT - Q
- Q, impact pressure input |
- Bleeds Status

For takeoff rating computation, the TCC requires also the status of the PMC (Off or On).

(1)Bleed corrections
The total bleed correction is the sum of elementary corrections according to altitude, whose gain is determined by means of a logic associated with the position of bleed or anti-ice valves.
The correction gain is defined by a valve logic associated with air bleed conditions and wing anti-ice or by the position of the anti-ice valves.

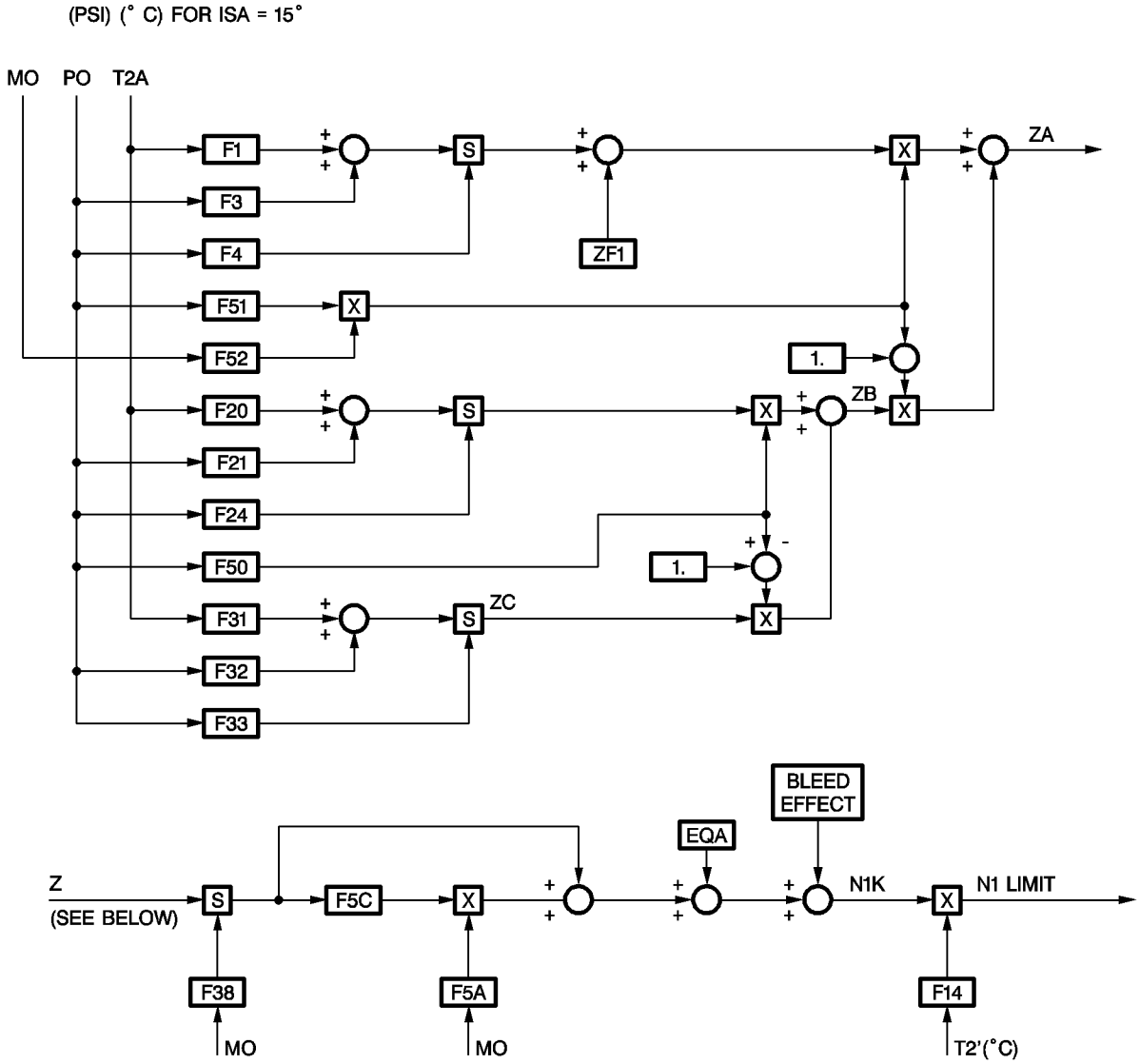
(2)Engine failure configuration
In order to avoid a thrust overshoot on the valid engine, the engine HP fuel shut-off valve cutoff in series with the air bleed cutoff condition is taken into account in the logic.

(3)TCC lightning protection (All links between PMC-TCC and Bleed Valves)

(a)Two-wire shielded twisted circuits
(with shielding correctly grounded in several points : ARINC 429)
- Induced voltage through common mode
  300 V - 10 µs, rise time 1 µs

(b)Airframe return circuits (short circuit or 28 V)
- Induced voltage through flux linkage
  ±2000 V - 10 µs, rise time 1 µs.
  Impedance : 6 Ω
  Rate : 5 cyc/mn

(Ref. Fig. 002)
(Ref. Fig. 003)
(Ref. Fig. 004)
(Ref. Fig. 005)



FOR MAX TAKE-OFF/GO-AROUND : Z=ZA
FOR MAX CONTINUOUS : Z=ZB
FOR MAX CLIMB : Z=ZC

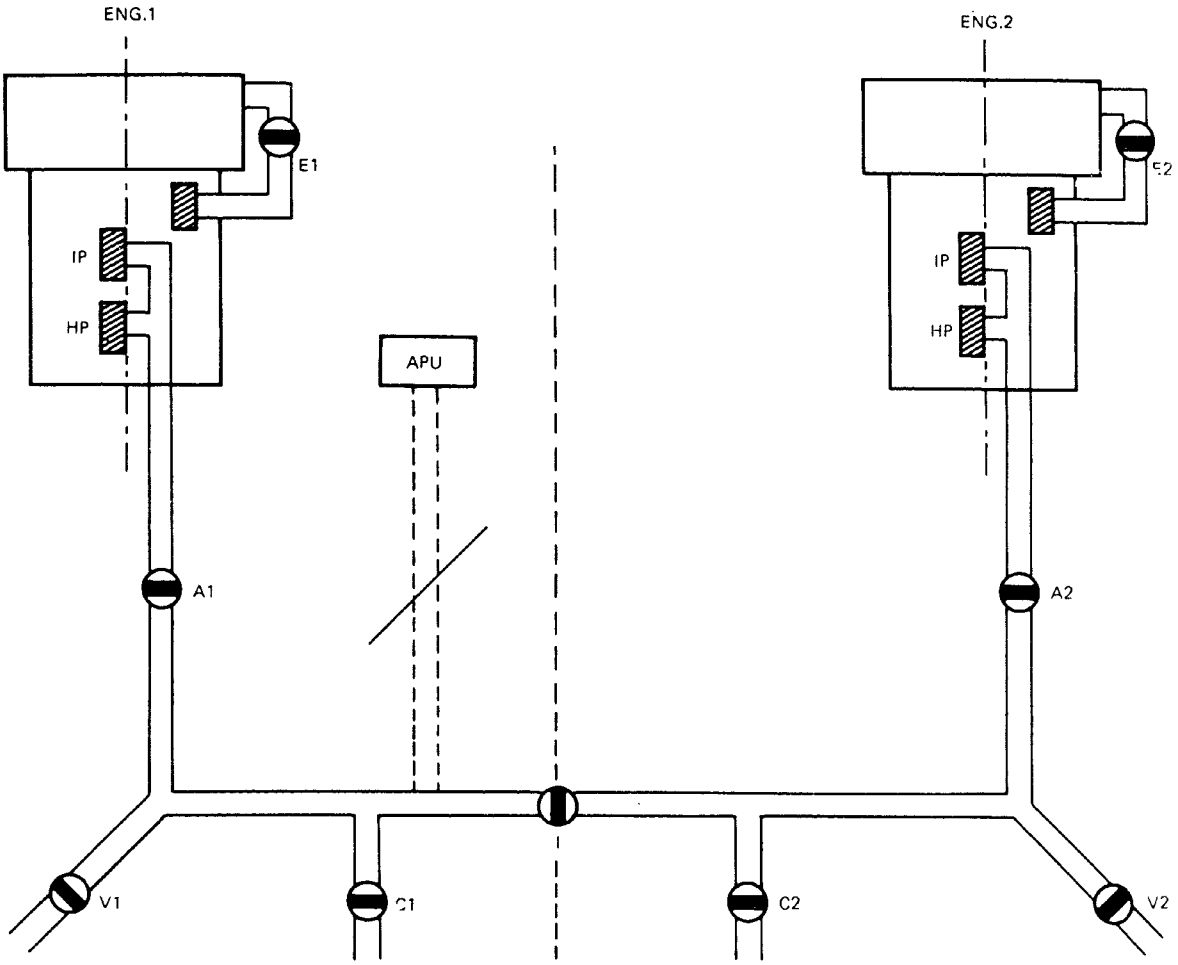
WITH $T2A = \frac{(T2 + 273.15) \times 4.9973}{4.9873 + MO^2} - 273.15$

FOR FLEXIBLE TAKE-OFF : Z=ZD WITH T2A DEFINED ON FLOWCHART 1.1
EQA - ENGINE QUALITY ADDER

FLOWCHART 1

BM5 22 36 00 0 FRN2 00

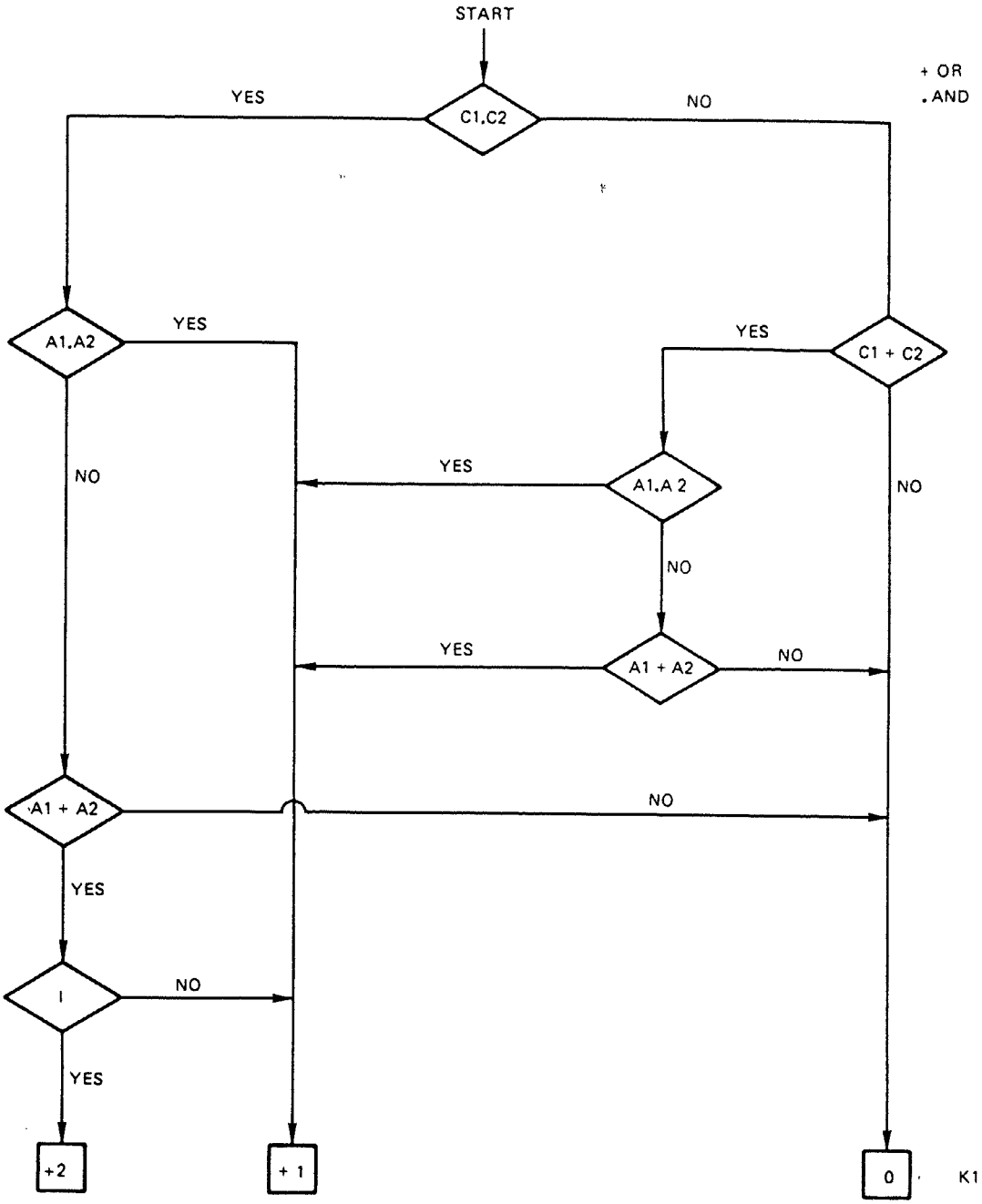
TCC N1 Limit Computation
Figure 002



A1/A2	ENGINE BLEED VALVE
E1/E2	NACELLE ANTI-ICE (NAI)
V1/V2	WING ANTI-ICE (WAI)
C1/C2	AIR CONDITIONING RACK (A/C)
A	CROSSFEED VALVE

BM5 22 36 00 0 FWM2 08 03
DM5

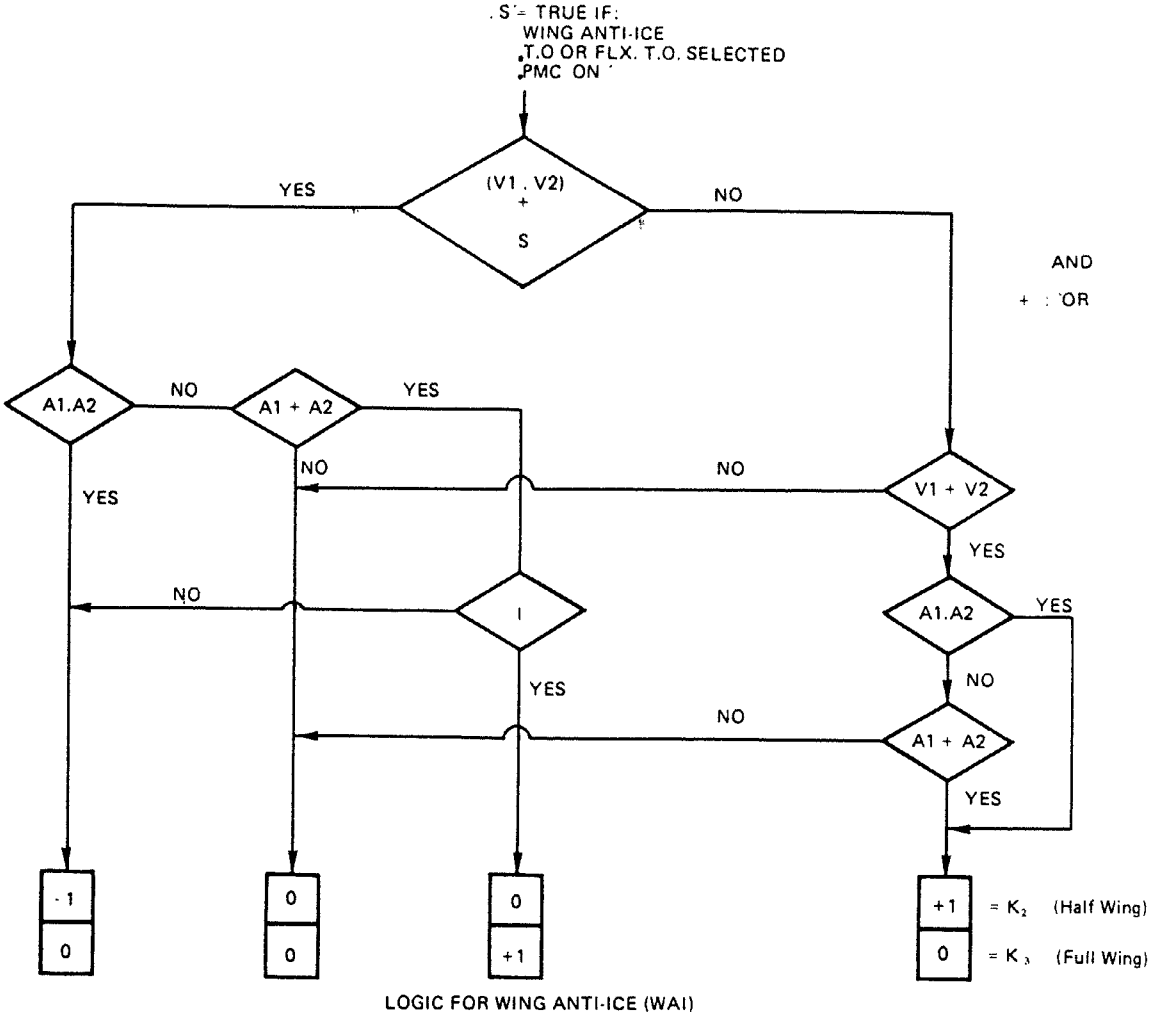
Bleed Valve Configuration
Figure 003



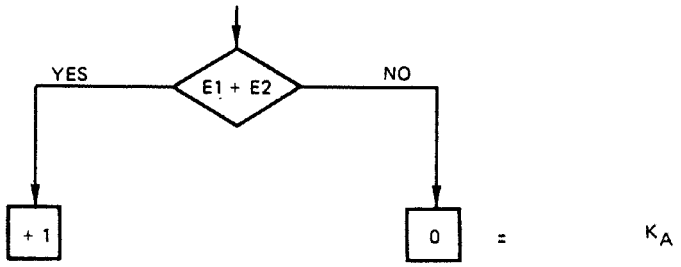
K1 CORRECTION GAIN CORRESPONDING TO AIR BLEED FOR AIR CONDITIONING

BM5 22 36 00 0 FXM2 08 03
DM5

Logic For Air Conditioning
Figure 004



LOGIC FOR WING ANTI-ICE (WAI)



BM5 22 36 00 0 FYM2

Logic for Nacelle Anti-Icing
Figure 005

THRUST CONTROL COMPUTER 301CJ1 - REMOVAL/INSTALLATION

1. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Blanking Caps
Referenced Procedure - 22-40-00, P. Block 501	System Monitor

(Ref. Fig. 401)

2. Procedure

A. Job Set-Up

(1)Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC2/28VDC	306CC2	103/G12
21VU	AFS/TCC/SYS1/28VDC	306CJ1	102/H 8
21VU	AFS/TCC/SY1/115VAC	308CJ1	102/H 9
21VU	AFS/TCC/SENSORS	305CJ1	102/H10
21VU	AFS/TRP PWR/SUPPLY	36CJ	102/H14
133VU	L/G PROX DET & RELAYS/SYS2/FLT/GND	1GB	331/V52

(2)Open access door 121BL and gain access to TCC to be removed.

B. Removal

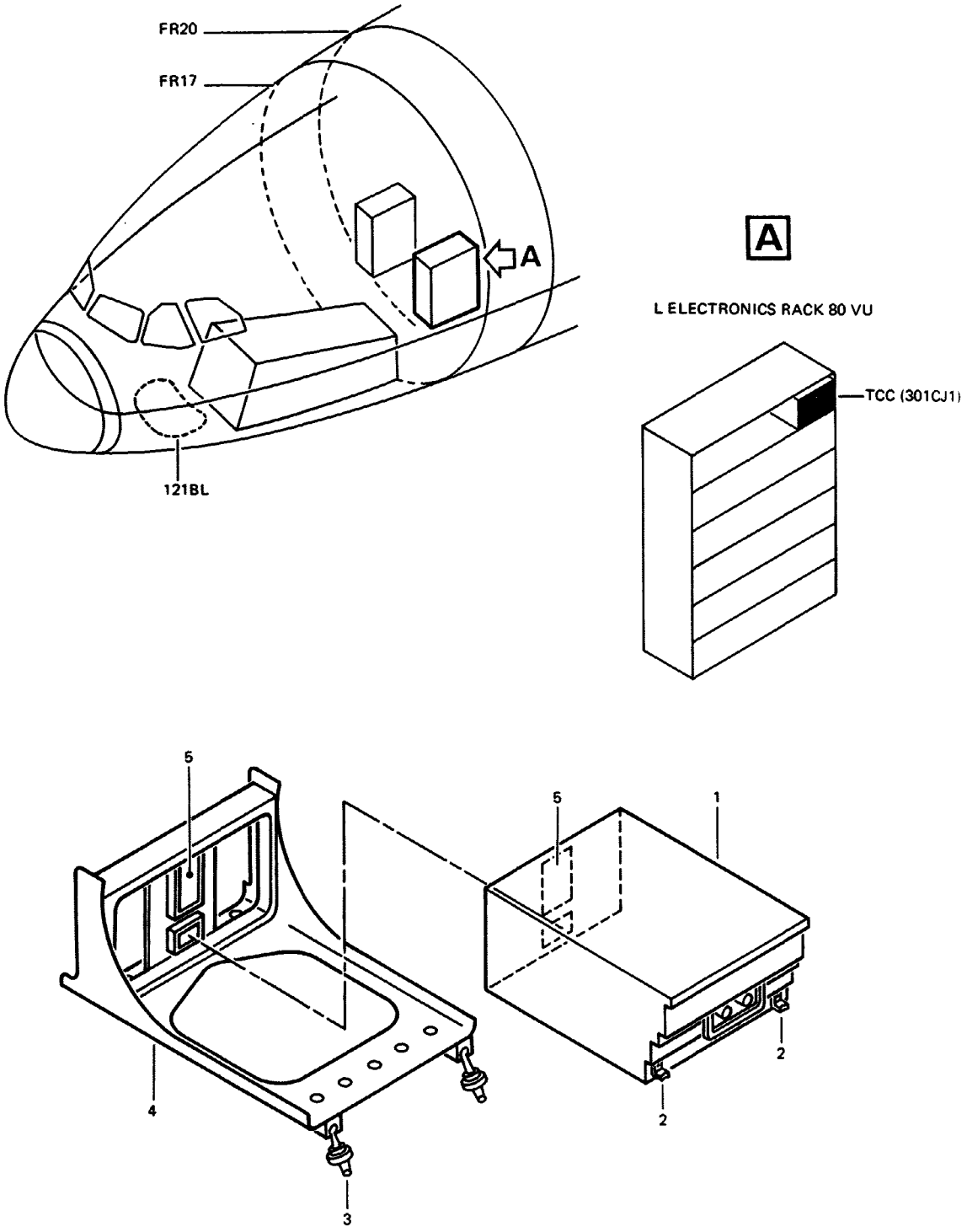
(1)Unscrew knurled nuts (3) at front of mount (4) and tilt them downwards.
(2)Withdraw thrust control computer (1) from mount (4).
(3)Cap electrical connectors (5).
(4)Remove thrust control computer (1).

C. Installation

(1)Clean and inspect thrust control computer adjacent area or interface.
(2)Remove blanking caps from electrical connectors (5).
(3)Check for correct condition of electrical connectors (5).
(4)Position thrust control computer (1) on mount (4).
(5)Engage knurled nuts (3) in studs (2) and tighten.
(6)Remove safety clips and tags and close the following circuit breakers :
- 306CA2, 305CJ1, 306CC1, 306CC2, 306CJ1, 308CJ1, 36CJ and 1GB.

D. Test

Carry out AFS test (Ref. 22-40-00, P. Block 501, paragraph 1.A.).
Carry out LAND test (Ref. 22-40-00, P. Block 501, paragraph 1.B.), if aircraft is operated in actual AUTO LAND cat 3 conditions.



Thrust Control Computer
Figure 401

E. Close-Up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Close access door 121BL.

THRUST CONTROL COMPUTER 301CJ1 (301CJ2) - REMOVAL/INSTALLATION

1. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Blanking Caps
Referenced Procedure - 22-40-00, P. Block 501	System Monitor

(Ref. Fig. 401)

2. Procedure

A. Job Set-Up

(1)Open, safety and tag the following circuit breakers :

(a)For TCC1

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP2-FD2/FCC2/28VDC	306CA2	102/H 6
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC2/28VDC	306CC2	103/G12
21VU	AFS/TCC/SYS1/28VDC	306CJ1	102/H 8
21VU	AFS/TCC/SY1/115VAC	308CJ1	102/H 9
21VU	AFS/TCC/SENSORS	305CJ1	102/H10
21VU	AFS/TRP PWR/SUPPLY	36CJ	102/H14
133VU	L/G PROX DET & RELAYS/SYS2/FLT/GND	1GB	331/V52

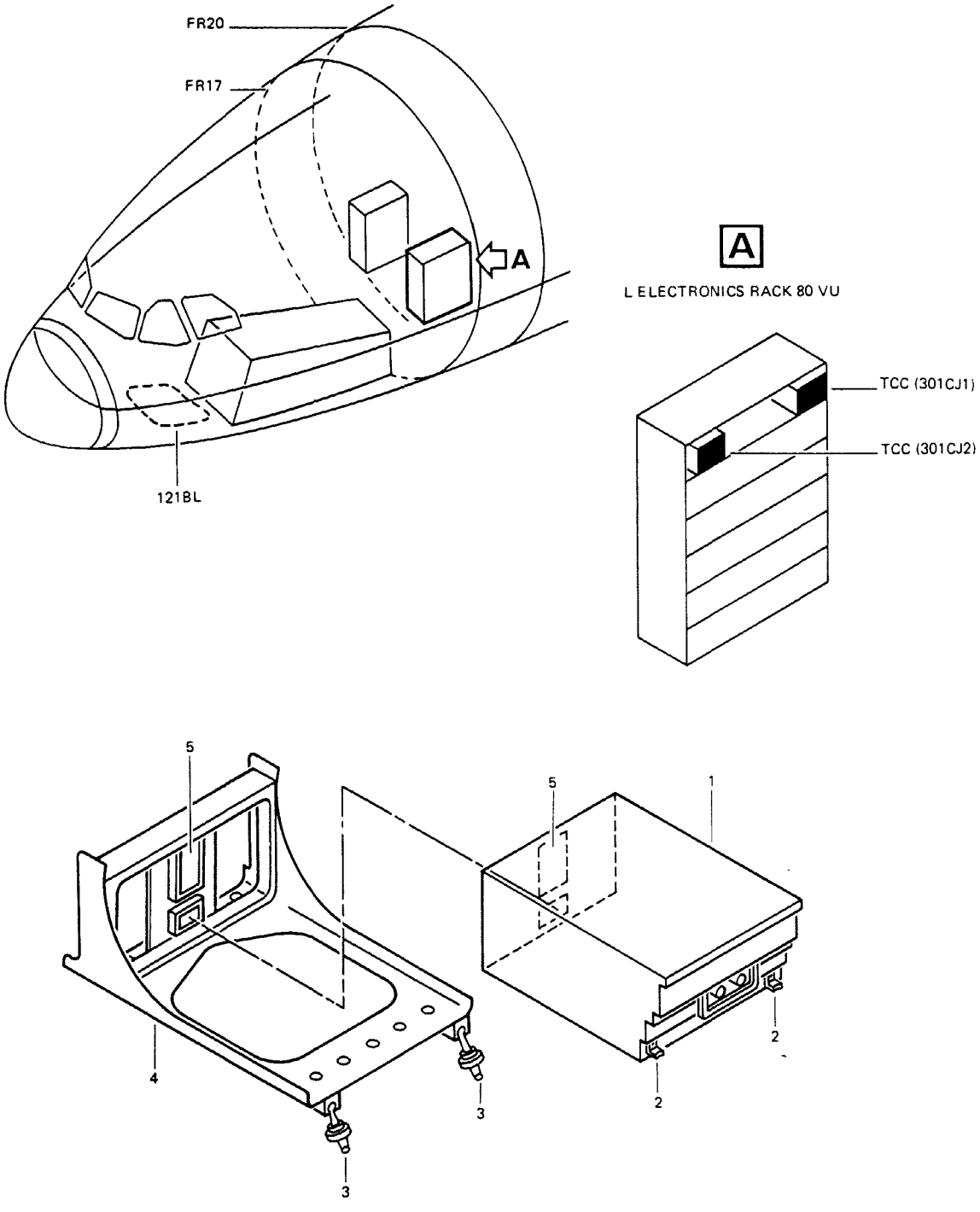
(b)For TCC2

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/AP1-FD1/FCC1/28VDC	306CA1	102/H 3
21VU	AFS/TCC/SYS2/28VDC	306CJ2	102/H12
21VU	AFS/TCC/SY2/115VAC	308CJ2	102/H11
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC2/28VDC	306CC2	103/G12
21VU	AFS/TCC/SENSORS	305CJ1	102/H10
21VU	AFS/TRP PWR/SUPPLY	36CJ	102/H14
133VU	L/G PROX DET & RELAYS/SYS1/RELAYS & RETRACT CTL	2GB	332/U51

(2)Open access door 121BL and gain access to TCC to be removed.

B. Removal

(1)Unscrew knurled nuts (3) at front of mount (4) and tilt them downwards.
(2)Withdraw thrust control computer (1) from mount (4).
(3)Cap electrical connectors (5).
(4)Remove thrust control computer (1).



Thrust Control Computer
Figure 401

C. Installation
(1)Clean and inspect thrust control computer adjacent area or interface.
(2)Remove blanking caps from electrical connectors (5).
(3)Check for correct condition of electrical connectors (5).
(4)Position thrust control computer (1) on mount (4).
(5)Engage knurled nuts (3) in studs (2) and tighten.
(6)Remove safety clips and tags and close the following circuit breakers :
(a)For TCC1 : 306CA2, 305CJ1, 306CC1, 306CC2, 306CJ1, 308CJ1, 36CJ and 1GB.
(b)For TCC2 : 306CA1, 305CJ1, 306CC1, 306CC2, 306CJ2, 308CJ2, 36CJ and 2GB.

D. Test
Carry out AFS test (Ref. 22-40-00, P. Block 501, paragraph 1.A.).
Carry out LAND test (Ref. 22-40-00, P. Block 501, paragraph 1.B.), if aircraft is operated in actual AUTO LAND cat 3 conditions.

E. Close-Up
(1)Make certain that working area is clean and clear of tools and miscellaneous items of equipment.
(2)Close access door 121BL.

1. ATS Arming

A. TCC Arming Logic

(1)General

The levers are held in armed position if the monitoring and command logic circuits associated with each computer are valid.

It is to be noted for each computer that power is supplied to the clutch if :

- the monitoring and command logics are validated,
- the arming lever is in the **ON** position,
- self-latching relay is energized.

The self-latching relay is supplied via the arming lever.

The self-latching relay provides grounding of the common point of the clutches.

System disconnection causes the corresponding arming lever to return to the **OFF** position.

(2)TCC arming logic principle

The arming logic is achieved in the command and monitoring channels.

Lever self-holding is possible only if the command and monitoring logics are correct.

(3)Disarming

(a)Signals directly prohibiting arming whatever the flight phase :

- disconnection of a dynamometric rod connector
- disconnection of the actuator or coupling unit connector
- arming circuit failure (self-latching relay)
- computer internal failure (components, power supply permanent failure, etc...).

(b)Signals directly causing disarming :

- triggering by watchdogs 1 or 2
- failure detected by comparators C1 or C2.

(c)Combined signals causing disarming

(d)Signals causing disarming but enabling reconfiguration

- loss of **ADC**1 or 2
- loss of one **FMC** when **AP**s and **FD**s are disengaged.
- in active **PROFILE** mode when **SPD TARGET** or **THR TARGET** is **NCD**.

(Ref. Fig. 001)

(Ref. Fig. 002)

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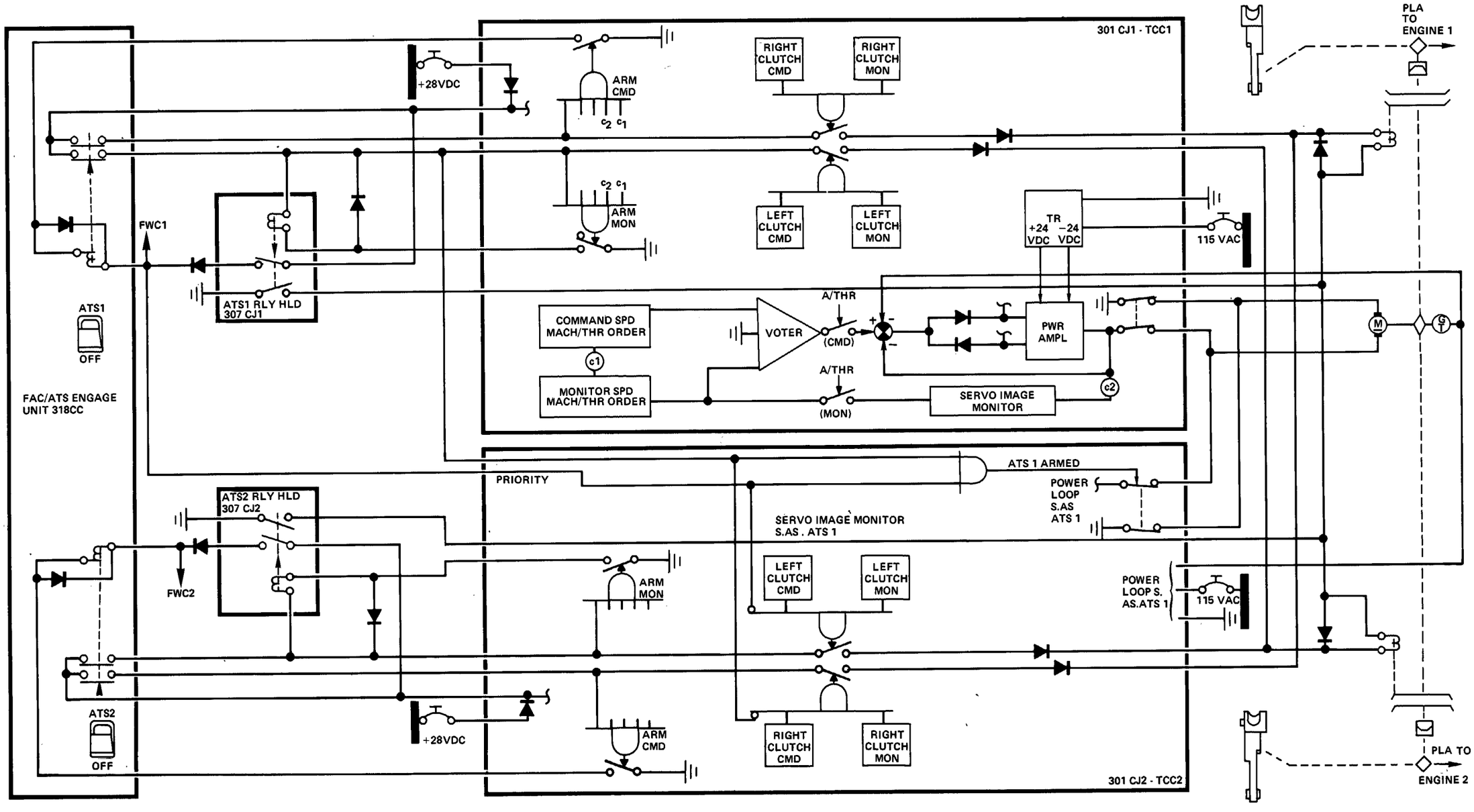
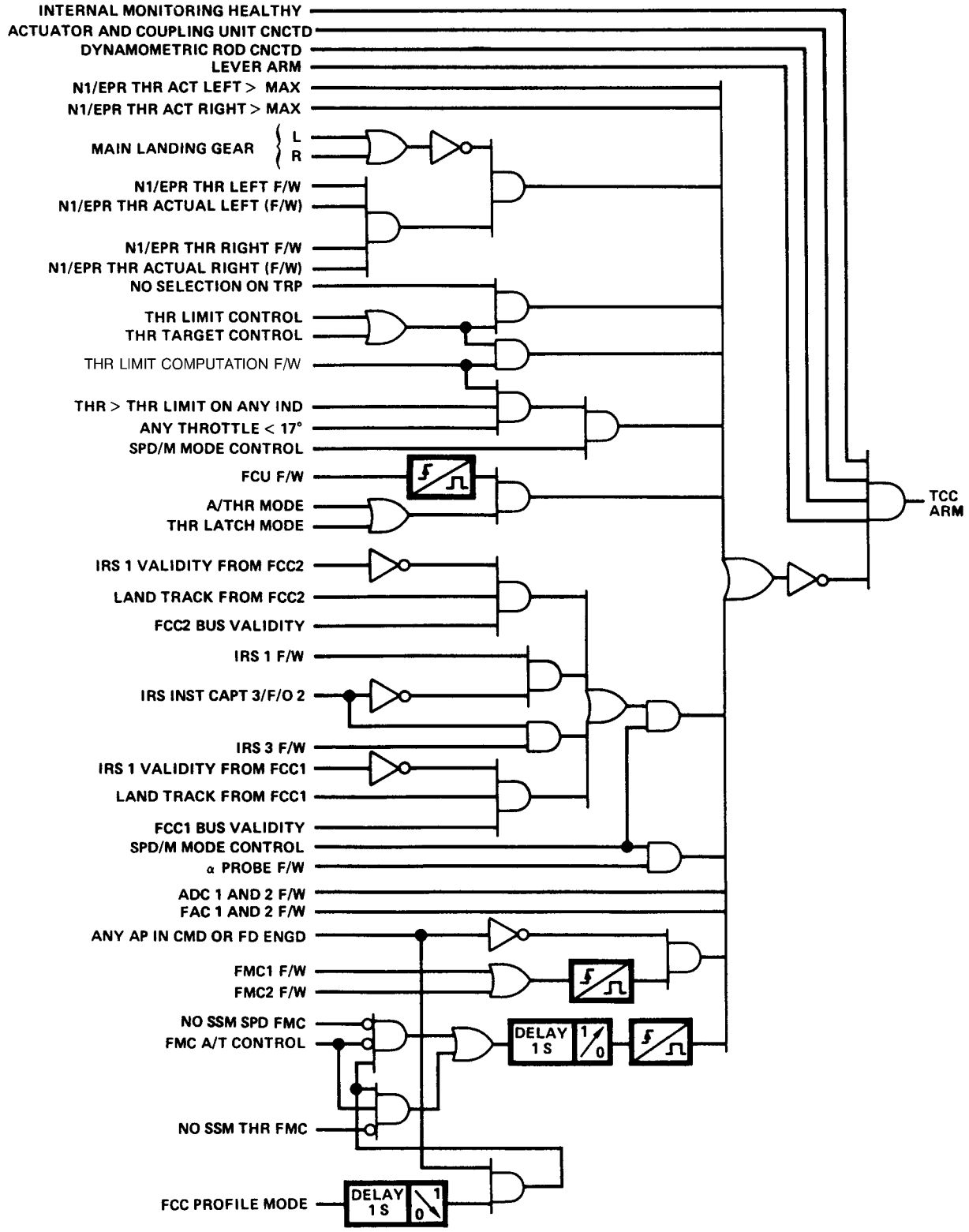


BM5 22 37 00 0 DAB2 24 19
DM5

ATS - Arming Principle
Figure 001

EFFECTIVITY: 001-100,

PIA



BM5 22 37 00 0 DEF2 01

TCC Arming and Disarming Logic
Figure 002

2. ATS Engagement

A. Power Control (See 22-34).

B. Operating Modes

(1)Mode organization

Upon system arming, two main modes are available.

(a)A/THR mode

The A/THR mode corresponds to the automatic operation of the system in response to commands given either by the pilot or by peripherals (FCC, FMC).

(b)THR LATCH mode

This is a standby mode which enables recovery of the limit thrust of the mode selected on the TRP after operation of the TO/GA lever or if the alpha floor protection is activated (FAC).

(Ref. Fig. 003)

(2)A/THR mode

(a)Characteristics

The A/THR mode enables engine automatic control in various modes in relation to the peripheral system demands : FCC, FMC.

(b)Engagement/Disengagement

The A/THR Mode will disengage on the ground :

- if the throttle control levers are moved to the idle position (6°) 4 s after A/THR engagement (case of acceleration-stop)

- to prevent taking off with a mode other than TO or FLX TO selected on the TRP. A condition associated with the selection of MCT, CR, CL, and VC ≤ 60 kt disengages the A/THR mode.

(c)Display

On ground, green A/THR indication is displayed when A/THR pushbutton switch on FCU is pressed and if engines are stopped.

In flight, cyan A/THR indication is displayed after retard phase of LVL CHANGE in descent.

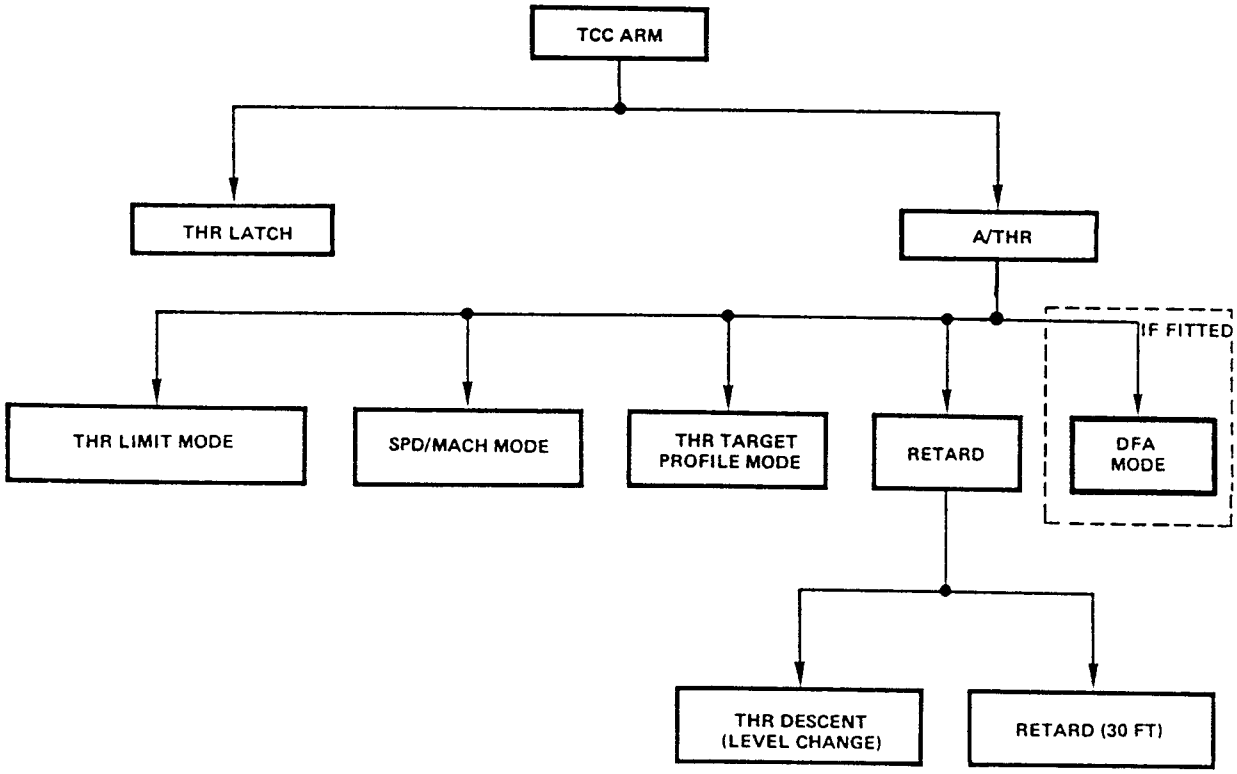
(Ref. Fig. 004)

(3)THR LATCH mode

(a)Characteristics

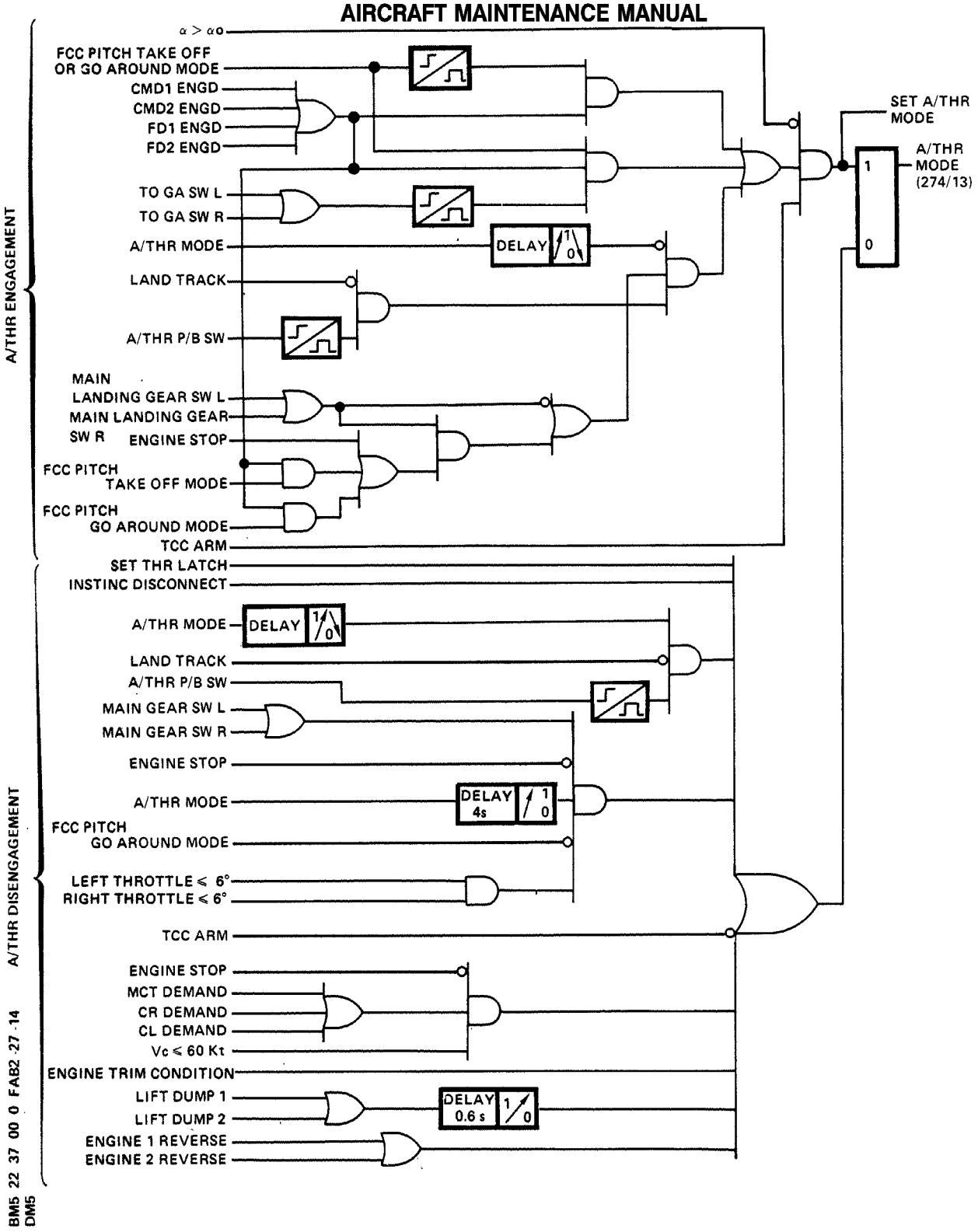
The THR LATCH mode is a standby mode. It is used to recover the limit thrust manually by means of TO/GA levers or automatically if alpha floor protection is requested by the FAC.

(b)Engagement/disengagement



BM5 22 37 00 0 DCA2 00

Mode Organization
Figure 003

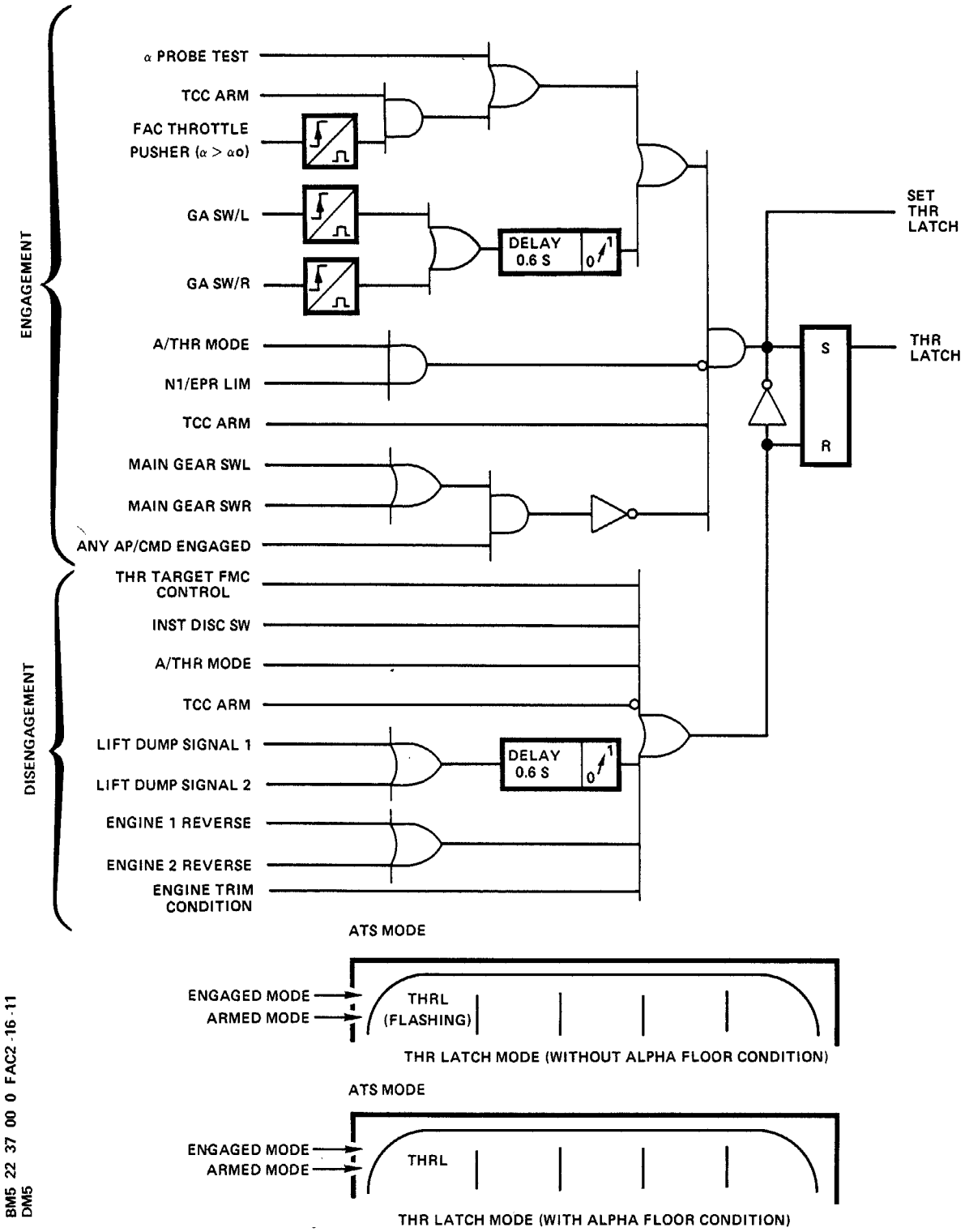


A/THR Mode
Figure 004

NOTE : Engagement is made possible if the TCC is armed and if the limit thrust is not already requested by the A/THR automatic mode. This mode cannot be engaged if AP is in CMD as soon as main GEAR is pressed (LAND phase).

(c)Display

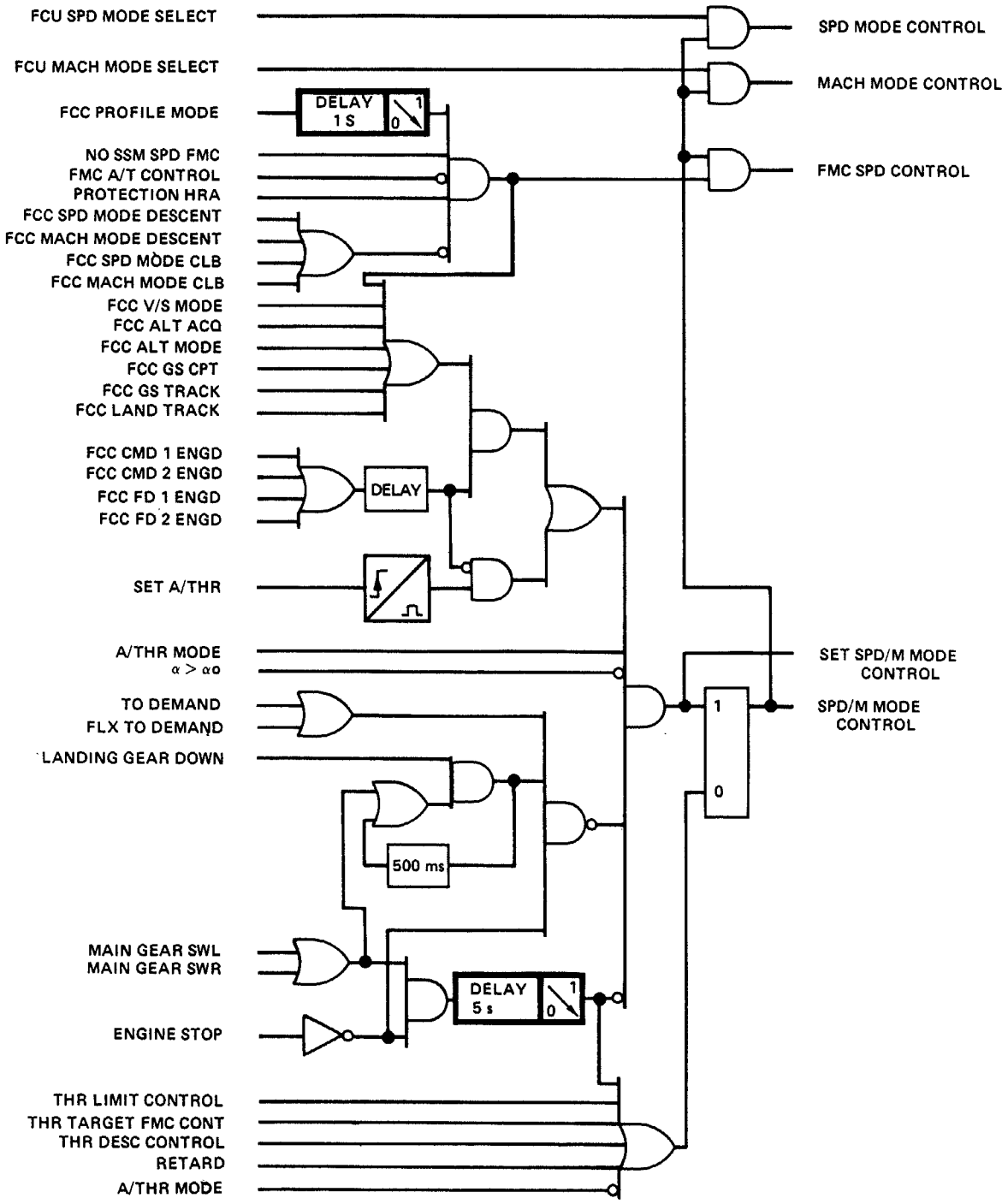
When the THR LATCH mode is engaged out of alpha floor protection, the green THR L indication flashes in the ATS mode column.
(Ref. Fig. 005)



BM5 22 37 00 0 FAC2 16 11
DM5

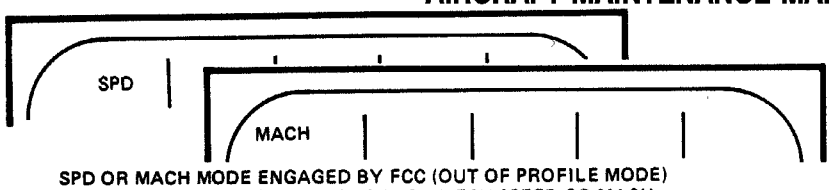
THR Latch Mode - Display
Figure 005

(4)SPEED/MACH mode control
(a)SPEED hold characteristics
This mode is used to maintain automatically the speed selected on the FCU through the TCC, slaving feedback being provided by the aircraft speed given by ADC1 (or ADC2 if ADC1 failed).
Vmin and Vmax values generated by the FAC are used as protections.
Speed cannot be greater than Vmax or lower than Vmin.
(b)MACH hold
Same as SPEED hold (with MACH from FCU and ADC).
Vmin and Vmax protection is also performed.
(c)SPD/M protection by THR channel
When in speed or Mach acquisition mode and in order to prevent exceeding N1/EPR limit, automatic switching is performed on the THR channel when the speed channel output command becomes greater than the command given by the THR channel. This switching is active all the time a speed command is greater than the THR command.
(d)Display
Green SPD or MACH is displayed in the ATS Mode column on the PFDs.
Green P SPD or P MACH comes on when SPD or MACH mode is engaged upon FMC request.
(e)Engagement/disengagement
(Ref. Fig. 006)
(Ref. Fig. 007)
(5)Throttle control lever automatic retard
The throttle control levers can be moved automatically towards the idle position when in one of the following phases :
- descent controlled by the FCC (LVL/CH)
- profile controlled by the FMC
NOTE : In these cases the throttle control levers move to the idle position at 1°/s.
- at 30 ft.
NOTE : In this case the throttle control levers move to the idle position at 2°/s.
(a)Descent controlled by the FCC (THR descent control)
1 Engagement/disengagement
(b)Descent controlled by the FMC (FMC idle command)
1 Engagement/disengagement
(c)Throttle control lever retard during flare
1 Engagement/disengagement
(d)Display
During the retard phase, the RETARD indication on the PFDs comes on.
When idle is reached or if the control lever inovement is stopped during the retard phase in THR DESCENT or IDLE FMC, the cyan A/THR indication on the PFDs comes on.
(Ref. Fig. 008)
(Ref. Fig. 009)
(6)Not applicable.

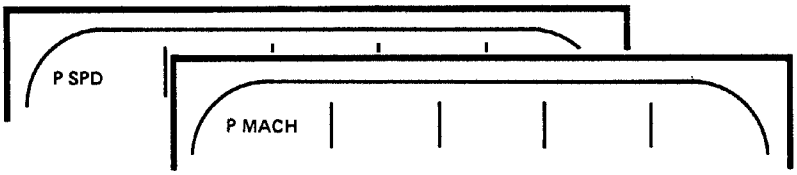


BM5 22 37 00 0 DAH2 15 10
DM5

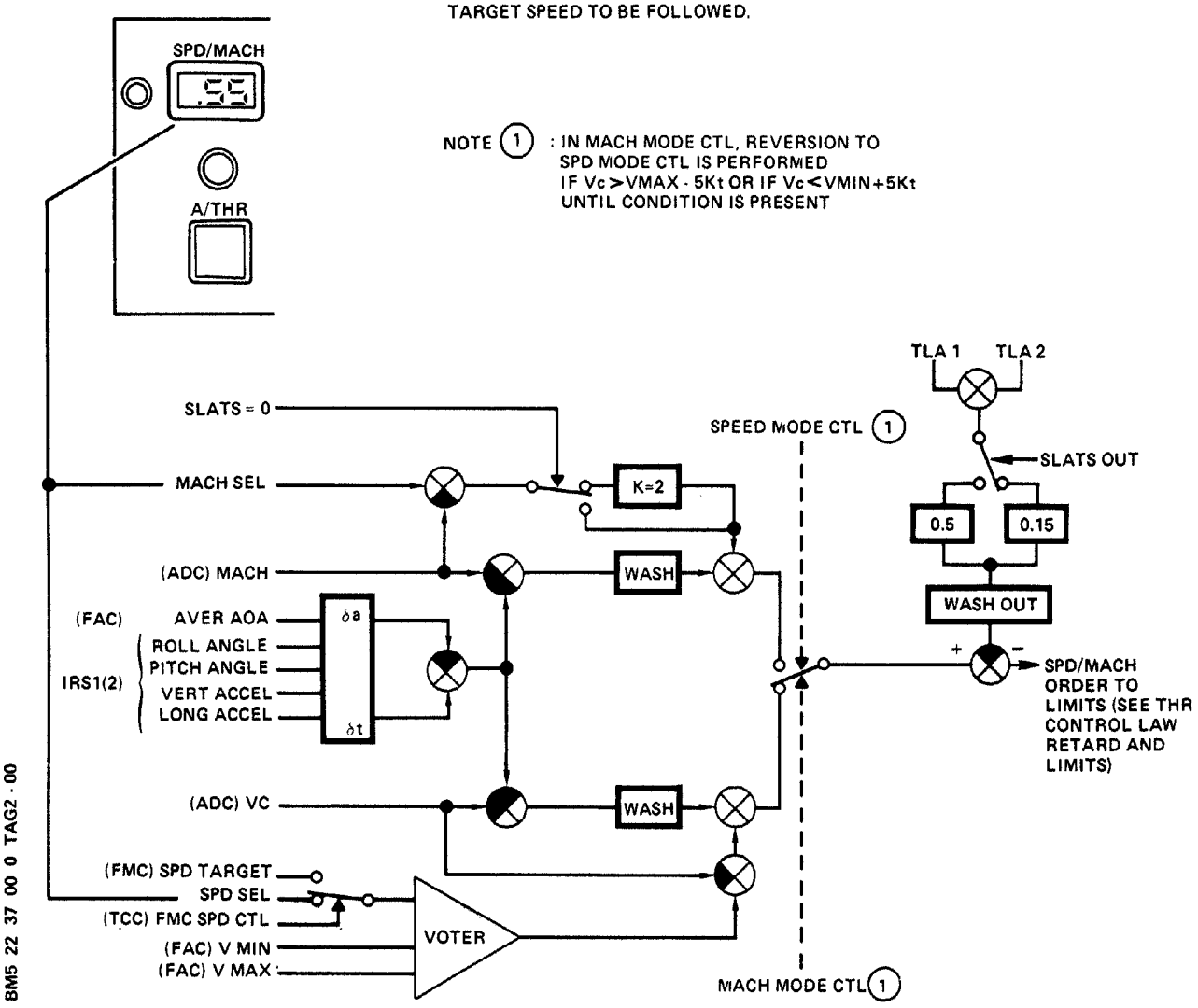
SPD/MACH Mode Control - Engagement/Disengagement
Figure 006



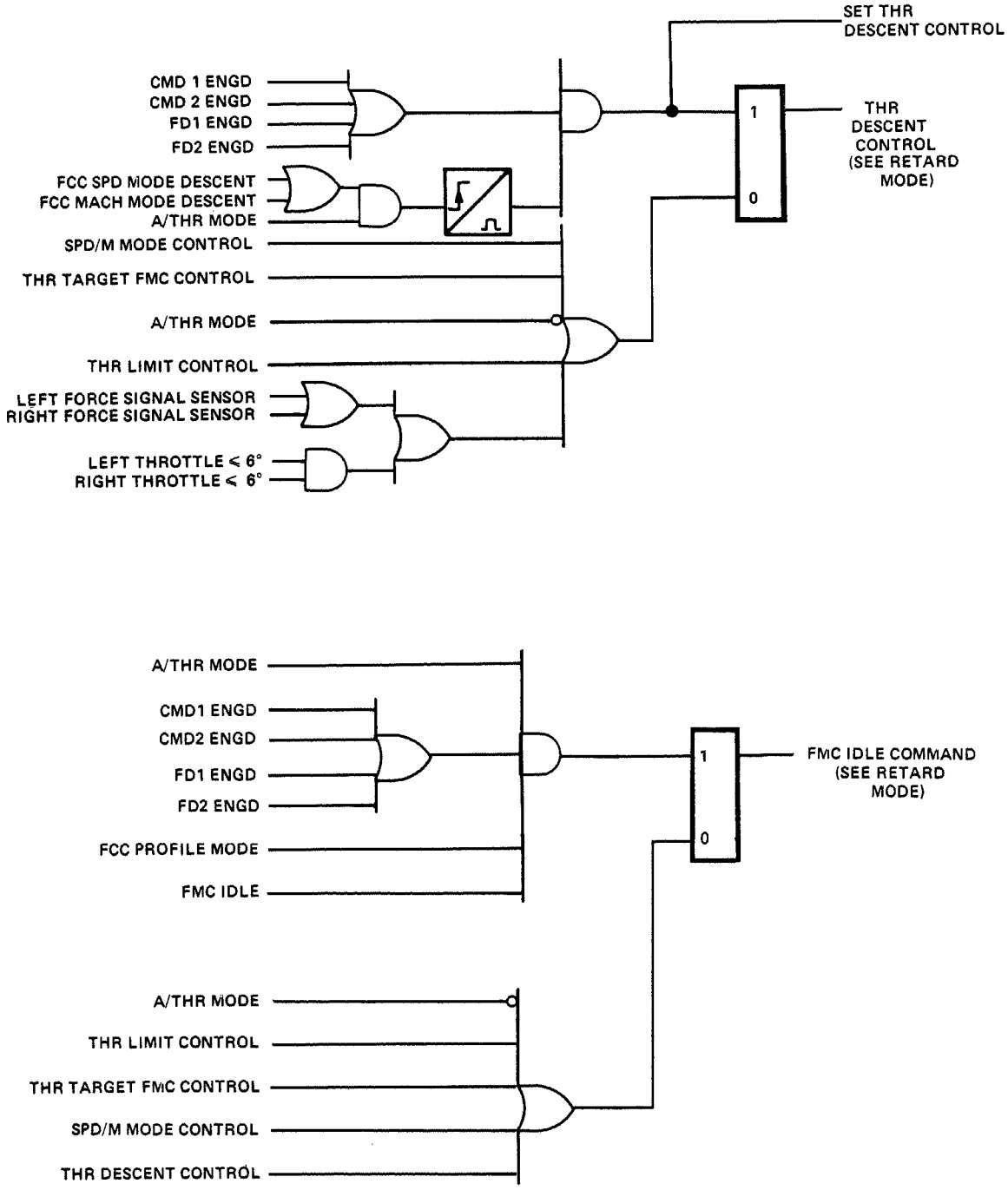
SPD OR MACH MODE ENGAGED BY FCC (OUT OF PROFILE MODE) OR MANUALLY BY A/THR ENGAGEMENT. FCU SPEED OR MACH DISPLAY GIVES THE VALUE TO BE FOLLOWED.



SPD OR MACH MODE ENGAGED BY FCC/FMC. FMC GIVES THE TARGET SPEED TO BE FOLLOWED.

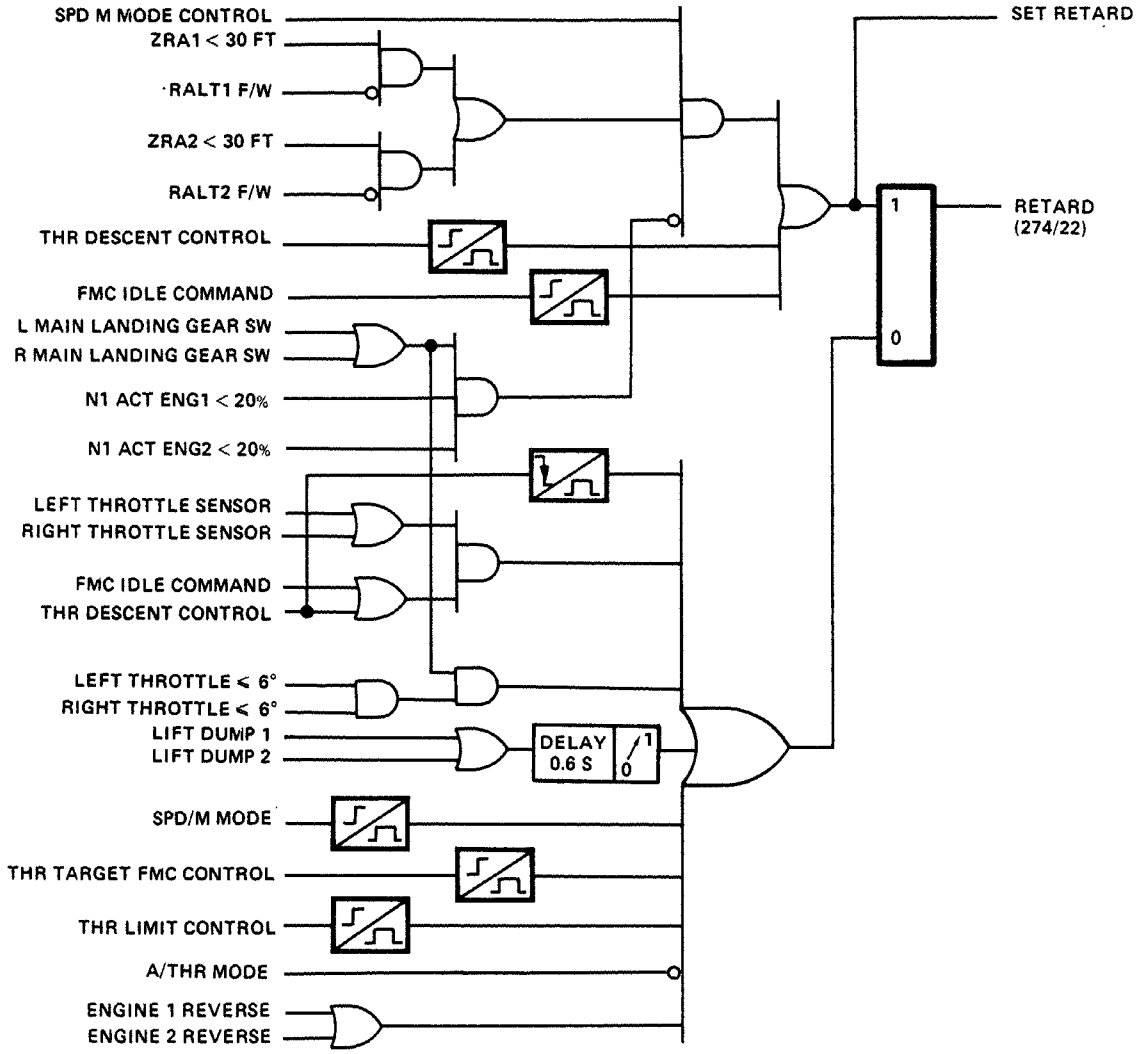


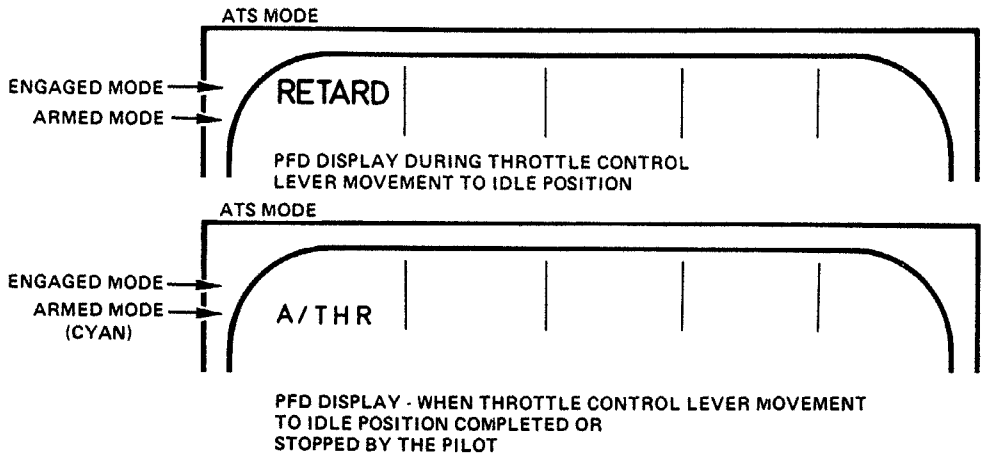
SPD/MACH Mode Control Laws and Display
Figure 007



BM5 22 37 00 0 TAJ2 27 11
DM5

THR Descent Control/FMC Idle Command
Figure 008




ATS MODE
ENGAGED MODE
ARMED MODE
RETARD
PFD DISPLAY DURING THROTTLE CONTROL LEVER MOVEMENT TO IDLE POSITION
ATS MODE
ENGAGED MODE
ARMED MODE (CYAN)
A/THR
PFD DISPLAY - WHEN THROTTLE CONTROL LEVER MOVEMENT TO IDLE POSITION COMPLETED OR STOPPED BY THE PILOT

BM5 22 37 00 0 TAK2 19 10
DM5

Logic Retard Mode/PFD Display
Figure 009

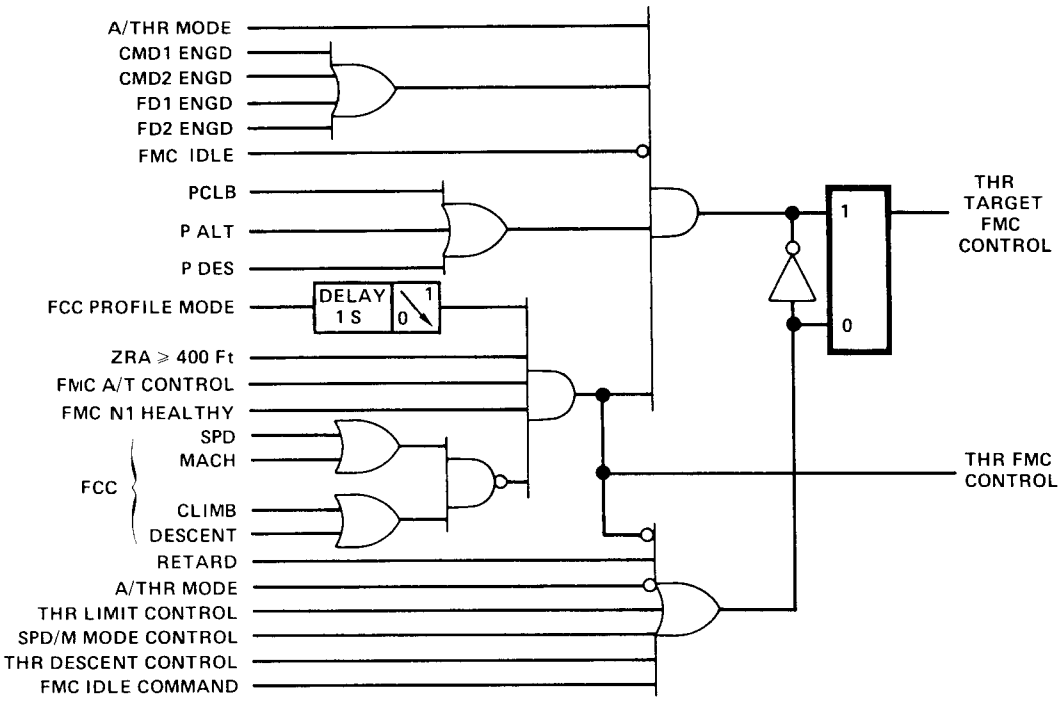
(7)THR TARGET mode - FMC control
The THR TARGET mode is engaged automatically if the FCC is engaged in PROFILE mode so that the TCC acquires the THR TARGET signal delivered by the FMC.
The signal delivered by the FMC is limited in its upper limit by the THR LIMIT signal and in its lower limit by the P.CLIMB condition.
(a)Engagement/disengagement
(b)Display
(Ref. Fig. 010)

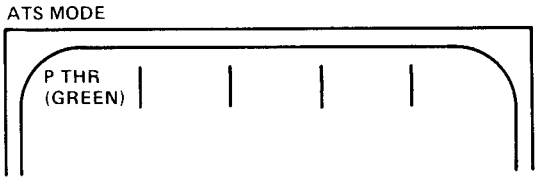
(8)N1 LIMIT mode control
The N1 LIMIT mode enables acquisition of the limit rating corresponding to the mode selected on the TRP.
(a)Operation at take-off using PCSs
Take-off limit thrust corresponds to TO or FLX TO thrust.
Both throttle control levers are operated by the autothrottle actuator and move at 3°/s to the corresponding position where the N1 command of the considered engine is equal to the N1 limit.
The corresponding throttle control lever is declutched when this condition is fulfilled.
Systematic declutching occurs at 100kt to protect the system against untimely throttle control lever retard during take-off.
(b)Operation at take-off without PMCs
Automatic take-off with PMC, selected OFF is prohibited.
(c)N1 LIMIT control in flight
The slaving law is : N1 limit of selected mode-max N1 command of both engines.
(d)Engagement/disengagement
(e)Display
Green THR is displayed when the thrust mode is active with A/THR function engaged.

(Ref. Fig. 011)

(9)Declutch logic
As the TCC is armed when a mode is selected, the throttle control levers are clutched.
They can be declutched :
- separately in TAKE-OFF mode when the conditions on/N1 limit - N1 command/are obtained.
- Simultaneously at 100kt, if the conditions previously mentioned are not obtained.
- If the pilot applies a load on a throttle control lever, the corresponding clutch is declutched. When the load is no longer applied the lever is clutched.

(a)Display
The cyan THR indication appears when declutching occurs in take-off phase.





THR ENGAGED BY FCC/FMC. FMC GIVES THR TARGET (DISPLAYED ON TRP) TO BE FOLLOWED BY ATS

BM5 22 37 00 0 TAL2 00

FMC THR TARGET Control
Figure 010

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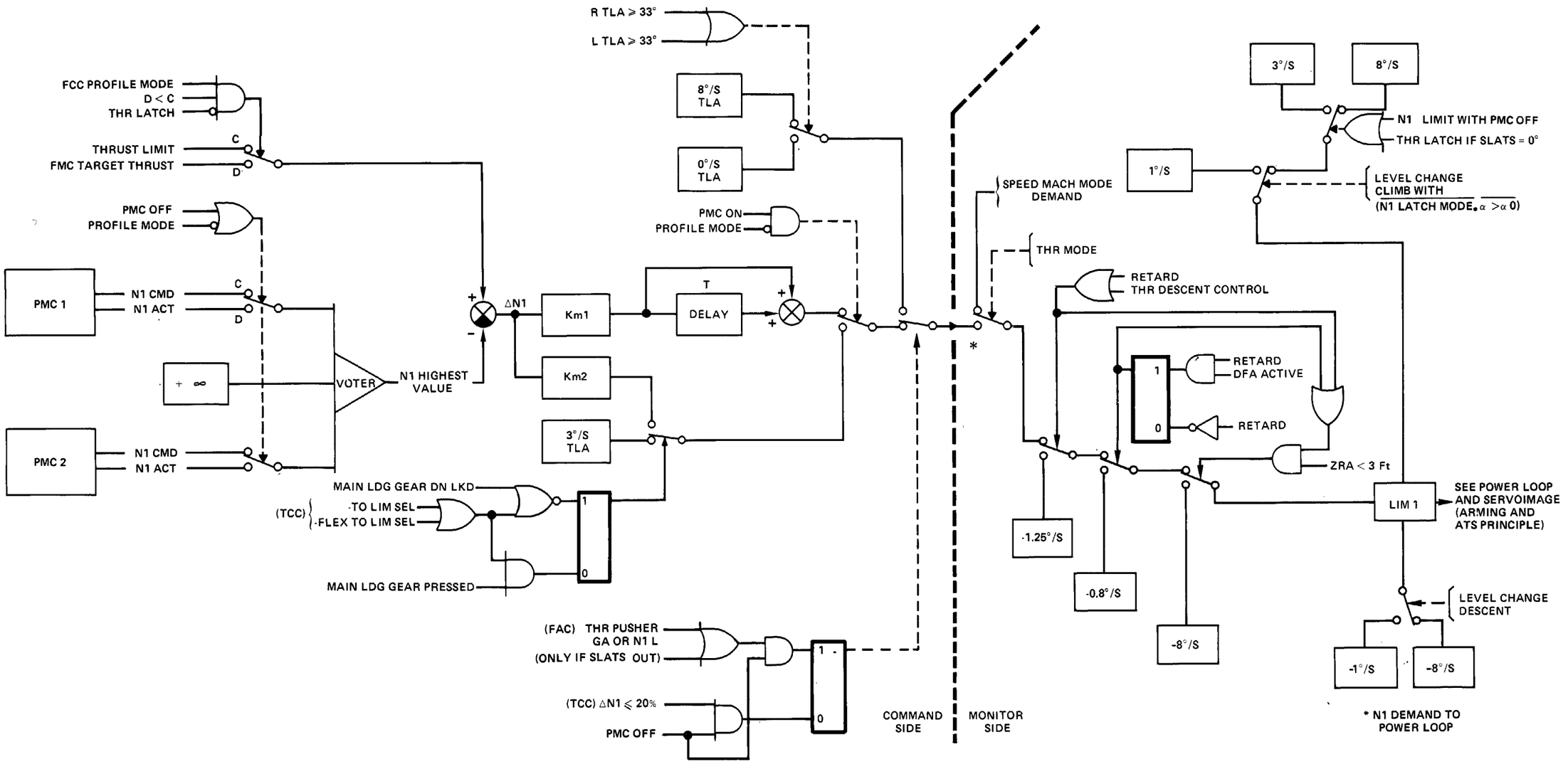
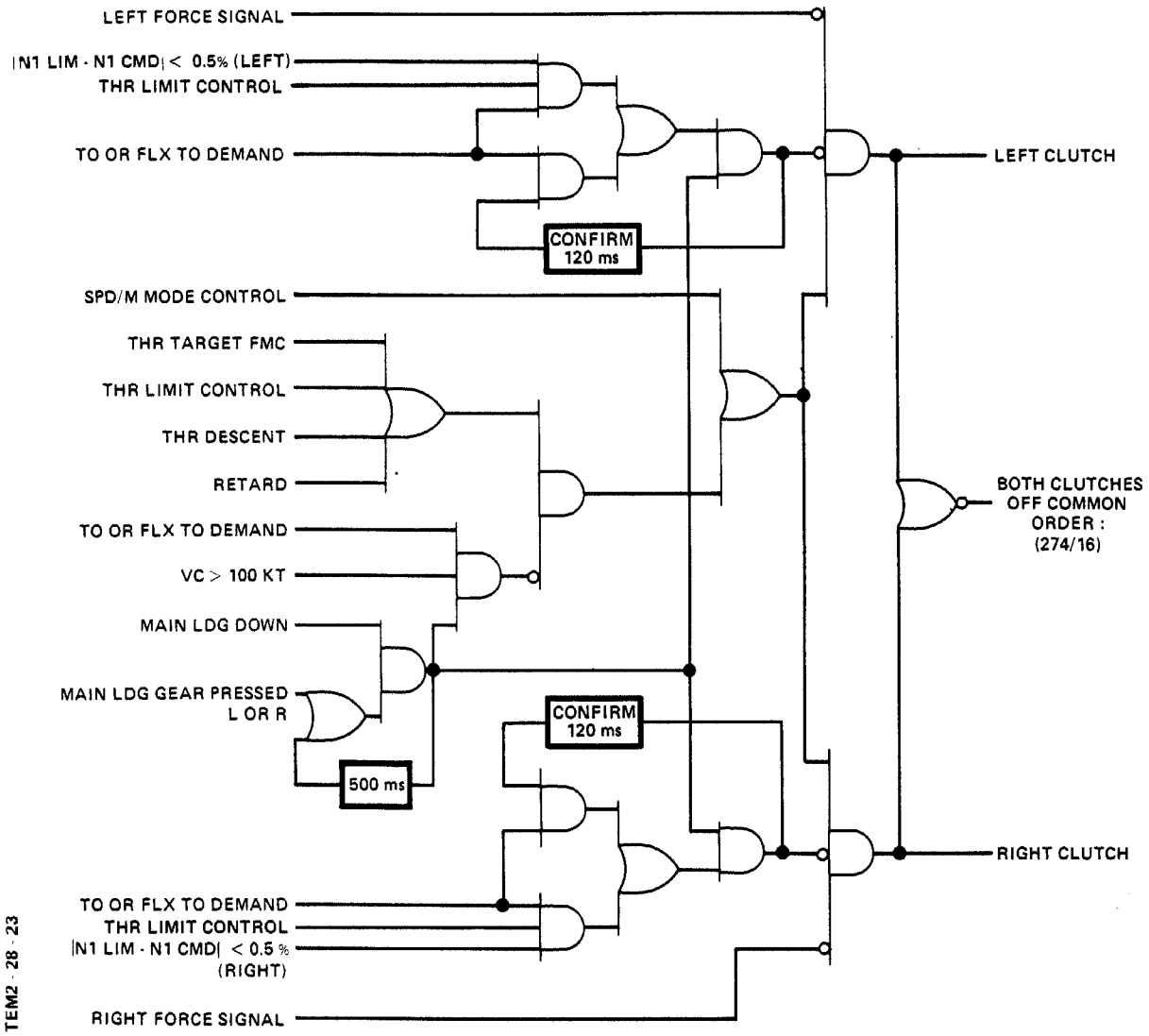


BM5 22 37 00 0 FAP2 21 16
DM5

N1 Control Laws
Figure 011

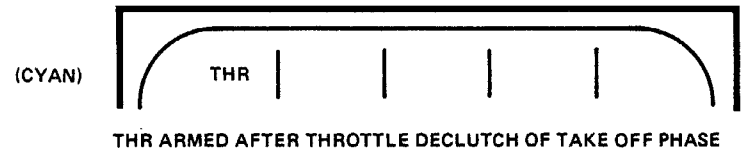
Cyan THR indicates that both throttle control levers are declutched but THR mode is always engaged.

(10)Throttle control lever displacement stops
(Ref. Fig. 012)
(Ref. Fig. 013)

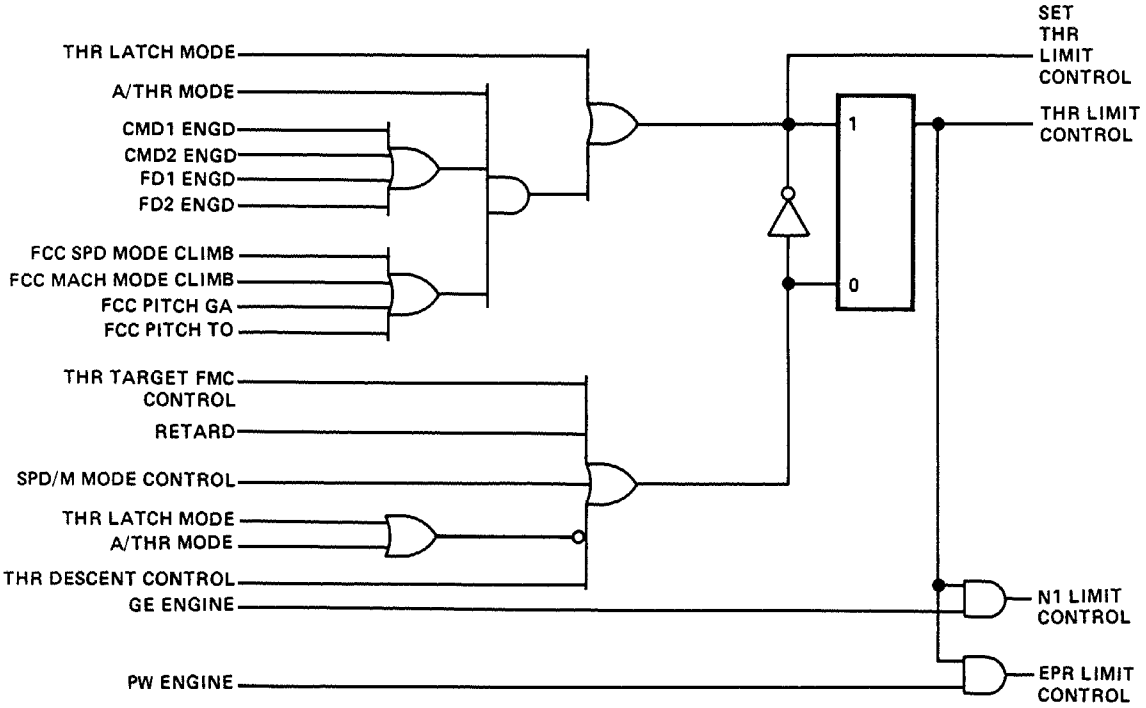


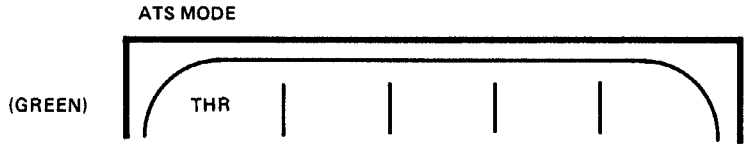
BM5 22 37 00 0 TEM2 28 - 23
DM5

Declutch Logic
Figure 012



THR ARMED AFTER THROTTLE DECLUTCH OF TAKE OFF PHASE





THR ENGAGEMENT BY FCC OR MANUALLY BY TO/GA LEVERS.
TCC GIVES THR LIMIT TO BE FOLLOWED.

BM5 22 37 00 0 DAN2 15 10
DM5

THR Limit Mode Control
Figure 013

(11)THR limit computation failure
In the event of a THR limit failure, only SPEED mode can be selected in automatic mode.

(12)Throttle pusher (alpha floor)

(a)Engagement
Protection against high angle-of-attack is active as soon as the TCC is armed. To this end, the TCC is directly provided with the alpha floor command signal coming from FAC1 or FAC2 depending on the position of the PITCH TRIM and YAW DAMPER engage levers.

(b)Display
The green THR L indication comes on steady on the PFDs as long as the alpha floor condition is present. The indication flashes as soon as this condition disappears.
(Ref. Fig. 014)

(13)Thrust selection management
This function is used to select different thrust ratings (function of flight phase or aircraft weight).

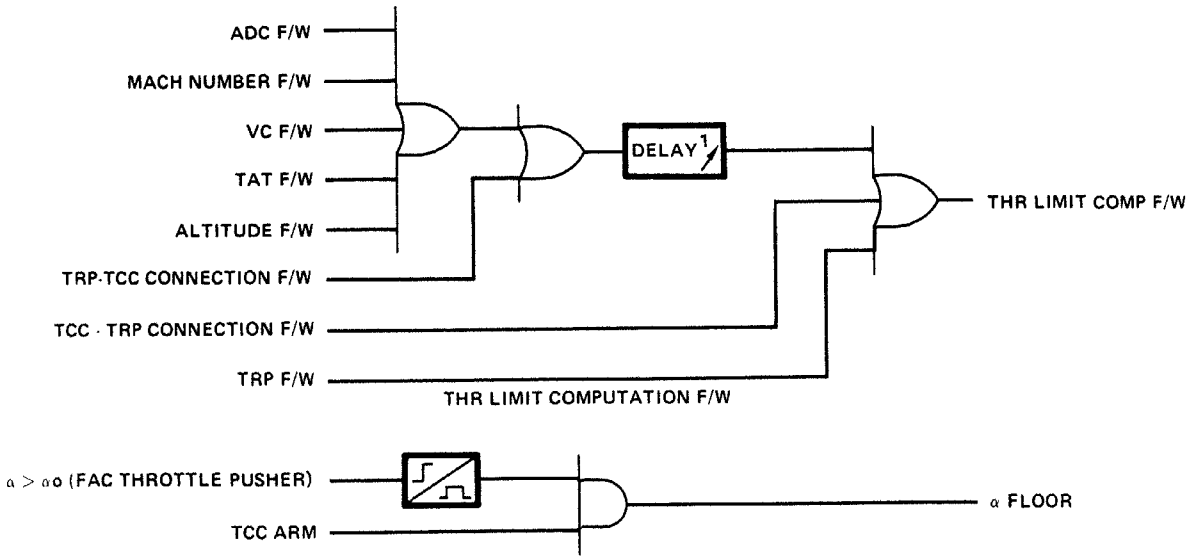
(a)GA THRUST LIMIT selection
1 Manual selection : in flight by pressing TO/GA pushbutton switch.
2 Automatic selection : when slats are extended if at least one AP or one FD is engaged and if the PROFILE mode is not engaged.
3 Display : TO/GA pushbutton switch light comes on and GA appears in LIM MODE display.

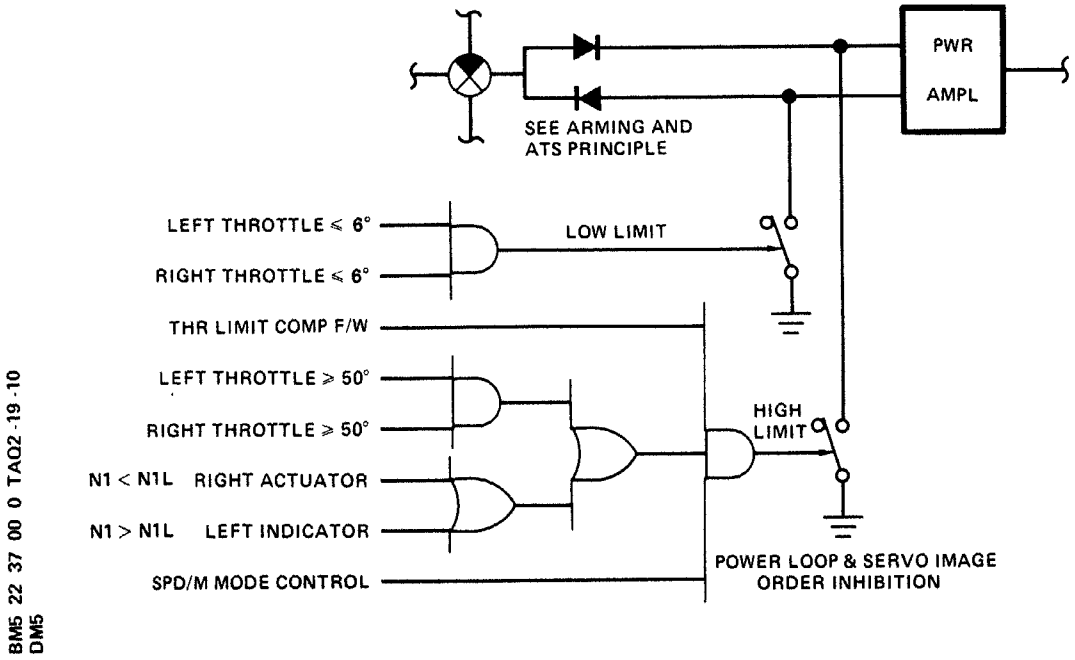
(b)TO THRUST selection (selection of maximum take-off thrust).
1 Manual selection : on the ground by pressing TO/GA pushbutton switch
2 Automatic selection : upon aircraft energization
3 Display : TO/GA pushbutton switch light comes on and TO appears in LIM MODE display

(c)FLX TO THRUST selection
Selection of a derated take-off thrust in the event of reduced take-off weight or long runway enables to increase engine component life and achieve noise abatement take-off procedures. Derating is selected by means of the fictitious temperature selector knob.
When the fictitious temperature is greater than the TAT a flexible take-off target value computed by the TCC appears in the TRP TARGET display.
1 Manual selection only : by means of the FLX TO pushbutton switch.
2 Display : FLX TO pushbutton switch light comes on and TO appears in LIM MODE display.

(d)Climb/Cruise/MCT thrust selections
1 Manual selection : by pressing corresponding pushbutton switch.
2 Automatic selection by the FMC CL/CR/MCT orders if AUTO mode is selected and altitude > 400 ft.
3 Display :
- CL/CR/MCT appears in LIM MODE display
- in manual selection corresponding CL/CR/MCT pushbutton switch light comes on.
- in automatic selection (FMC orders) only AUTO pushbutton switch light is on.

(e)Auto thrust selection (CL/CR/MCT thrust selection automatic management)
This function allows FMC automatic selection of CL/CR/MCT thrust modes.



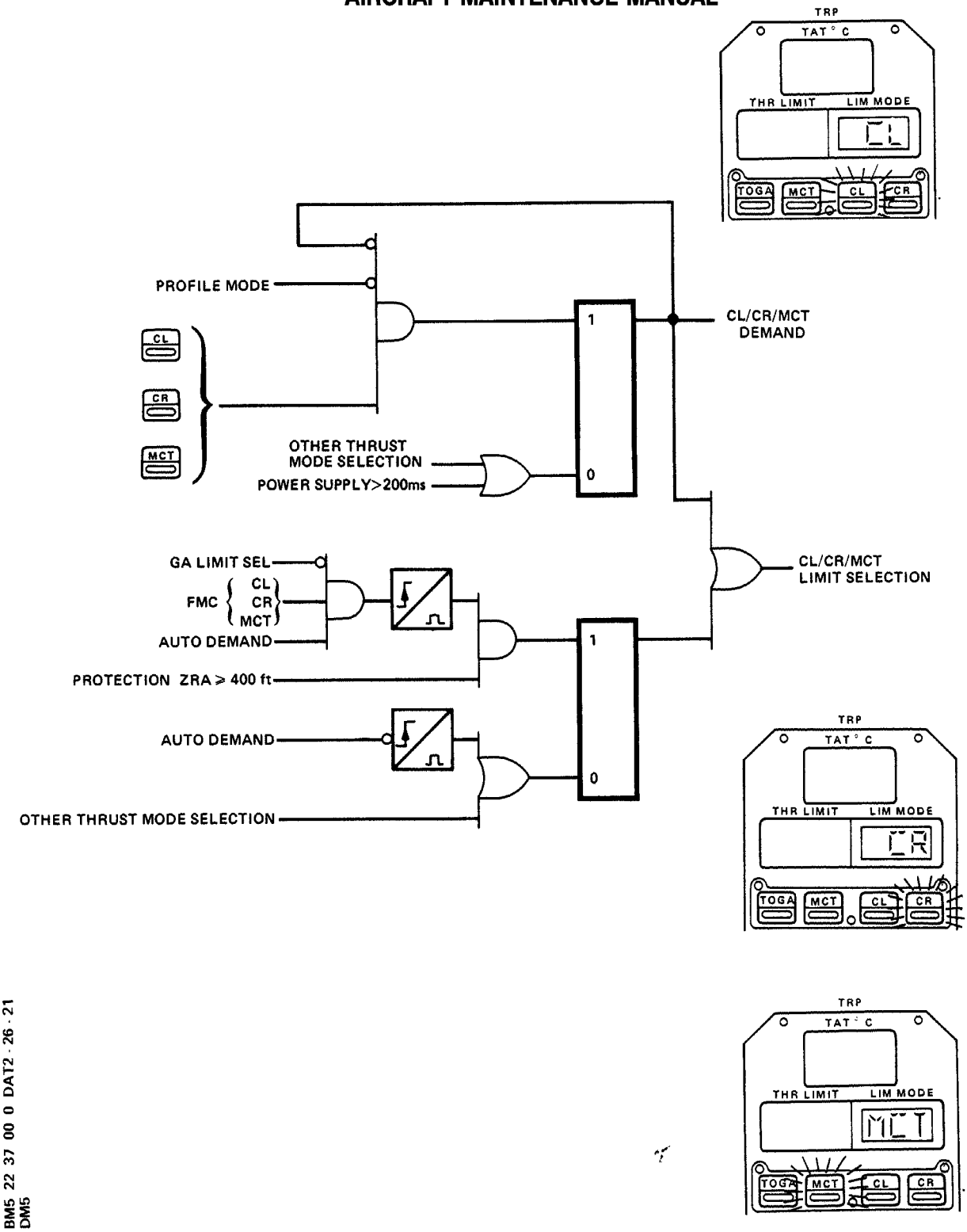


Throttle Control Lever Stop Logic/THR Limit Computation F/W
Figure 014

1 Manual selection by pressing the AUTO pushbutton switch above 400 ft. if the FMC sends a CL/CR/MCT order enables FMC-TCC coupling without selecting FCC PROFILE mode and management of thrust mode selection by the FMC.

2 Automatic selection
The AUTO mode is selected automatically as soon as the last one of the following conditions is present : PROFILE mode engagement, altitude > 400ft and presence of an FMC command (CL/CR/MCT).

3 Display
The AUTO pushbutton switch light comes on.
The relevant thrust mode automatically engaged by the FMC appears in the LIM MODE display.
(Ref. Fig. 015)
(Ref. Fig. 016)
(Ref. Fig. 017)
(Ref. Fig. 018)



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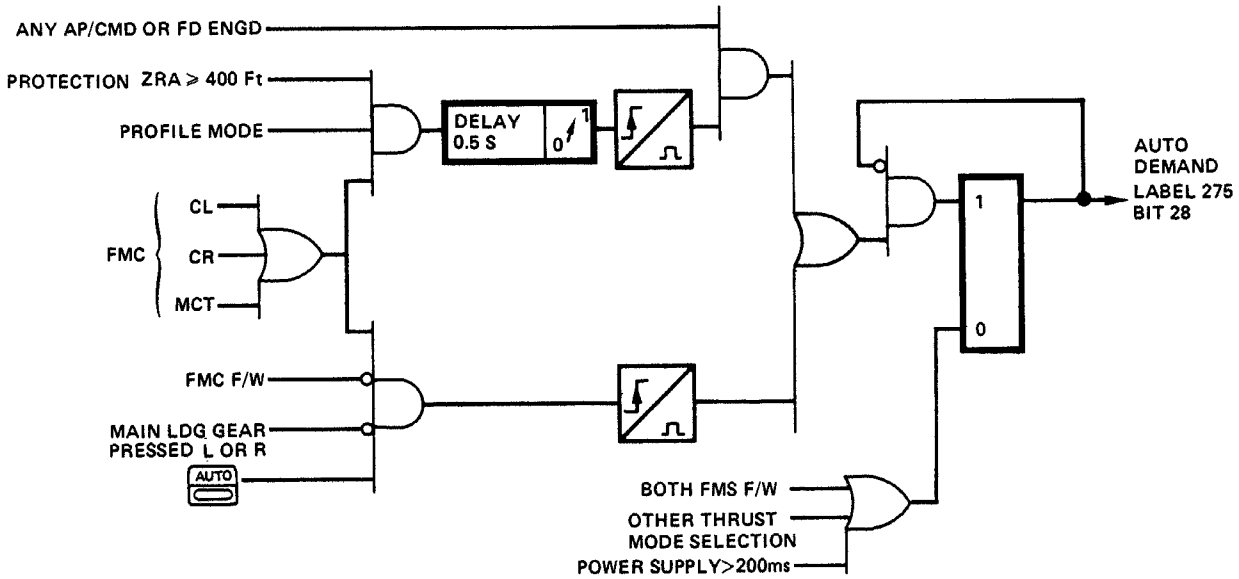
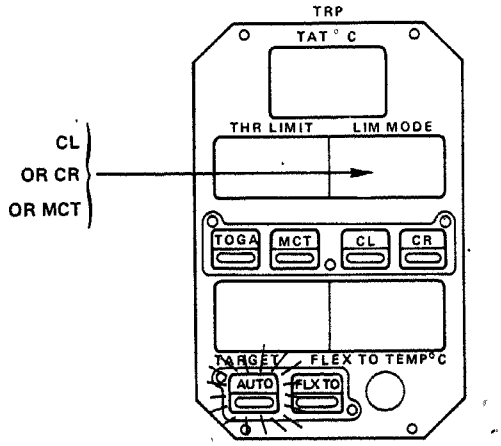
Climb/Cruise/Continuous Demand - Selection and Display
Figure 015





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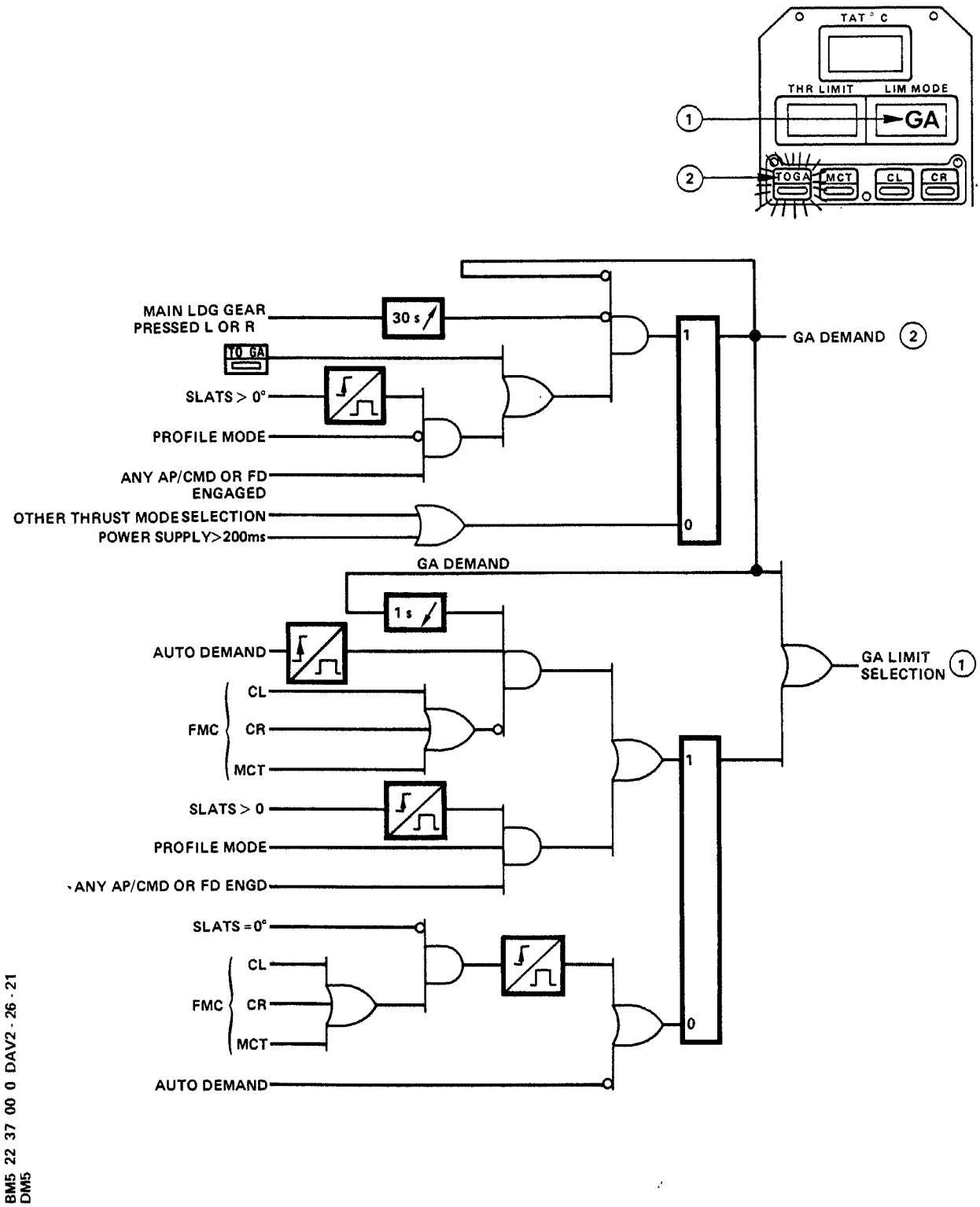
Auto Demand and Display
Figure 016



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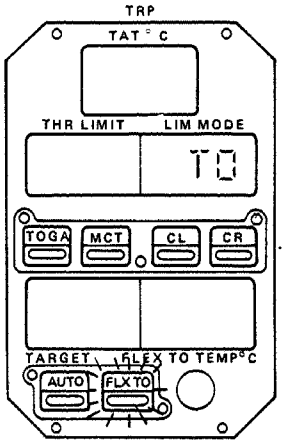
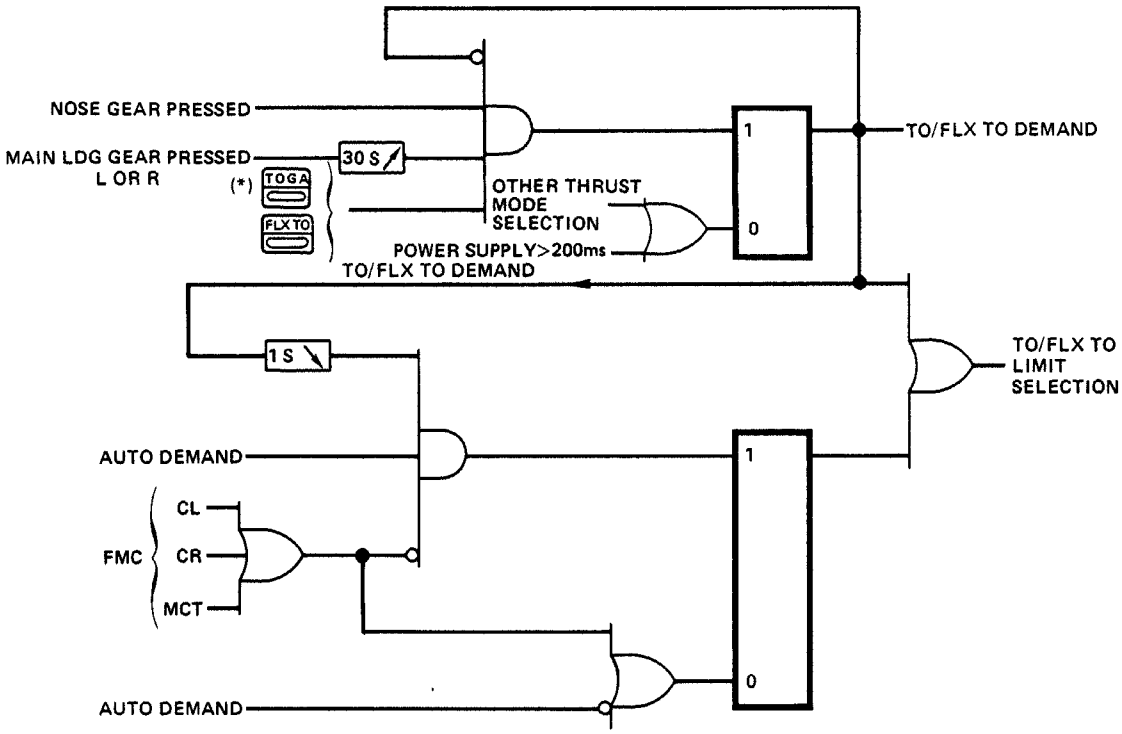
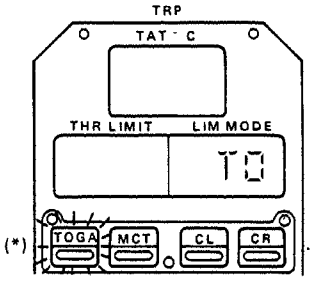
Go Around Demand - Selection and Display
Figure 017







(*) NOTE . TO MODE IS AUTOMATICALLY SELECTED AT POWER ON

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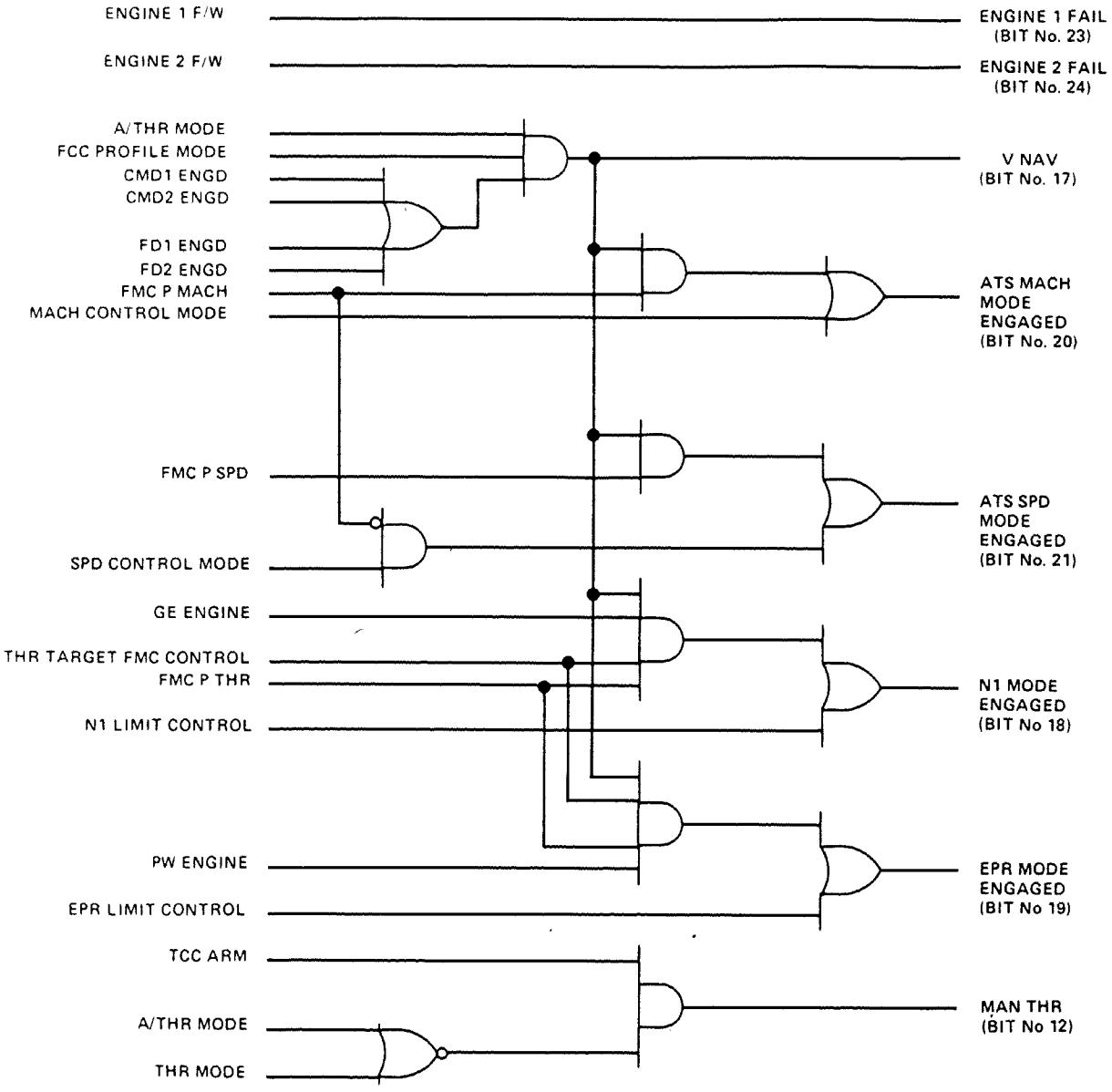
Take-off/FLX Take-off Demand Selection and Display
Figure 018

(14)Definition of label 274 discrete bits
The label defines the bit logics used in the processing of display conditions on the EFIS displays. The bit combinations are performed in the EFIS.
- Display (Ref. 22-35).
(Ref. Fig. 019)

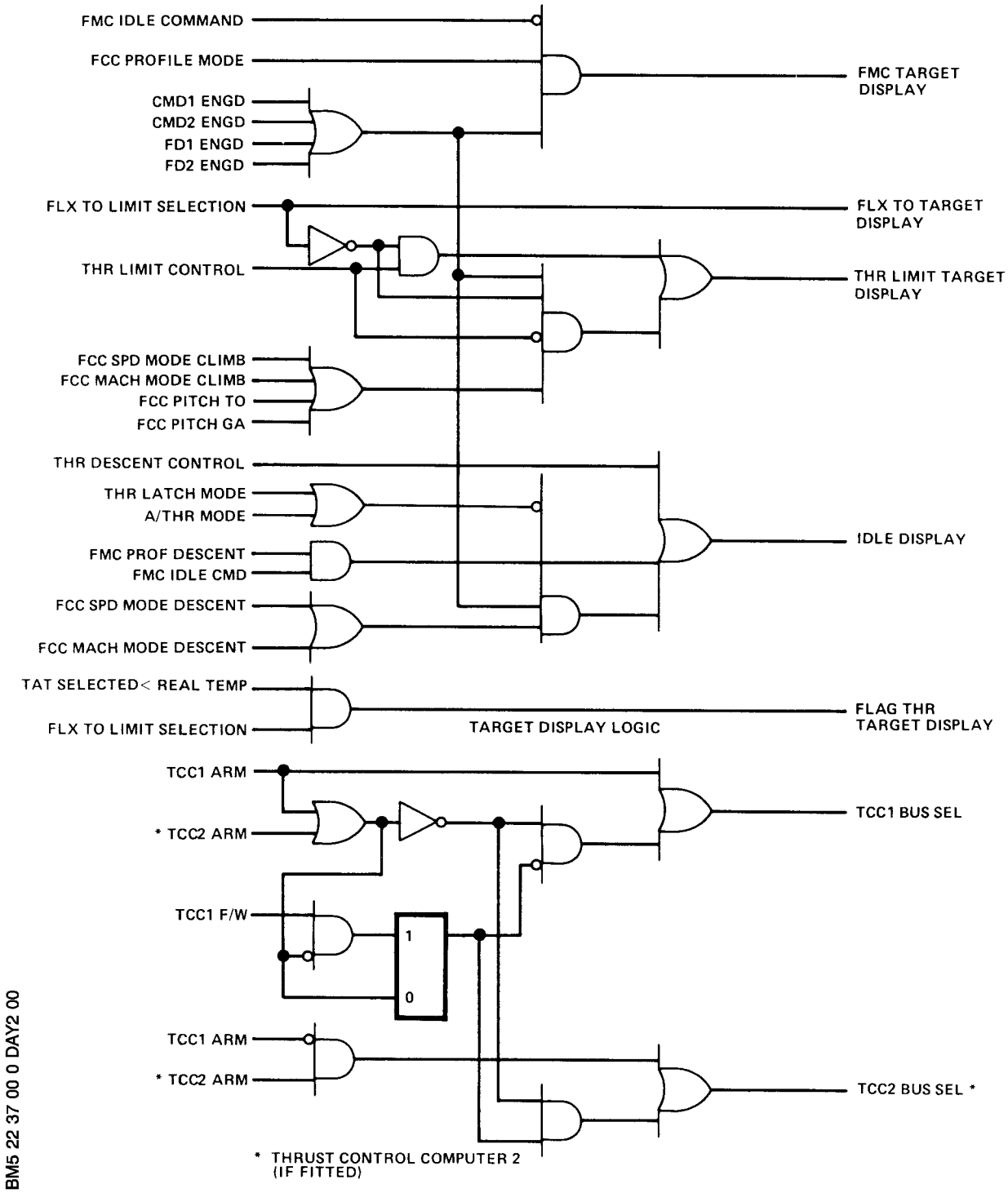
(15)Bus selection logic
(a)Flight control computer
FCC1 or FCC2 bus selection is made according to the commands delivered by the FCU.
(b)Flight management computer
FMC bus selection is given by the FCCs when PROFILE mode is engaged or in the case of limit mode management by detections of FMC F/W. FMC1 has priority, FMC2 is in standby.
(c)Flight augmentation computer
Selection of FAC1 or FAC2 data bus is made depending on the PITCH TRIM and YAW DAMPER lever engagement.
FAC1 has priority over FAC2.
In FAC1 PITCH TRIM engagement has priority over YAW DAMPER engagement.
When the slats are extended, selection of the FACs depends on the ADC selection made in the TCC.
(Ref. Fig. 020)

(16)Display of dashes on the TRP
(a)TAT display
Dashes appear in the TAT display if the TAT parameter on ADC1 and 2 is faulty or if the TRP ARINC receiver no longer operates.
(b)THR LIMIT display
Dashes appear in the event of THR limit computation failure or if the TRP is detected faulty.

(17)TARGET display logic
The TARGET display can show :
- the TARGET value given by the FMC
- the THR limit value if the THR LIMIT mode is active
- at takeoff when the FLX TO mode is selected, the THR limit value computed in relation to the selected flexible takeoff temperature
- I-L-when a throttle control lever retard is requested by the FCC (LEVEL CHANGE in DESCENT) or, when in PROFILE mode, the FMC generates an IDLE signal or when the A/THR or THR LATCH modes cannot be engaged and that a throttle control lever retard is requested by the FCC (see Fig. Para 2.B. (5)(b) 1)).



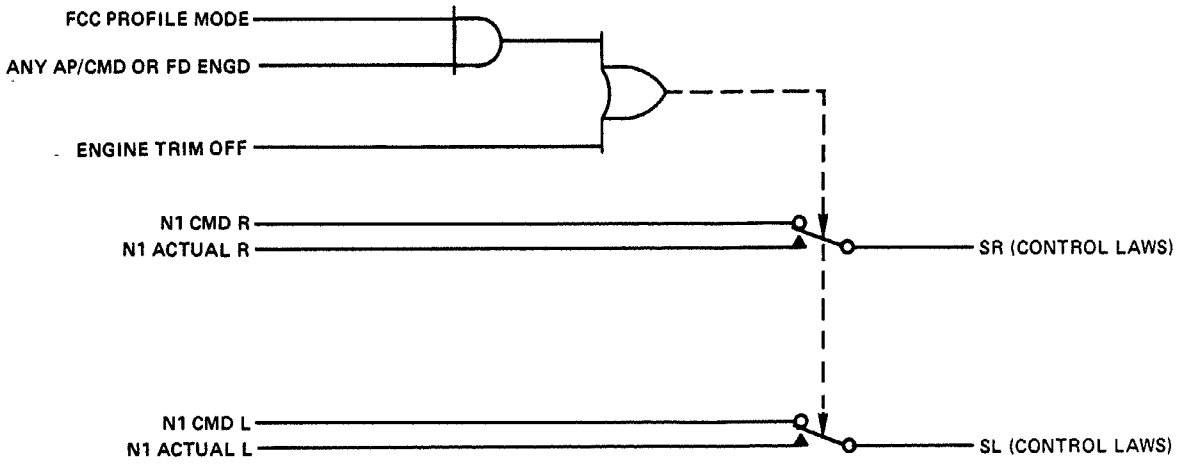
TCC Word ARINC Definition (Label 274)
Figure 019

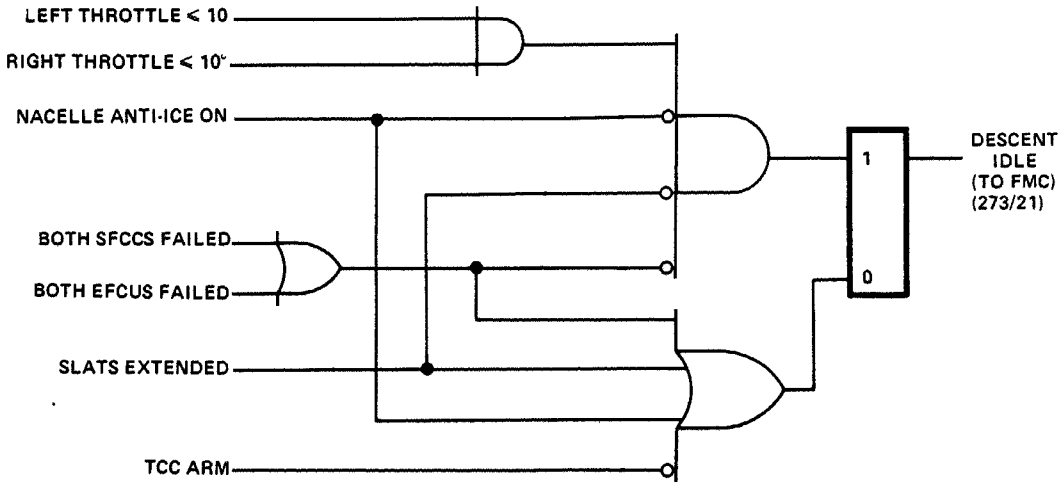


TCC/TRP Bus Validity
Figure 020

(18)TCC bus selection in the TRP
(a)In dual TCC configuration, TCC1 bus is selected.
In the event of TCC1 failure the TRP reads TCC2 bus.
If both TCCs are faulty, the TRP directly reads FMC1 bus to enable display of the TARGET value and ADC1 bus to enable display of ADC1 TAT.
(b)In single TCC configuration, TCC1 bus is selected.
If the TCC is faulty, the TRP directly reads FMC1 and ADC1 buses.
(19)Descent idle logic to FMC
(20)Logics associated with engine computers
(a)Selection of parameters given by the engine trim functions
Under normal operating conditions, i.e. without engine trim failure the control law is : N1 limit - max. N1 command.
After an engine trim failure the pilot stops system operation by pressing the ENG TRIM pushbutton switch. The N1 command signals are replaced by the N1 actual signals.
These signals are transmitted even after the engine trim functions have been disengaged. The TCC slaving law in N1 LIMIT mode is :
N1 limit - N1 actual (max. value of both engines).
(Ref. Fig. 021)

R (b)Engine trim condition (Ref. Fig. 022)
This logic is active during an engine trim failure or if the TCC detects a discrepancy in engine trim operation.
For each engine trim, this failure is detected by comparing the N1 limit value and the N1 command value sent back by each component. This comparison is active on condition that TO, FLX TO or MCT modes are not selected and that main landing gear is not extended and that the engine trim functions have not been disengaged by the pilot.





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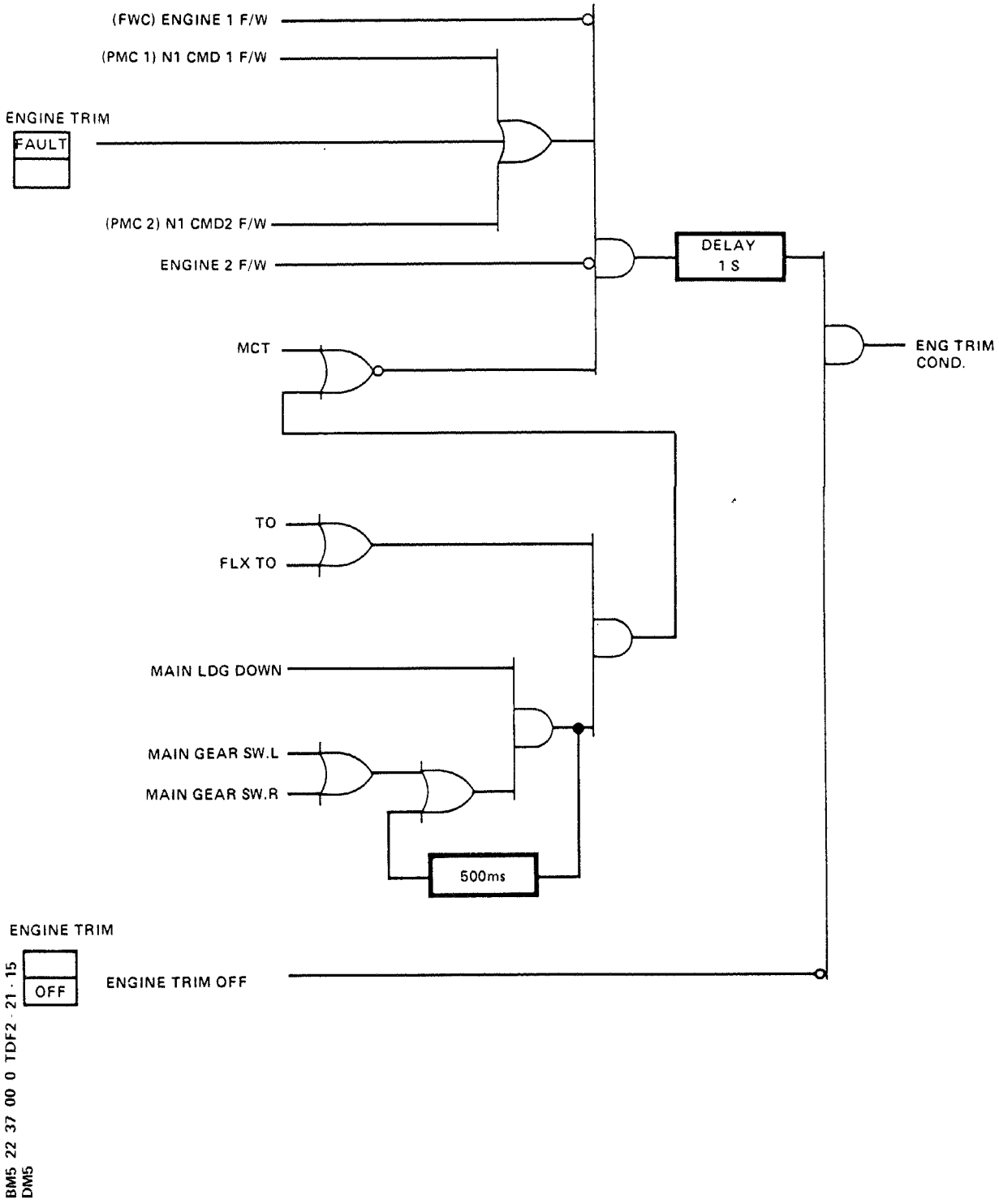
Engine Trim Functions/Selection of Parameters -
Descent Idle Logic to FMC
Figure 021



Engine Trim Condition
Figure 022

R

1. ATS Arming

A. ATS Arming Logic - General

(Ref. Fig. 001)

The lever is held in armed position if the monitoring and command logic circuits associated with the computer are valid.

It is to be noted for each computer that power is supplied to the clutch if :

- the monitoring and command logics are validated,
- the arming lever is in the **ON** position,
- self-latching relay is energized.

The self-latching relay is supplied via the arming lever and provides grounding of the common point of the clutches.

System disconnection causes the arming lever to return to the **OFF** position.

B. ATS Arming Logic - Principle

The arming logic is achieved in the command and monitoring channels.

Lever self-holding is possible only if the command and monitoring logics are correct.

(1) Disarming

(a) Signals directly prohibiting arming whatever the flight phase :

- disconnection of a dynamometric rod connector
- disconnection of the actuator or coupling unit connector
- arming circuit failure (self-latching relay)
- computer internal failure (components, power supply permanent failure, etc...).

(b) Signals directly causing disarming :

- triggering by watchdogs 1 or 2
- failure detected by comparators C1 or C2.

(c) Combined signals causing disarming (Ref. Fig. 002).

(d) Signals causing disarming but enabling reconfiguration (Ref. Fig. 003).

- loss of **ADC**1 or 2
- loss of one **FMC** when **AP**s and **FD**s are disengaged.
- in active **PROFILE** mode when in mode **SPD TARGET** or **THR TARGET** the signal is in **NCD** configuration.
- resolver failures linked to the throttle levers and detected by the **EECS**. These detections are active so long as the engines are not shut down.
- **ENG TRIM** failure detection by the **TCC** (channel **A** or **B** failures in one **EEC** or by volontary passing in **N1** mode by the crew) so long as the engines are not shut down.

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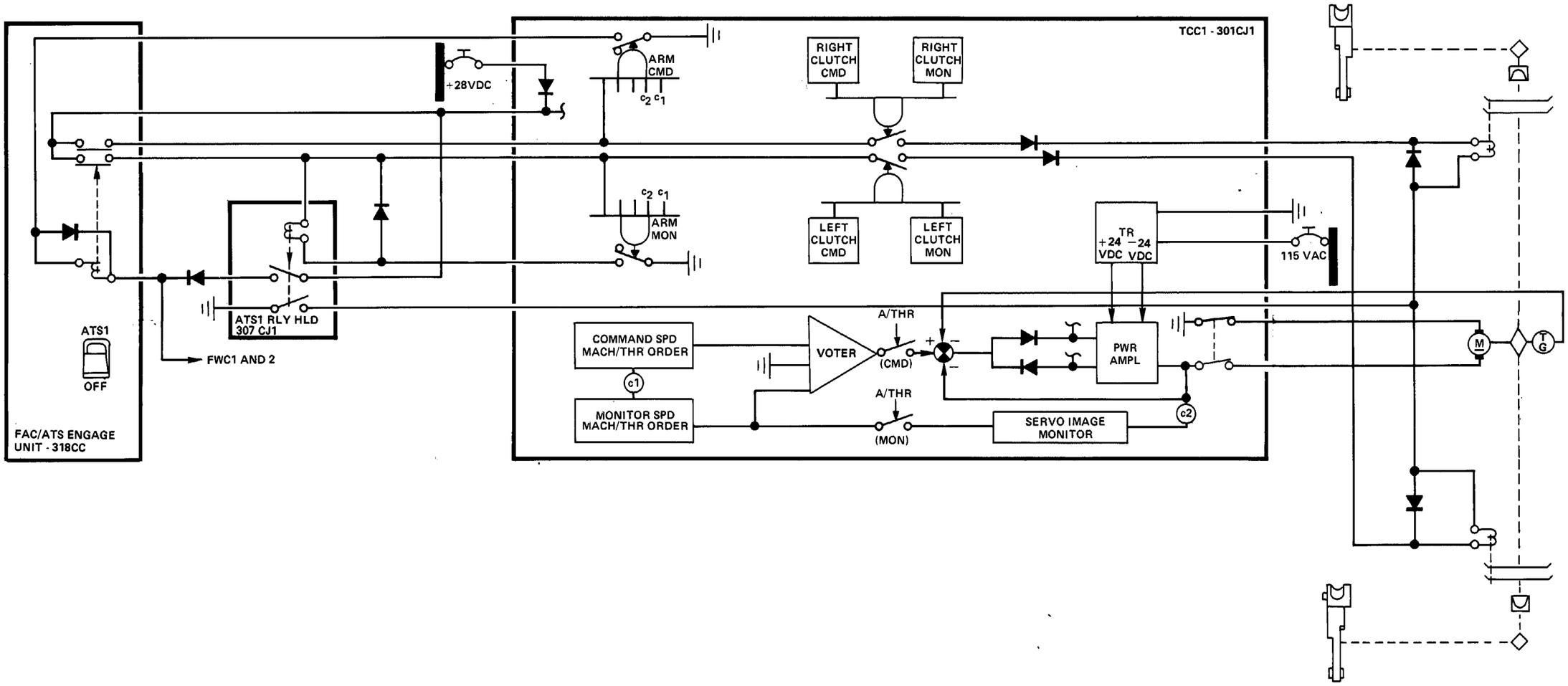

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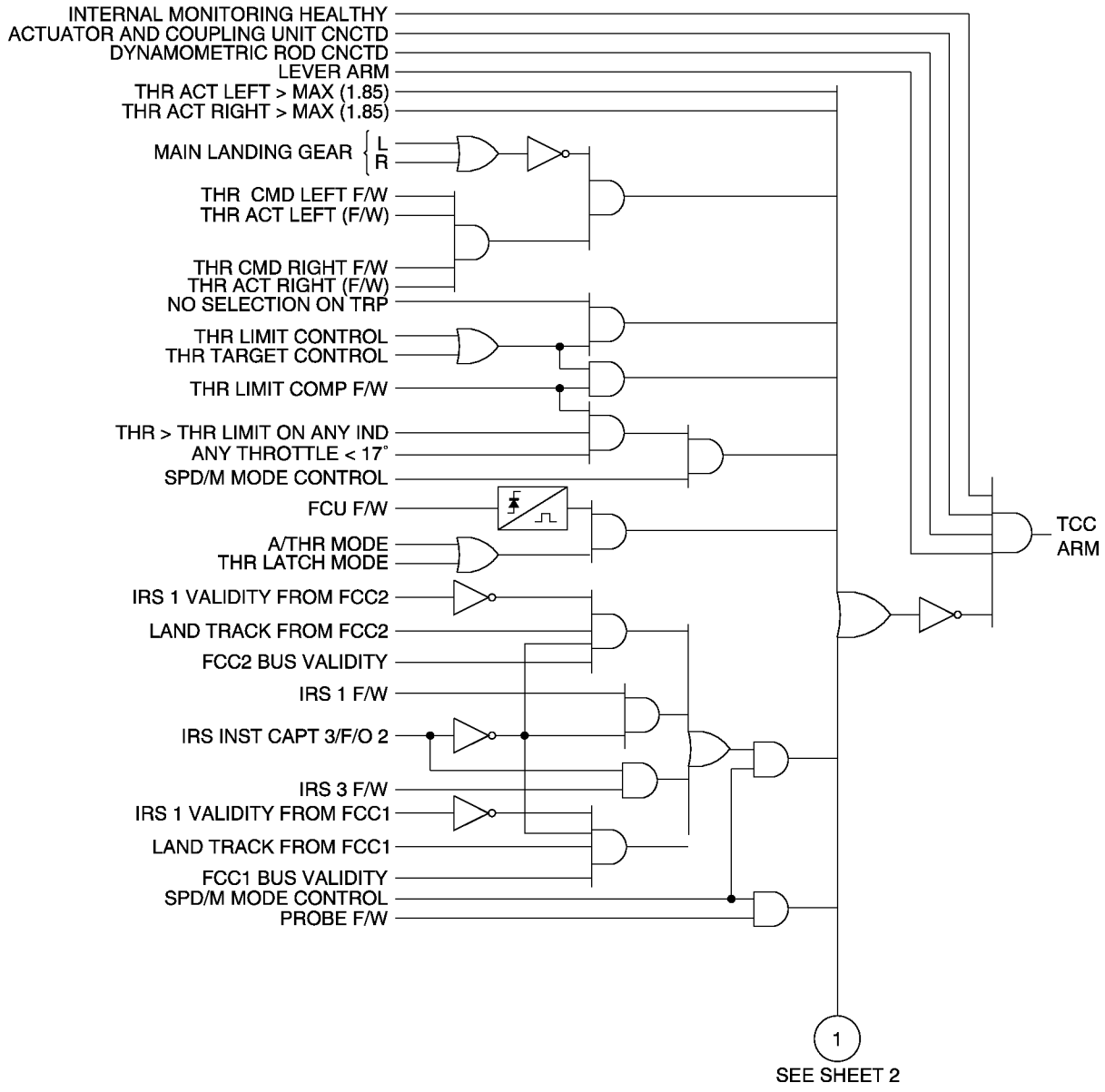
SPD/MACH Mode Control Laws and Display
Figure 001



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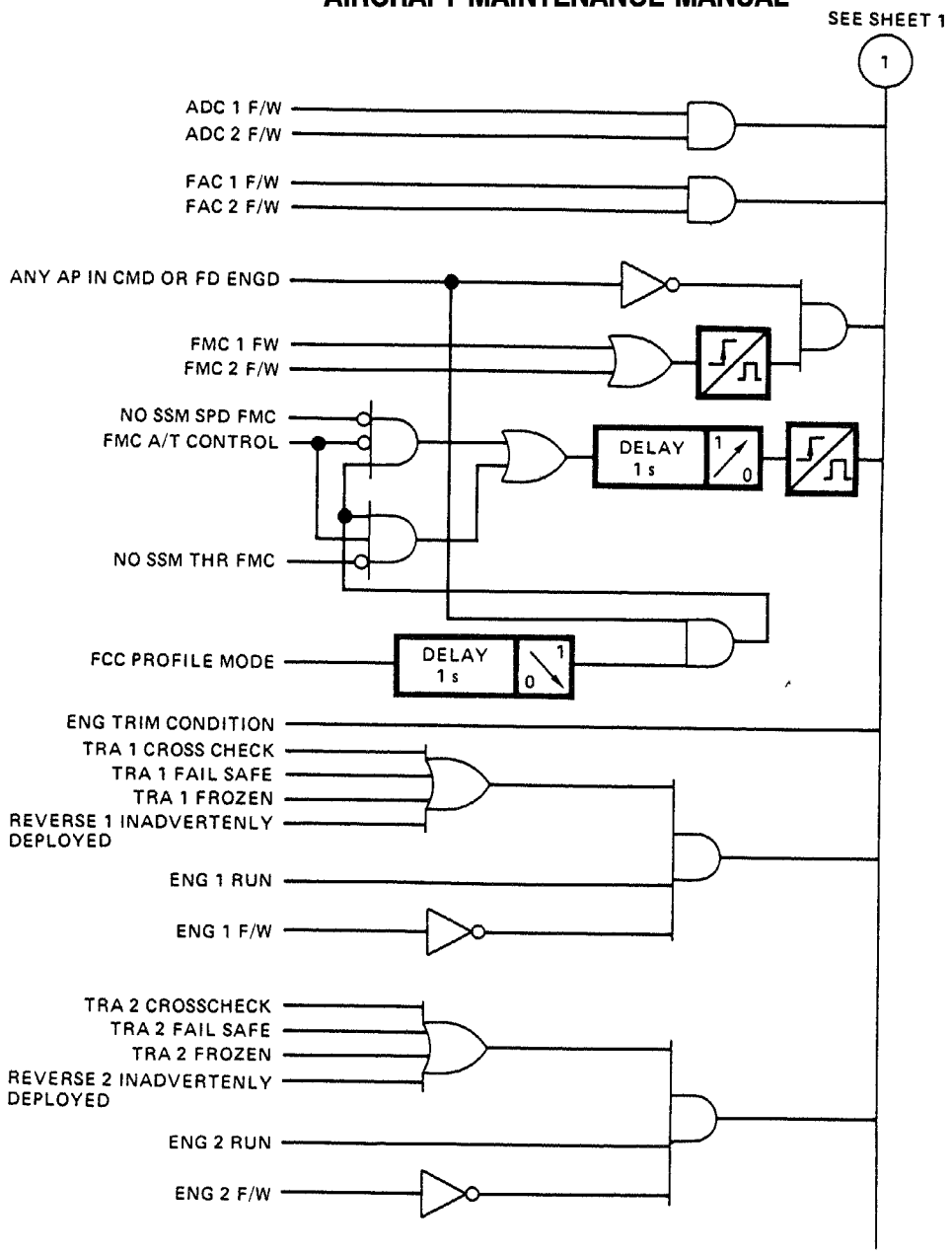
TCC Arming and Disarming Logic (Sheet 1/2)
Figure 002

R



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TCC Arming and Disarming Logic (Sheet 2/2)
Figure 003

2. ATS Engagement

A. Power Control (See 22-34).

B. Operating Modes

(1)Mode organization

Upon system arming, two main modes are available.

(a)A/THR mode

The A/THR mode corresponds to the automatic operation of the system in response to commands given either by the pilot or by peripherals (FCC, FMC).

(b)THR LATCH mode

This is a standby mode which enables recovery of the limit thrust of the mode selected on the TRP after operation of the TO/GA lever or if the alpha floor protection is activated (FAC).
(Ref. Fig. 004)

(2)A/THR mode

(a)Characteristics

The A/THR mode enables engine automatic control in various modes in relation to the peripheral system demands : FCC, FMC.

(b)Engagement/Disengagement

The A/THR engagement/disengagement conditions are given in following figure (Ref. Fig. 005).

(c)Display

On ground, green A/THR indication is displayed when A/THR pushbutton switch on FCU is pressed and if engines are stopped.
In flight, cyan A/THR indication is displayed after retard phase of LVL CHANGE in descent.

(3)THR LATCH mode

(a)Characteristics

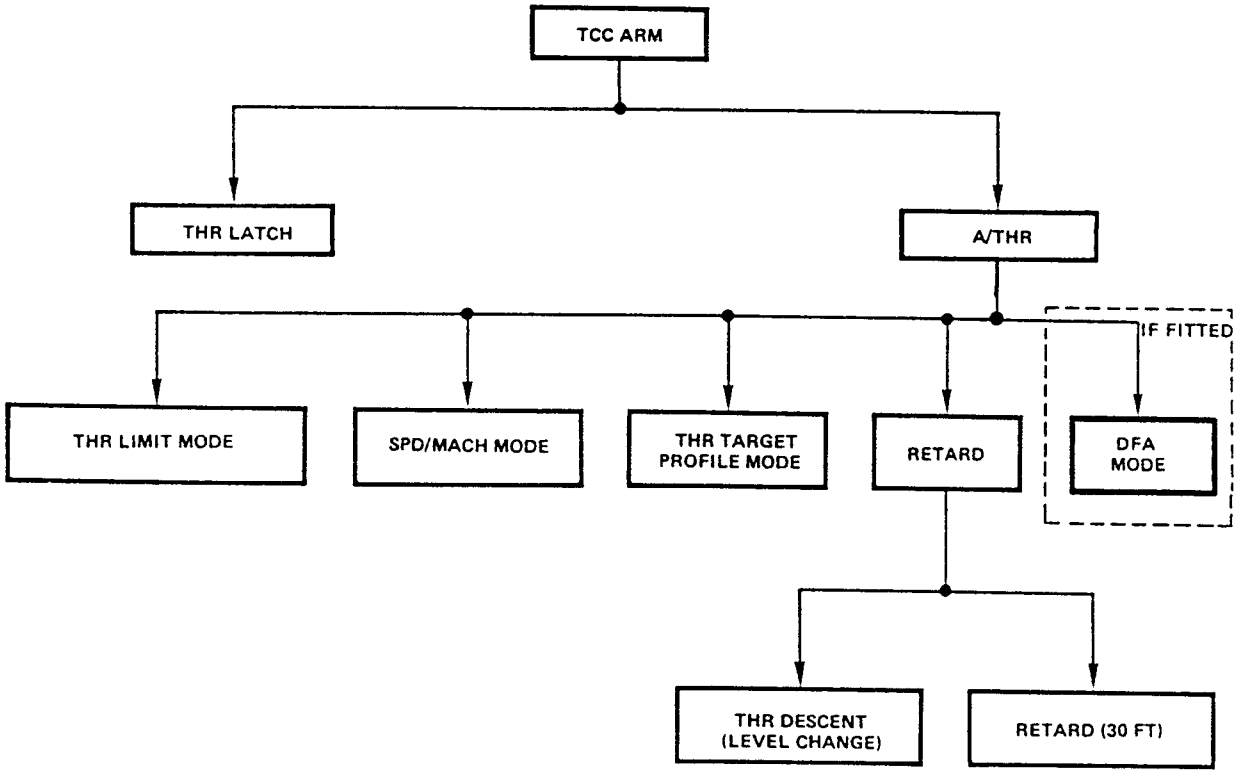
The THR LATCH mode is a standby mode. It is used to recover the limit thrust manually by means of TO/GA levers or automatically if alpha floor protection is requested by the FAC.

(b)Engagement/Disengagement

NOTE : Engagement is made possible if the TCC is armed and if the limit thrust is not already requested by the A/THR automatic mode. This mode cannot be engaged if AP is in CMD as soon as main GEAR is pressed (LAND phase).

(c)Display

When the THR LATCH mode is engaged out of alpha floor protection, the green THR L indication flashes in the ATS mode column. (The green THR L indication is steady with alpha floor condition).
(Ref. Fig. 006)



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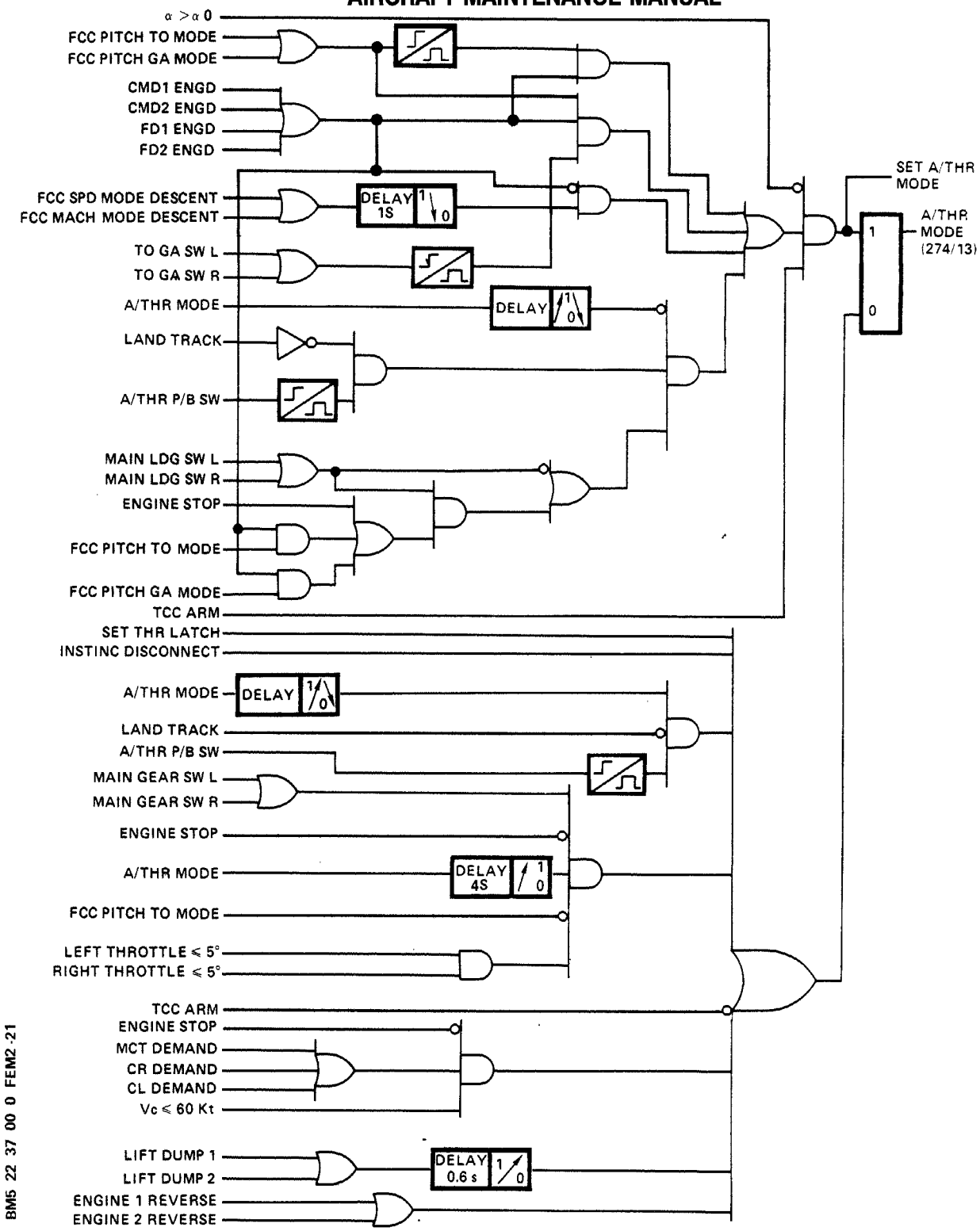
Mode Organization
Figure 004



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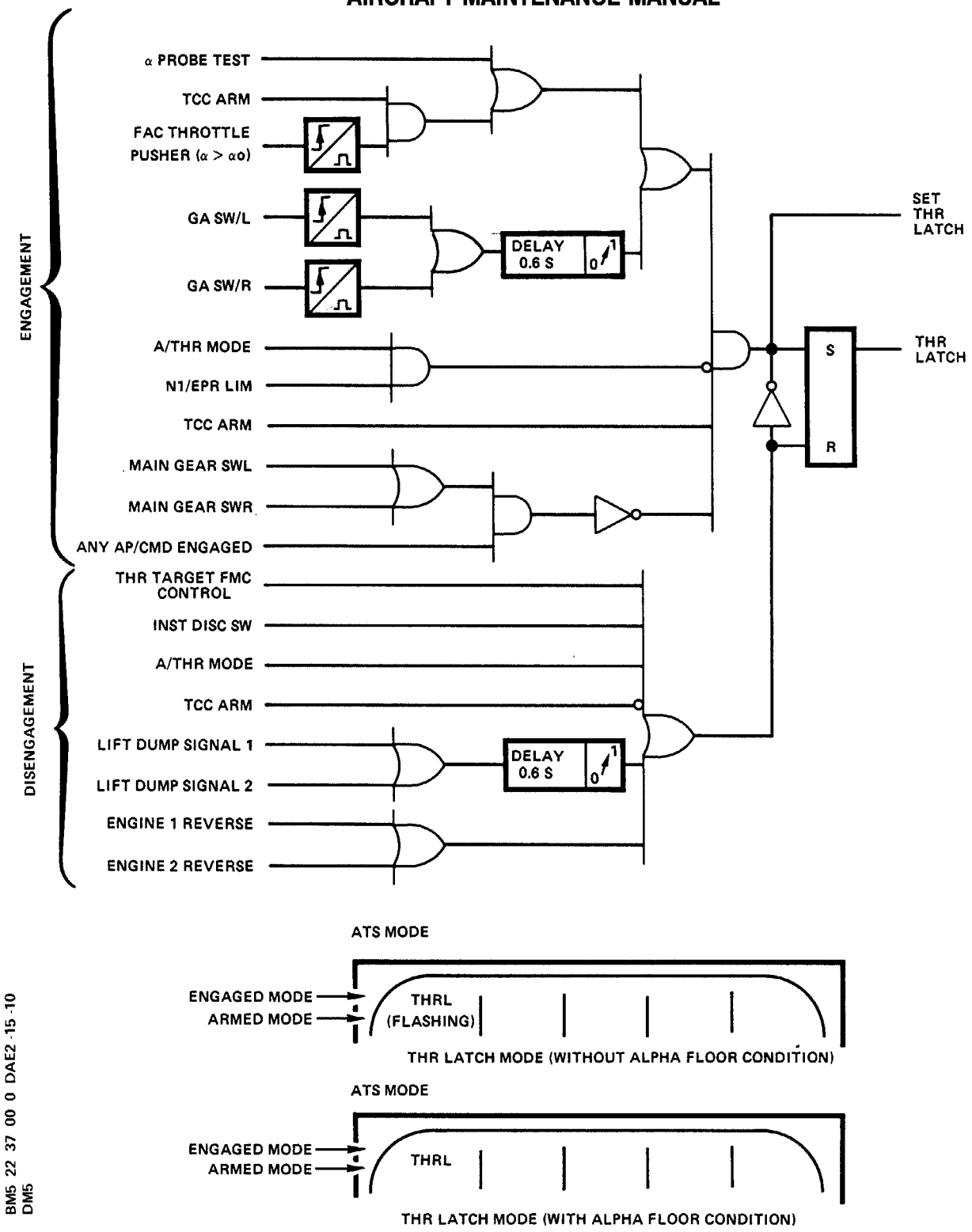
A/THR Mode - Engagement/Disengagement Conditions
Figure 005



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THR Latch Mode - Engagement/Disengagement Conditions and Display
Figure 006

(4)SPEED/MACH mode control

(a)SPEED hold characteristics

This mode is used to maintain automatically the speed selected on the FCU through the TCC, slaving feedback being provided by the aircraft speed given by ADC1 (or ADC2 if ADC1 failed).
Vmin and Vmax values generated by the FAC are used as protections.
Speed cannot be greater than Vmax or lower than Vmin.
In SPD hold mode in approach phase and in flap extended configuration, throttle control lever displacement speed signal is introduced at flap extension.

(b)MACH hold

Same as SPEED hold (with MACH from FCU and ADC).
Vmin and Vmax protection is also performed.

(c)SPD/M protection by THR channel

When in speed or Mach acquisition mode and in order to prevent exceeding EPR limit, automatic switching is performed on the THR channel when the speed channel output command becomes greater than the command given by the THR channel. This switching is active all the time a speed command is greater than the THR command.

(d)Display

Green SPD or MACH is displayed in the ATS Mode column on the PFDs.
Green P SPD or P MACH comes on when SPD or MACH mode is engaged upon FMC request.

(e)Engagement/Disengagement

When LVL/CH is active in descent phase, disengagement of AP/FDs must force the automatic engagement in SPD mode.
(Ref. Fig. 007)
(Ref. Fig. 008)

(5)Throttle control lever automatic retard

The throttle control levers can be moved automatically towards the idle position when in one of the following phases :
- descent controlled by the FCC (LVL/CH)
- profile controlled by the FMC (FMC idle command)

NOTE : In these cases the throttle control levers move to the idle position at 1°/s (THR Descent Control).

- at 30 ft.

NOTE : In this case the throttle control levers move to the idle position at 2°/s (Throttle control lever retard during FLARE).

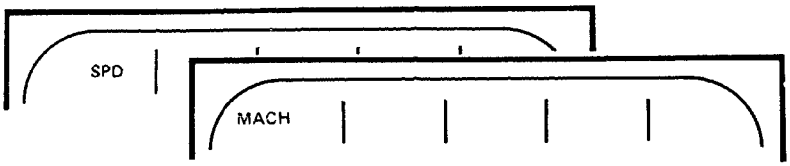
During the retard phase, the RETARD indication on the PFDs comes on.
When idle is reached or if the control lever movement is stopped during the retard phase in THR DESCENT or IDLE FMC, the cyan A/THR indication on the PFDs comes on.
(Ref. Fig. 009)
(Ref. Fig. 010)

(6)DFA mode

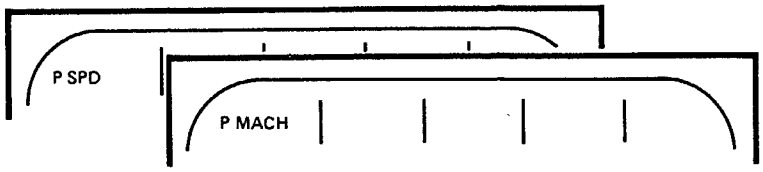
Not applicable

(7)THR TARGET mode - FMC control

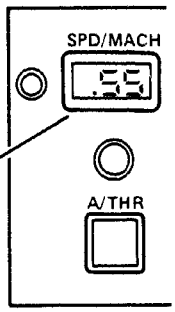
The THR TARGET mode is engaged automatically if the FCC is engaged in PROFILE mode so that the TCC acquires the THR TARGET signal delivered by the FMC.
The signal delivered by the FMC is limited in its upper limit by the THR



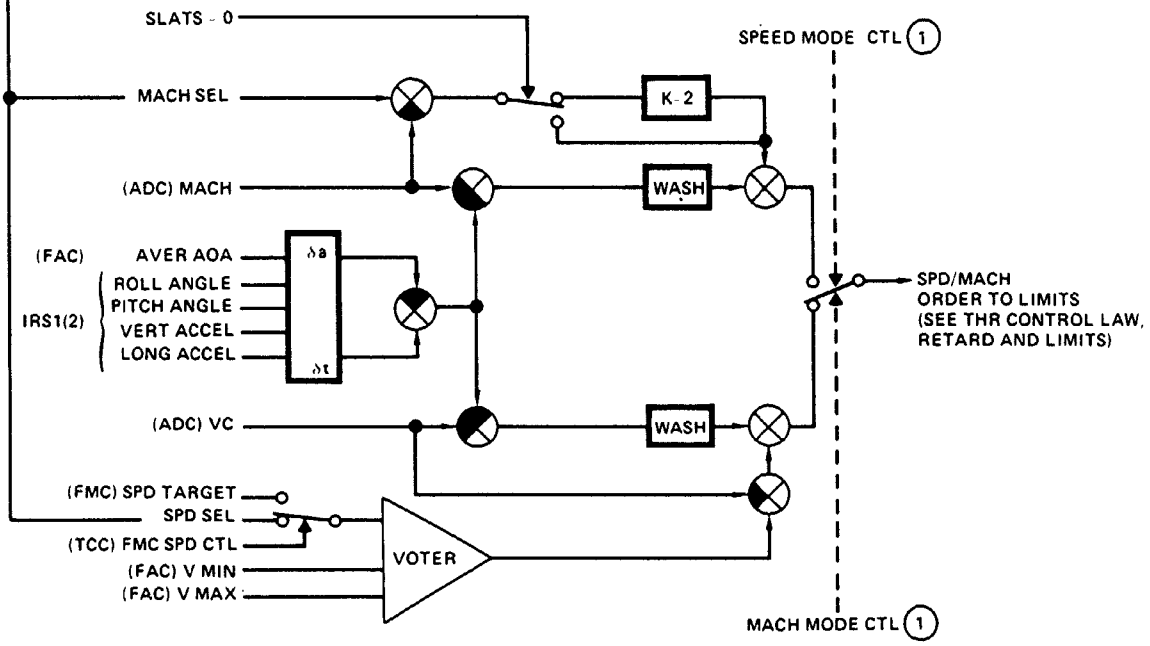
SPD OR MACH MODE ENGAGED BY FCC (OUT OF PROFILE MODE) OR MANUALLY BY A/THR ENGAGEMENT. FCU SPEED OR MACH DISPLAY GIVES THE VALUE TO BE FOLLOWED.



SPD OR MACH MODE ENGAGED BY FCC/FMC. FMC GIVES THE TARGET SPEED TO BE FOLLOWED.

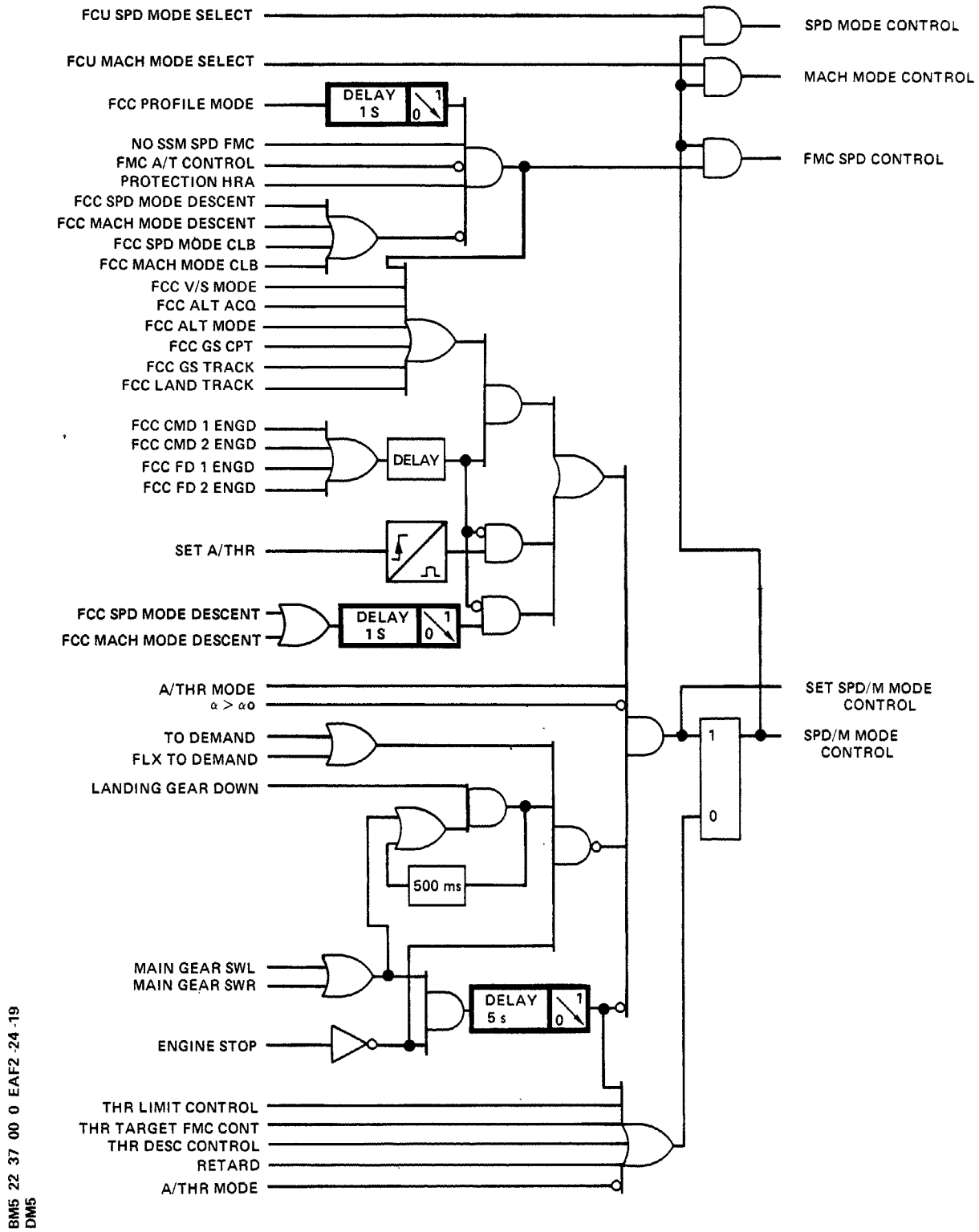


NOTE (1) IN MACH MODE CTL, REVERSION TO SPD MODE CTL IS PERFORMED IF Vc > VMAX - 5Kt OR IF Vc < VMIN + 5Kt UNTIL CONDITION IS PRESENT

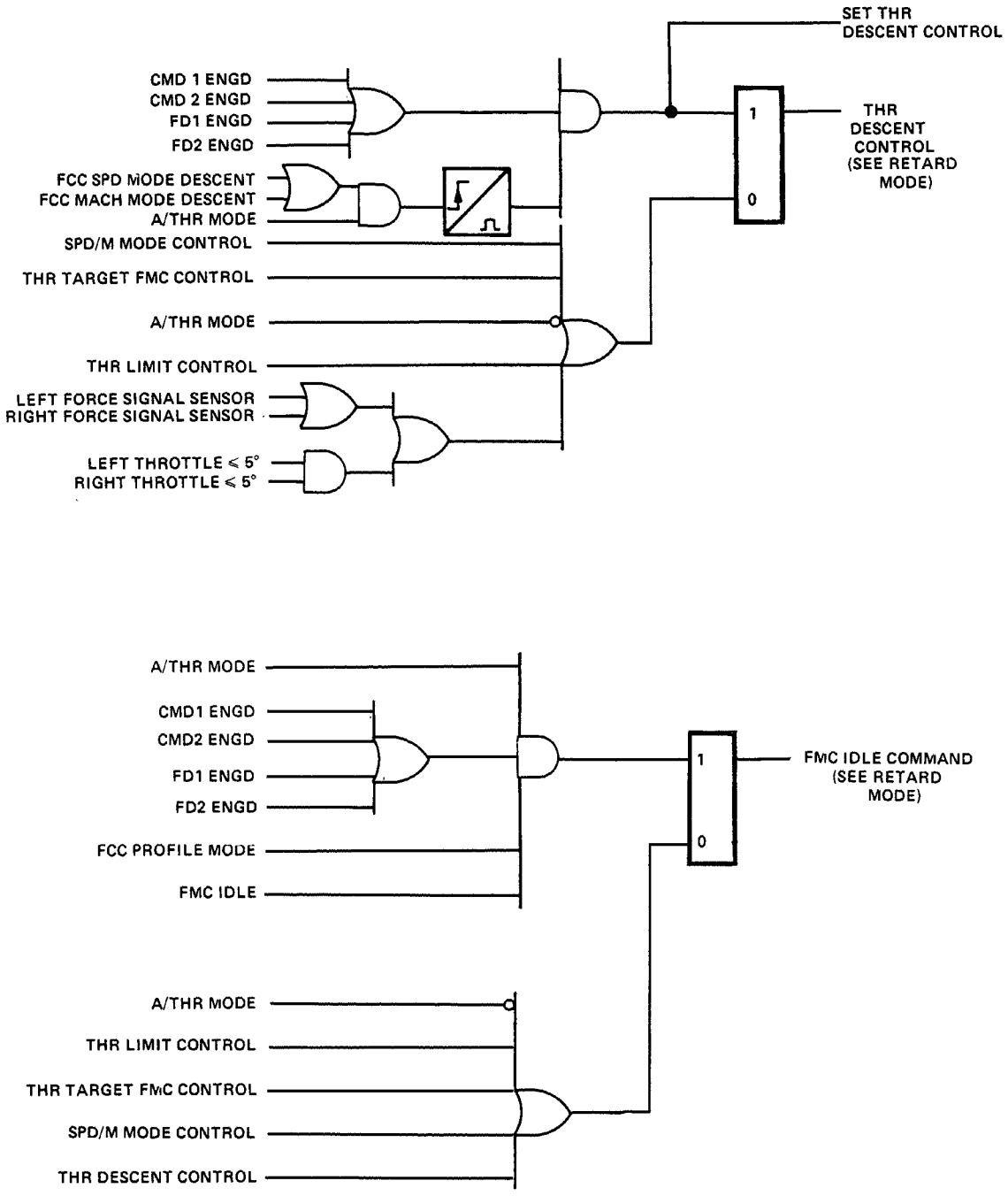


BM5 22 37 00 0 DAG2.13

SPD/MACH Mode Control Law and Display
Figure 007

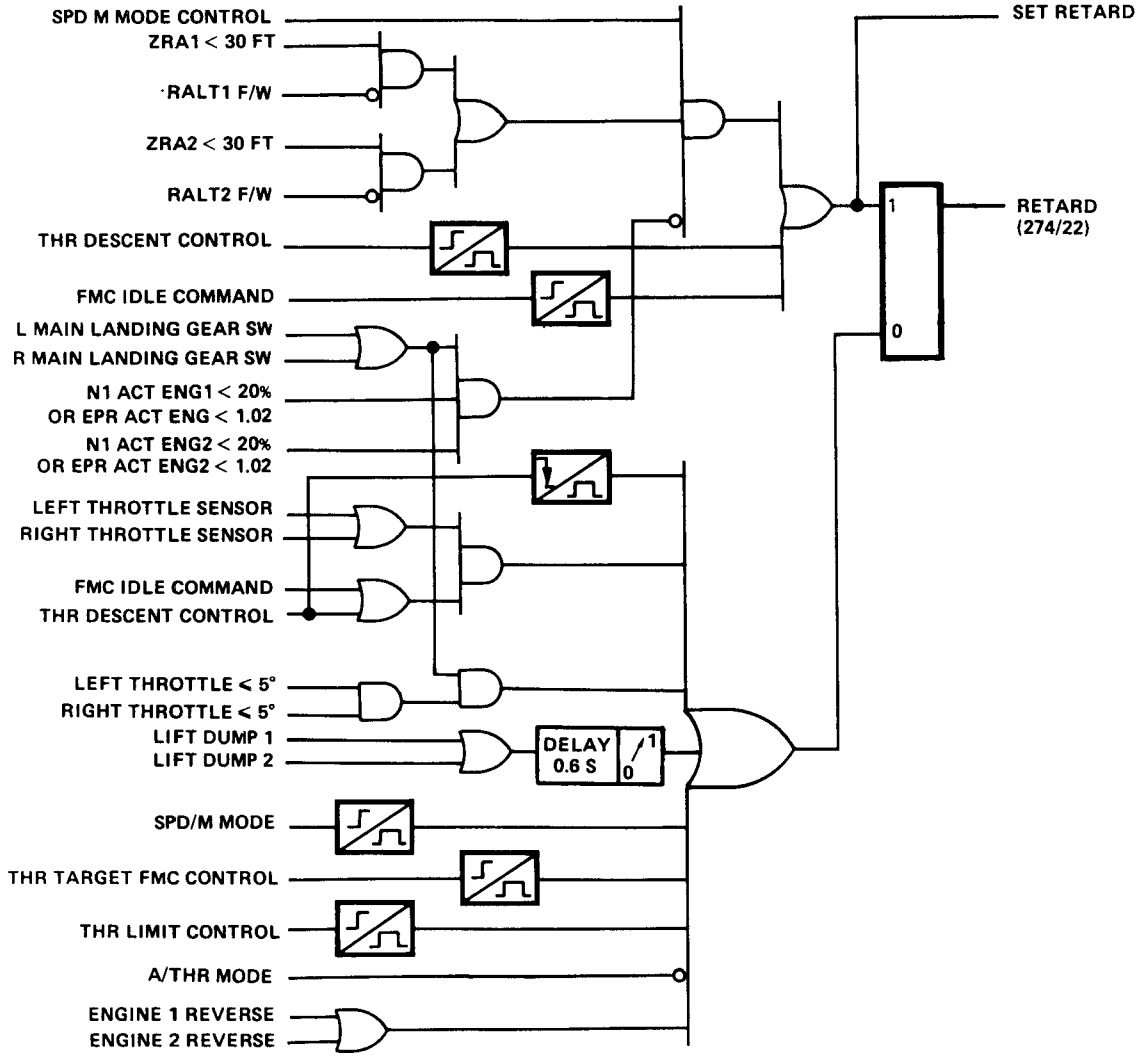


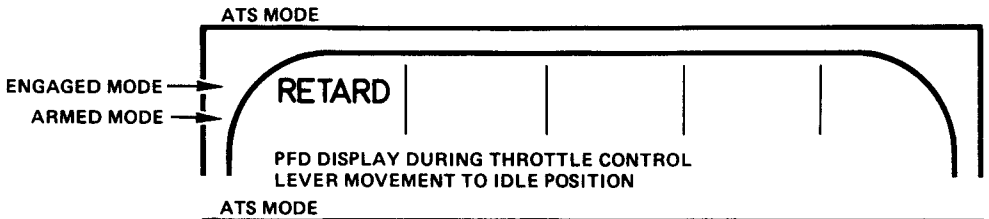
SPD/MACH Mode Control - Engagement/Disengagement Conditions
Figure 008



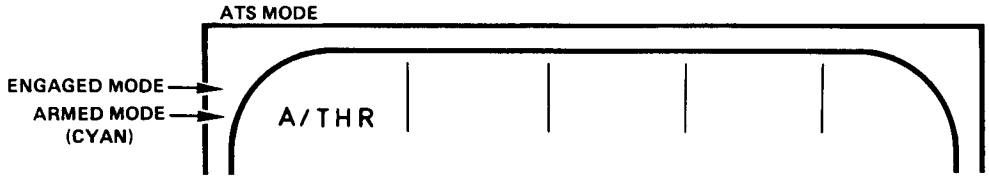
BM5 22 37 00 0 FAE2-16-11
DM5

THR Descent Control/FMC Idle Command
Figure 009





PFD DISPLAY DURING THROTTLE CONTROL LEVER MOVEMENT TO IDLE POSITION



PFD DISPLAY . WHEN THROTTLE CONTROL LEVER MOVEMENT TO IDLE POSITION COMPLETED OR STOPPED BY THE PILOT IN THE DESCENT CONTROL OR FMC IDLE COMMAND.

BM5 22 37 00 0 DAK2 01

Retard Mode Logic and Display
Figure 010

LIMIT signal and in its lower limit by the P.CLIMB condition.
Engagement/Disengagement conditions and display of FMC THR TARGET mode are given in following figure
(Ref. Fig. 011).

(8)EPR LIMIT mode control (Ref. Fig. 012)
The EPR LIMIT mode enables acquisition of the limit rating corresponding to the mode selected on the TRP.
Take-off limit thrust corresponds to TO or FLX TO thrust.
Both throttle control levers are operated by the autothrottle actuator and move at 3°/s to the corresponding position where the EPR command of the considered engine is equal to the EPR limit.
The corresponding throttle control lever is declutched when this condition is fulfilled.
Systematic declutching occurs at 100kt to protect the system against untimely throttle control lever retard during take-off.

(a)EPR LIMIT control in flight
The slaving law is : EPR limit of selected mod-greater EPR command.
The actuator stops operating (no declutch).

(b)Engagement/disengagement (Ref. Fig. 013)

(c)Display (Ref. Fig. 013)
Green THR is displayed when the thrust mode is active with A/THR function engaged.

(9)Declutch logic
As the TCC is armed when a mode is selected, the throttle control levers are clutched.
They can be declutched :
- separately in TAKE-OFF mode when the conditions on EPR limit - EPR command and THR limit control are obtained.
- Simultaneously at 100kt, if the conditions previously mentioned are not obtained.
- If the pilot applies a load on a throttle control lever, the corresponding clutch is declutched. When the load is no longer applied the lever is clutched.

(a)Display (Ref. Fig. 013)
The cyan THR indication appears when declutching occurs in take-off phase.
Cyan THR indicates that both throttle control levers are declutched but THR mode is always engaged.

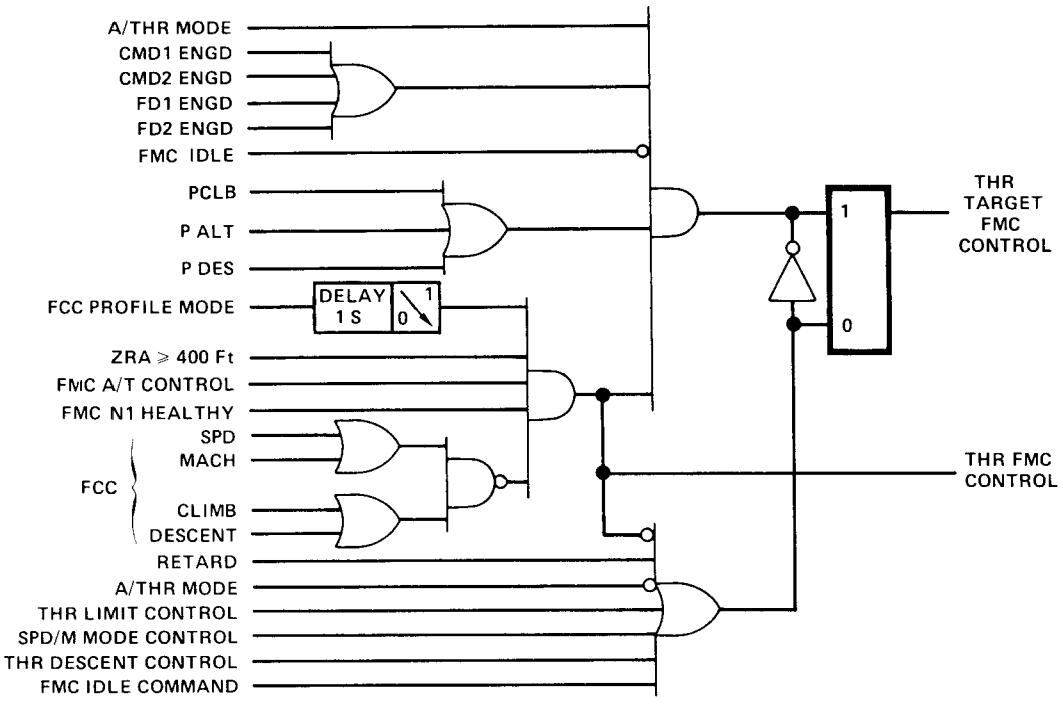
(10)Throttle control lever displacement stops (Ref. Fig. 012)

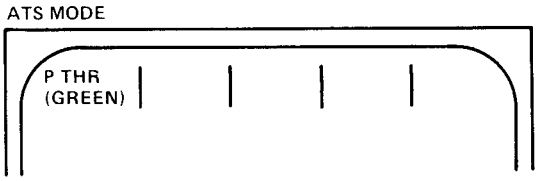
(11)THR limit computation failure
In the event of a THR limit failure, only SPEED mode can be selected in automatic mode.

(12)Throttle pusher (alpha floor)

(a)Engagement
Protection against high angle-of-attack is active as soon as the TCC is armed. To this end, the TCC is directly provided with the alpha floor command signal coming from FAC1 or FAC2 depending on the position of the PITCH TRIM and YAW DAMPER engage levers.

(b)Display
The green THR L indication comes on steady on the PFDs as long as the alpha floor condition is present. The indication flashes as soon as





THR ENGAGED BY FCC/FMC. FMC GIVES THR TARGET (DISPLAYED ON TRP) TO BE FOLLOWED BY ATS

BM5 22 37 00 0 TAL2 00

FMC THR TARGET Control - Engagement/Disengagement and Display
Figure 011

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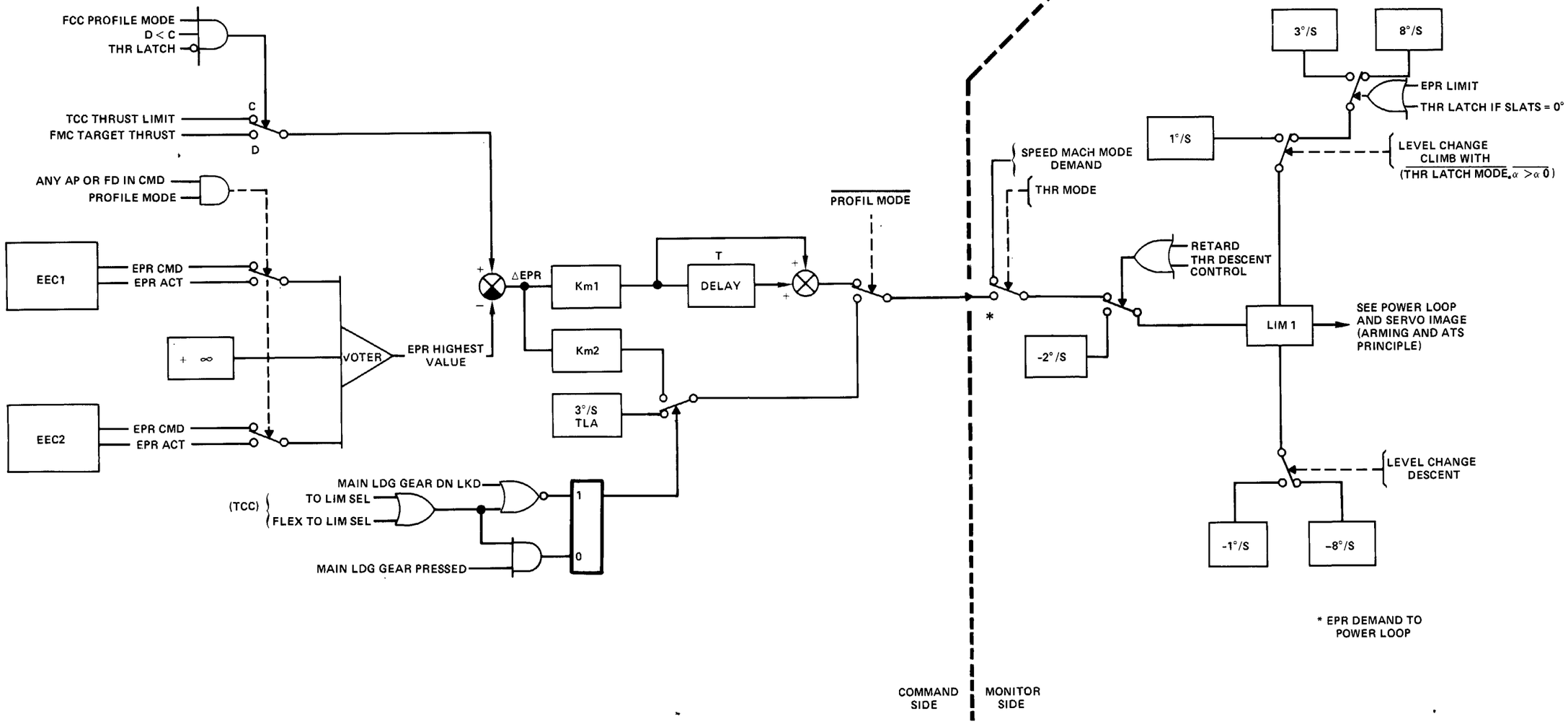
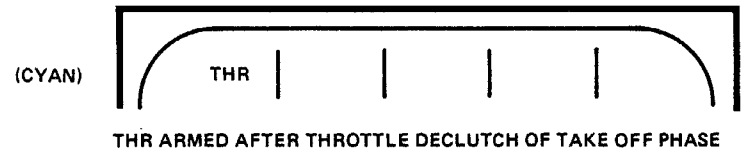
Printed in France

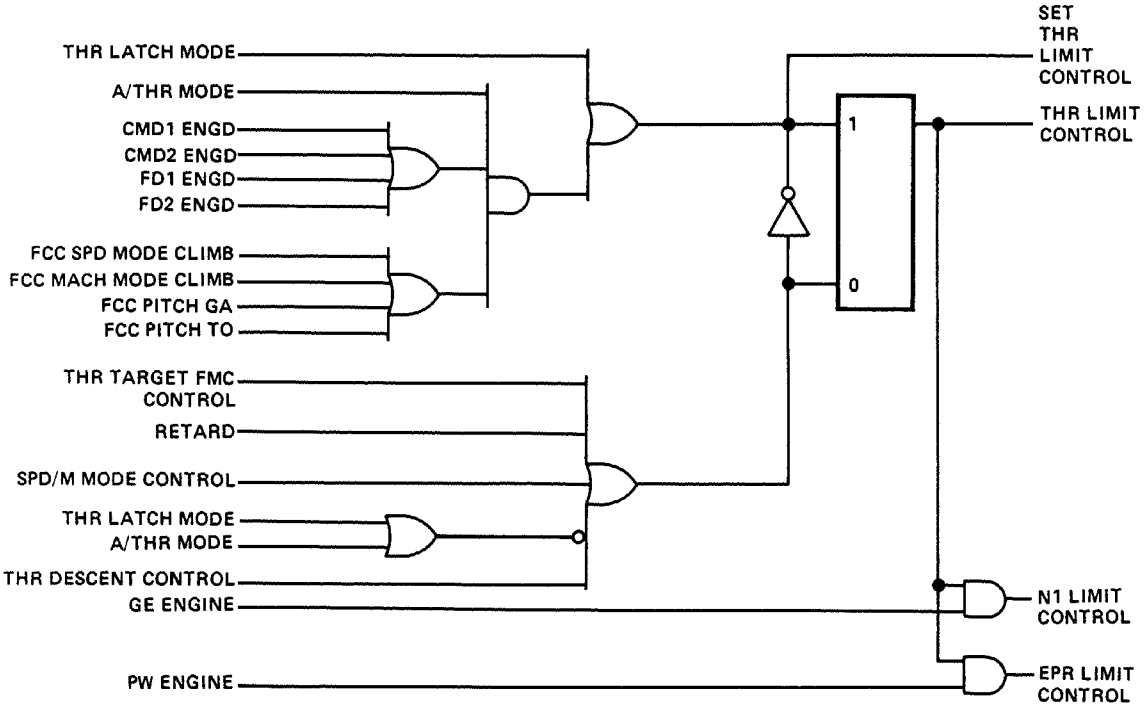


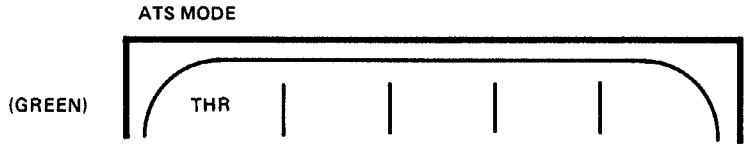
BM5 22 37 00 0 FRM2 22

EPR Control Law
Figure 012



THR ARMED AFTER THROTTLE DECLUTCH OF TAKE OFF PHASE



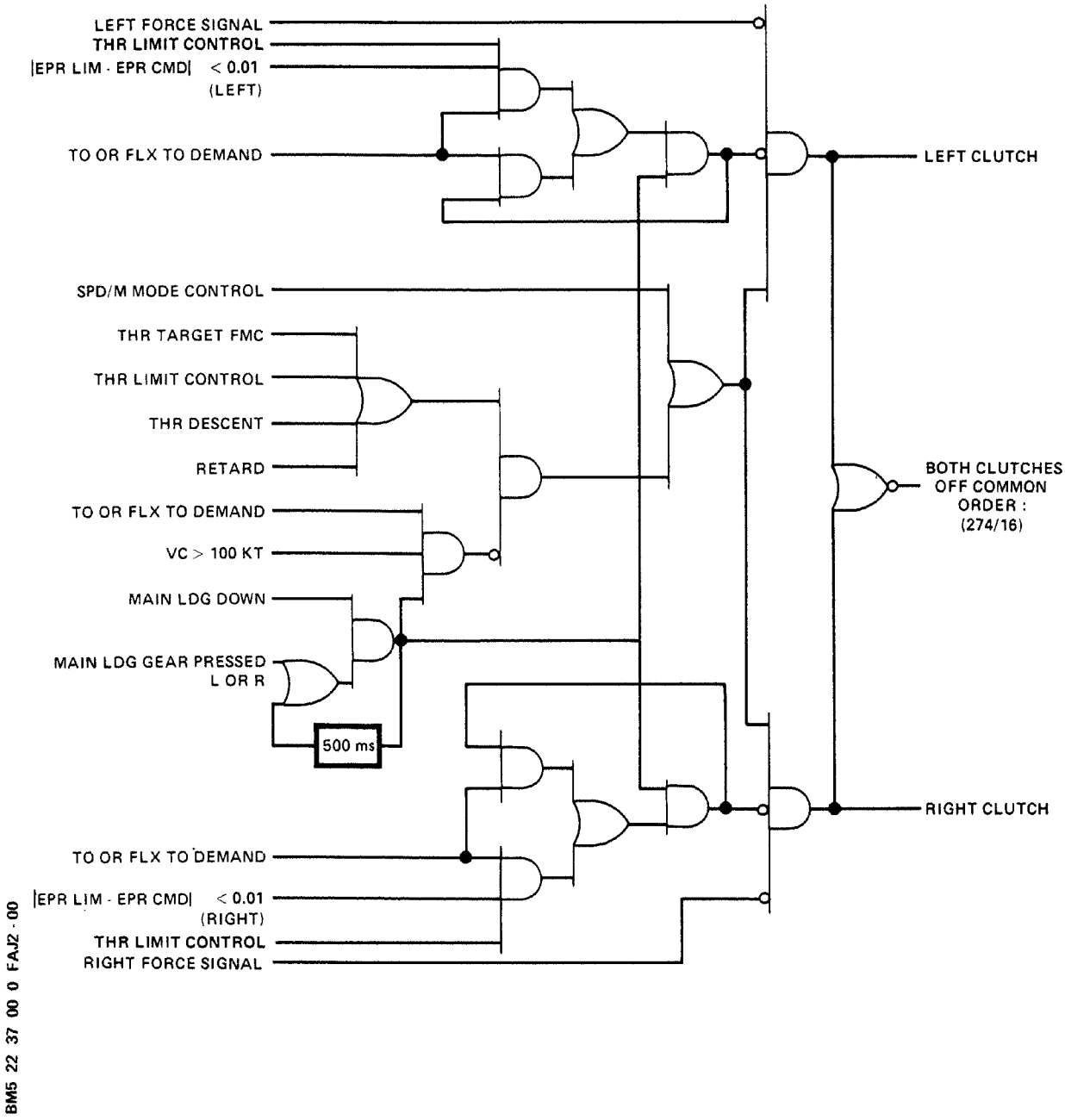


THR ENGAGEMENT BY FCC OR MANUALLY BY TO/GA LEVERS.
TCC GIVES THR LIMIT TO BE FOLLOWED.

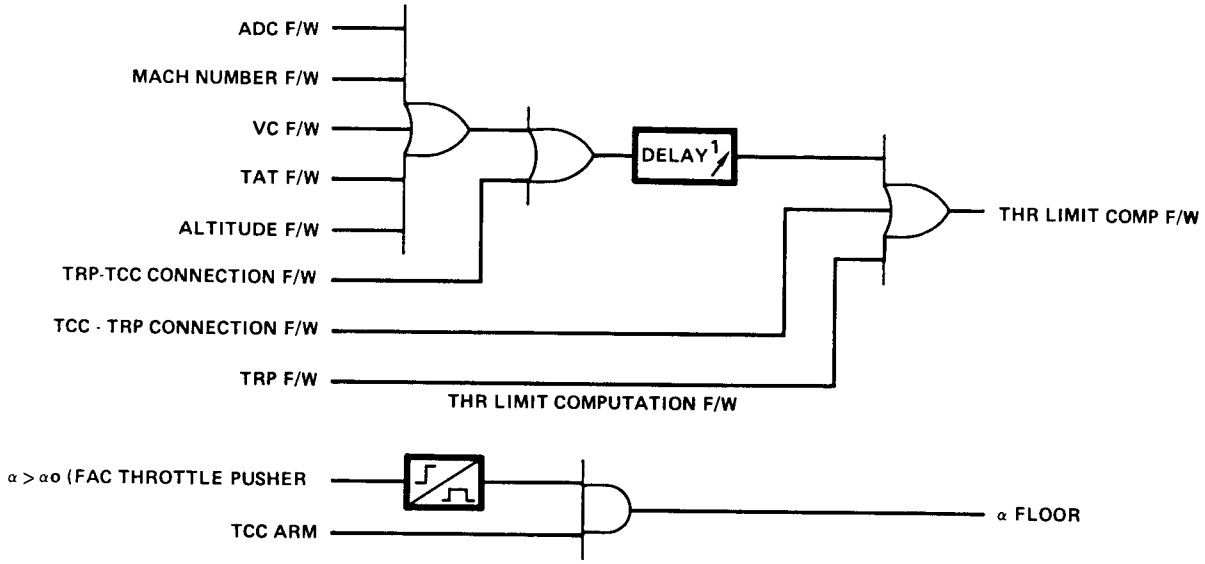
BM5 22 37 00 0 DAN2 15 10
DM5

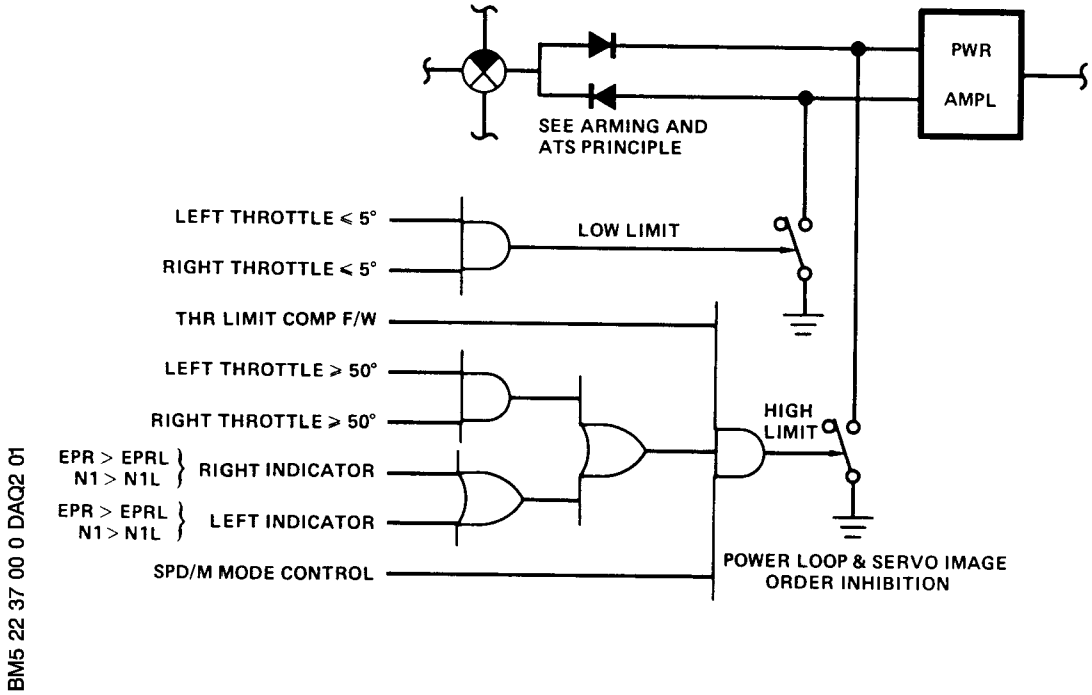
THR Limit Mode Control
Figure 013

this condition disappears
(Ref. Fig. 014).
(Ref. Fig. 015)



Declutch Logic
Figure 014





BM5 22 37 00 0 DAQ2 01

Throttle Control lever Stops/THR LIMIT Comp F/W/Alpha Floor Logics
Figure 015

(13)Thrust selection management
This function is used to select different thrust ratings (function of flight phase or aircraft weight).

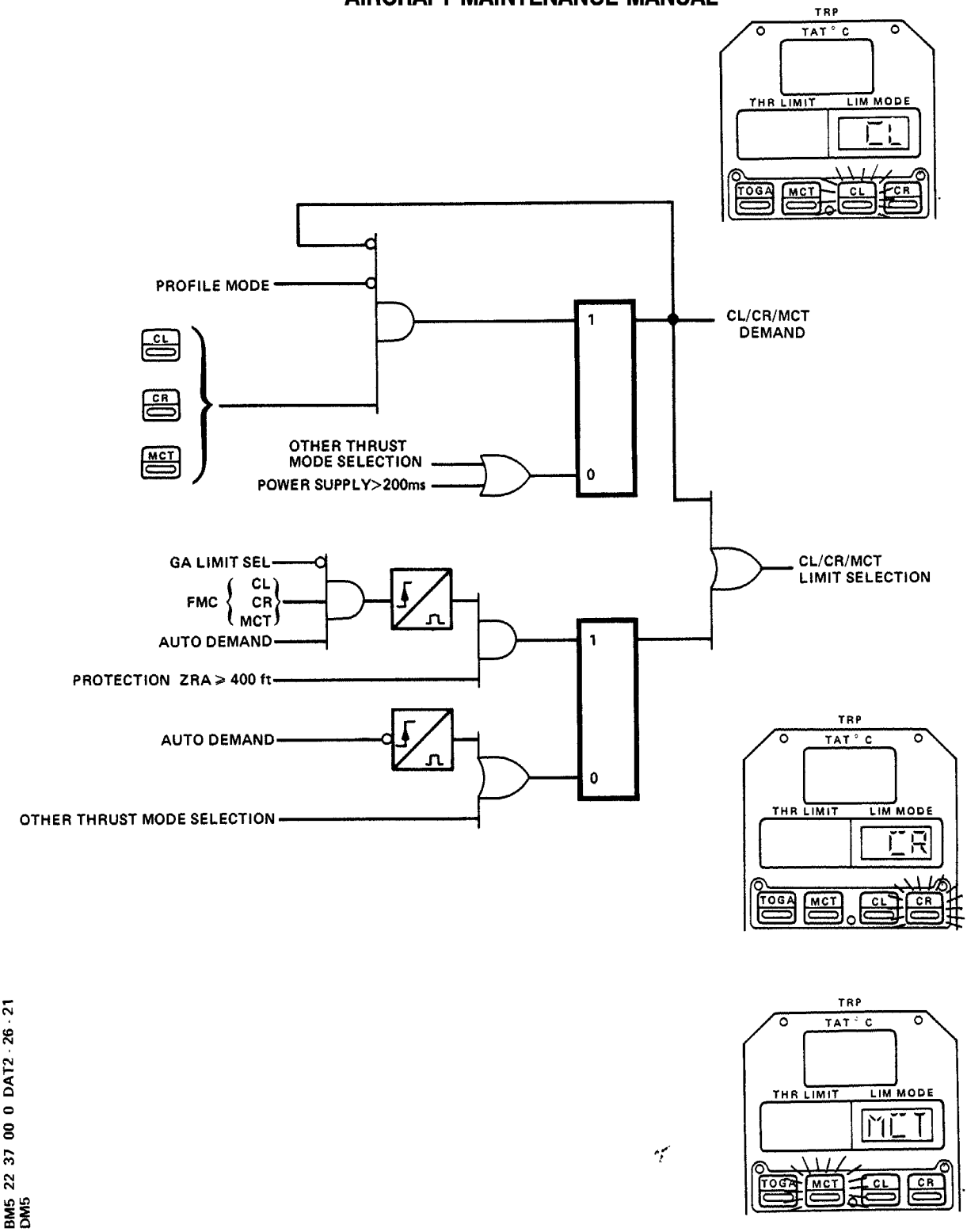
(a)GA THRUST LIMIT selection
1 Manual selection : in flight by pressing TO/GA pushbutton switch.
2 Automatic selection : when slats are extended if at least one AP or one FD is engaged and if the PROFILE mode is not engaged.
3 Display : TO/GA pushbutton switch light comes on and GA appears in LIM MODE display.

(b)TO THRUST selection (selection of maximum take-off thrust). (Ref. Fig. 019)
1 Manual selection : on the ground by pressing TO/GA pushbutton switch
2 Automatic selection : upon aircraft energization
3 Display : TO/GA pushbutton switch light comes on and TO appears in LIM MODE display

(c)FLX TO THRUST selection
Selection of a derated take-off thrust in the event of reduced take-off weight or long runway enables to increase engine component life and achieve noise abatement take-off procedures. Derating is selected by means of the fictitious temperature selector knob.
When the fictitious temperature is greater than the TAT a flexible take-off target value computed by the TCC appears in the TRP TARGET display.
1 Manual selection only : by means of the FLX TO pushbutton switch.
2 Display : FLX TO pushbutton switch light comes on and TO appears in LIM MODE display.

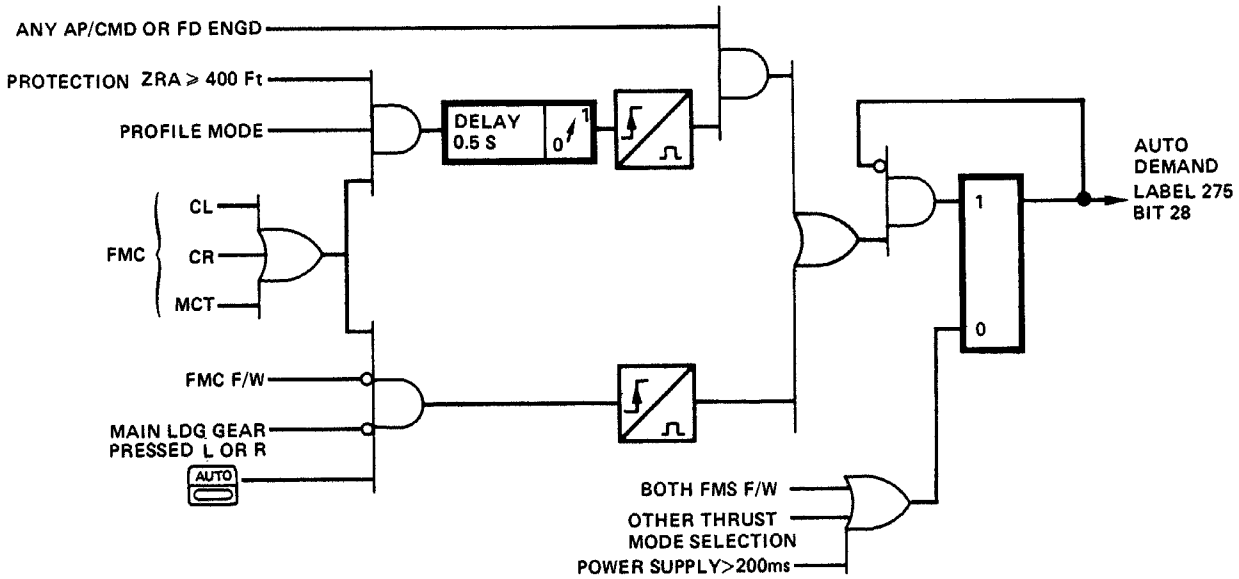
(d)Climb/Cruise/MCT thrust selections (Ref. Fig. 016)
1 Manual selection : by pressing corresponding pushbutton switch.
2 Automatic selection by the FMC CL/CR/MCT orders if AUTO mode is selected and altitude > 400 ft.
3 Display :
- CL/CR/MCT appears in LIM MODE display
- in manual selection corresponding CL/CR/MCT pushbutton switch light comes on.
- in automatic selection (FMC orders) only AUTO pushbutton switch light is on.

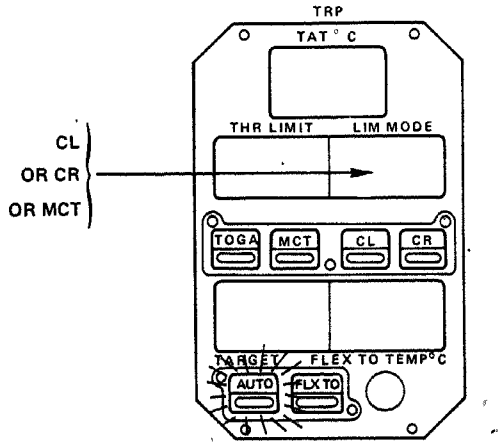
(e)Auto thrust selection (CL/CR/MCT thrust selection automatic management) (Ref. Fig. 017).
This function allows FMC automatic selection of CL/CR/MCT thrust modes.
1 Manual selection by pressing the AUTO pushbutton switch above 400 ft. if the FMC sends a CL/CR/MCT order enables FMC-TCC coupling without selecting FCC PROFILE mode and management of thrust mode selection by the FMC.
2 Automatic selection
The AUTO mode is selected automatically as soon as the last one of the following conditions is present : PROFILE mode engagement, altitude > 400ft and presence of an FMC command (CL/CR/MCT).
3 Display
The AUTO pushbutton switch light comes on.
The relevant thrust mode automatically engaged by the FMC appears in the LIM MODE display.



BM5 22 37 00 0 DAT2 26 21
DM5

Climb/Cruise/Continuous Demand - Selection and Display
Figure 016

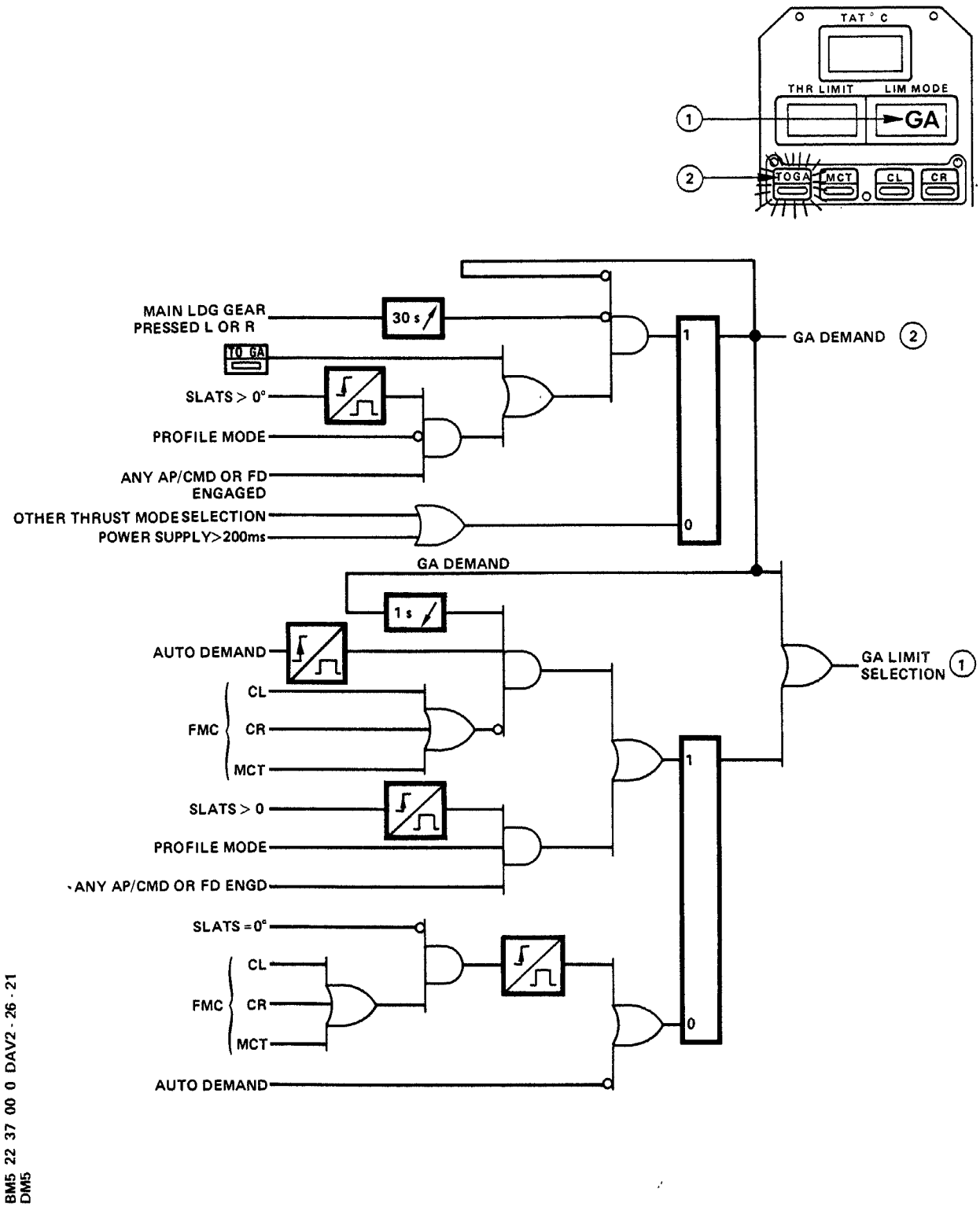




BM5 22 37 00 0 DAU2 - 26 - 21
DM5

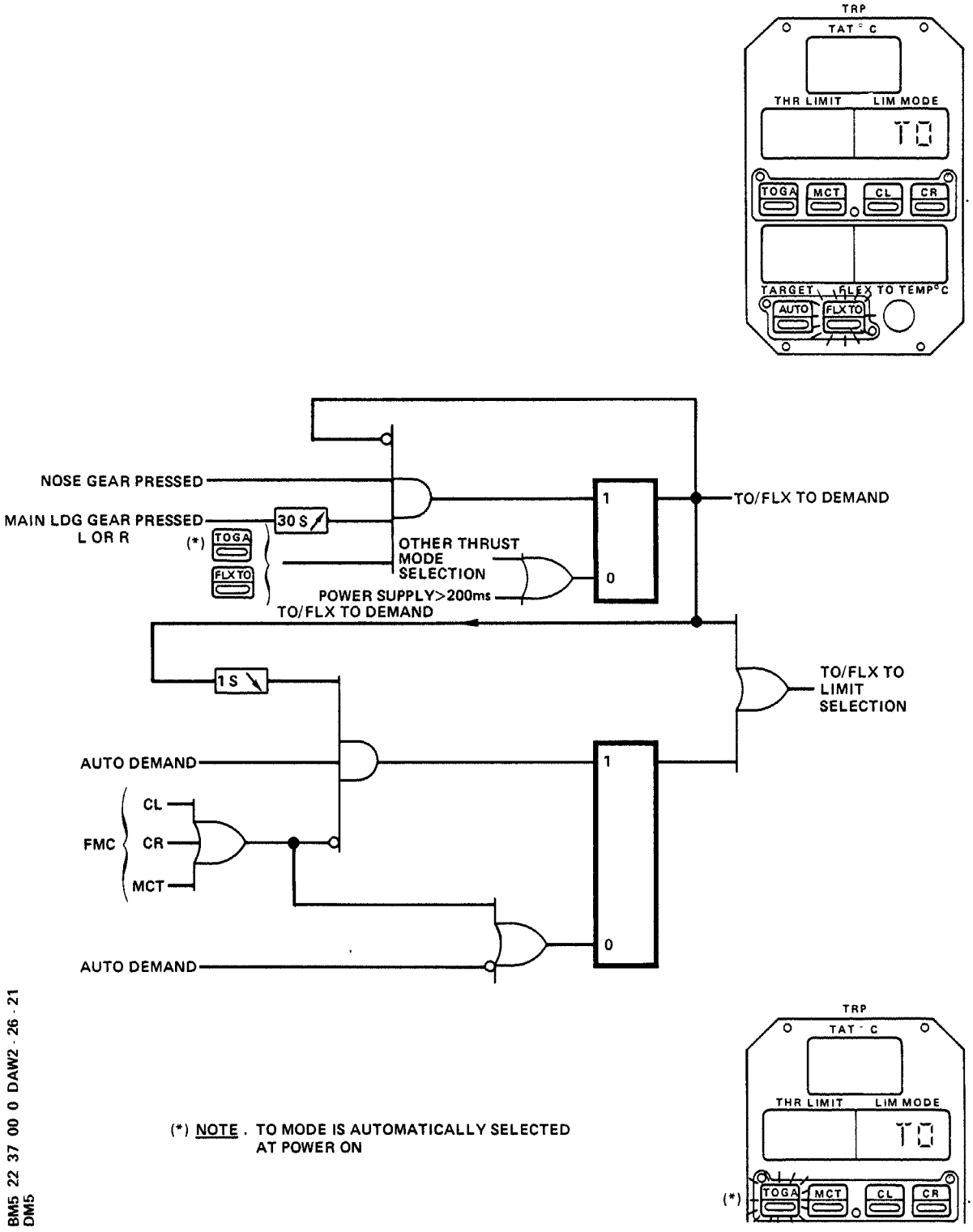
Auto Demand and Display
Figure 017

(Ref. Fig. 018)



BM5 22 37 00 0 DAV2 - 26 - 21
DM5

Go Around Demand - Selection and Display
Figure 018



(*) NOTE . TO MODE IS AUTOMATICALLY SELECTED AT POWER ON

BM5 22 37 00 0 DAW2 26 21
DM5

Take-off/FLX Take-off Demand Selection and Display
Figure 019

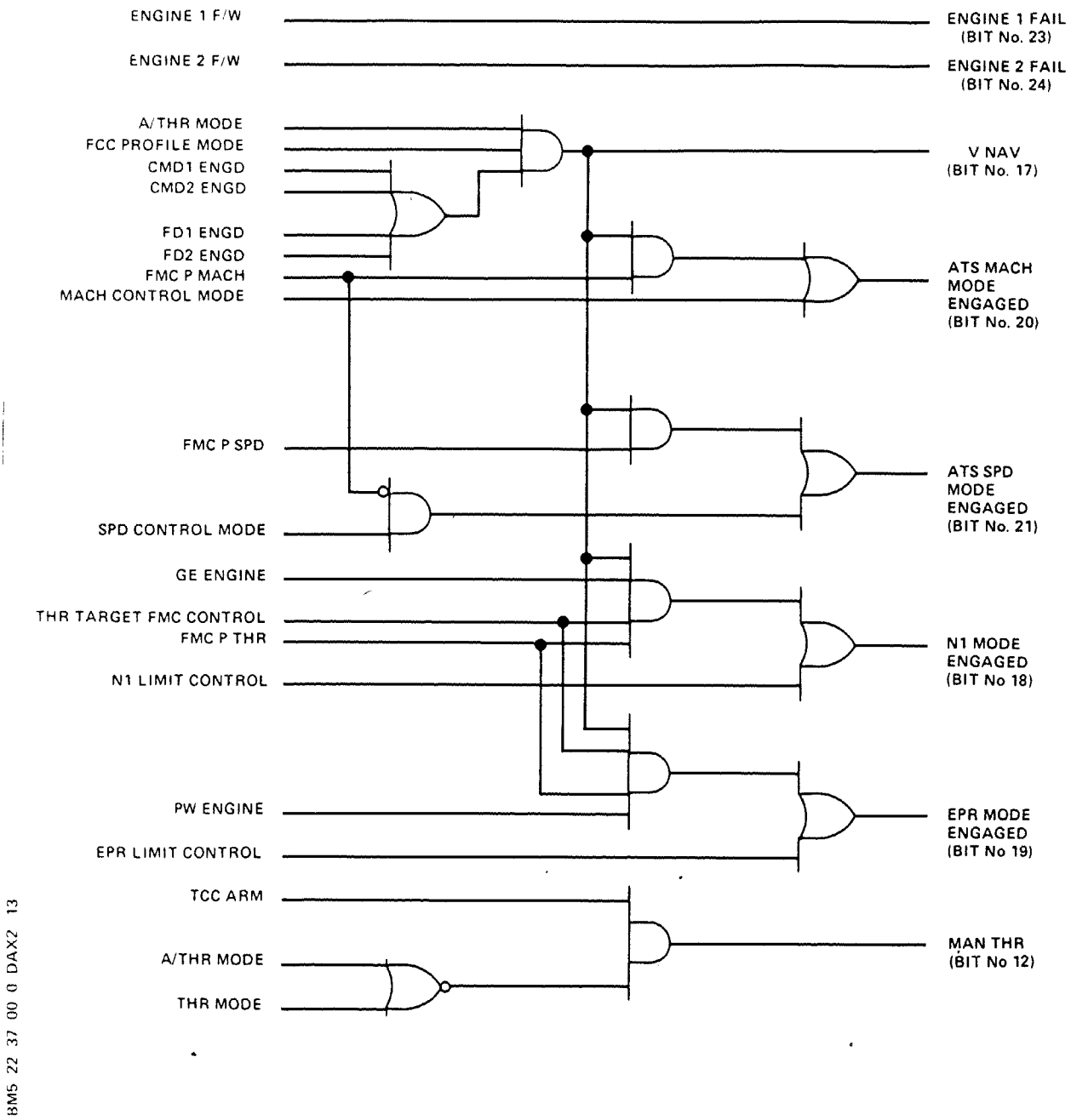
(14)Definition of label 274 discrete bits (Ref. Fig. 020)
The label defines the bit logics used in the processing of display conditions on the EFIS displays. The bit combinations are performed in the EFIS.
- Display (Ref. 22-35).

(15)Bus selection logic
(a)Flight control computer
FCC1 or FCC2 bus selection is made according to the commands delivered by the FCU.
(b)Flight management computer
FMC bus selection is given by the FCCs when PROFILE mode is engaged or in the case of limit mode management by detections of FMC F/W. FMC1 has priority, FMC2 is in standby.
(c)Flight augmentation computer
Selection of FAC1 or FAC2 data bus is made depending on the PITCH TRIM and YAW DAMPER lever engagement.
FAC1 has priority over FAC2.
In FAC1 PITCH TRIM engagement has priority over YAW DAMPER engagement.
When the slats are extended, selection of the FACs depends on the ADC selection made in the TCC.

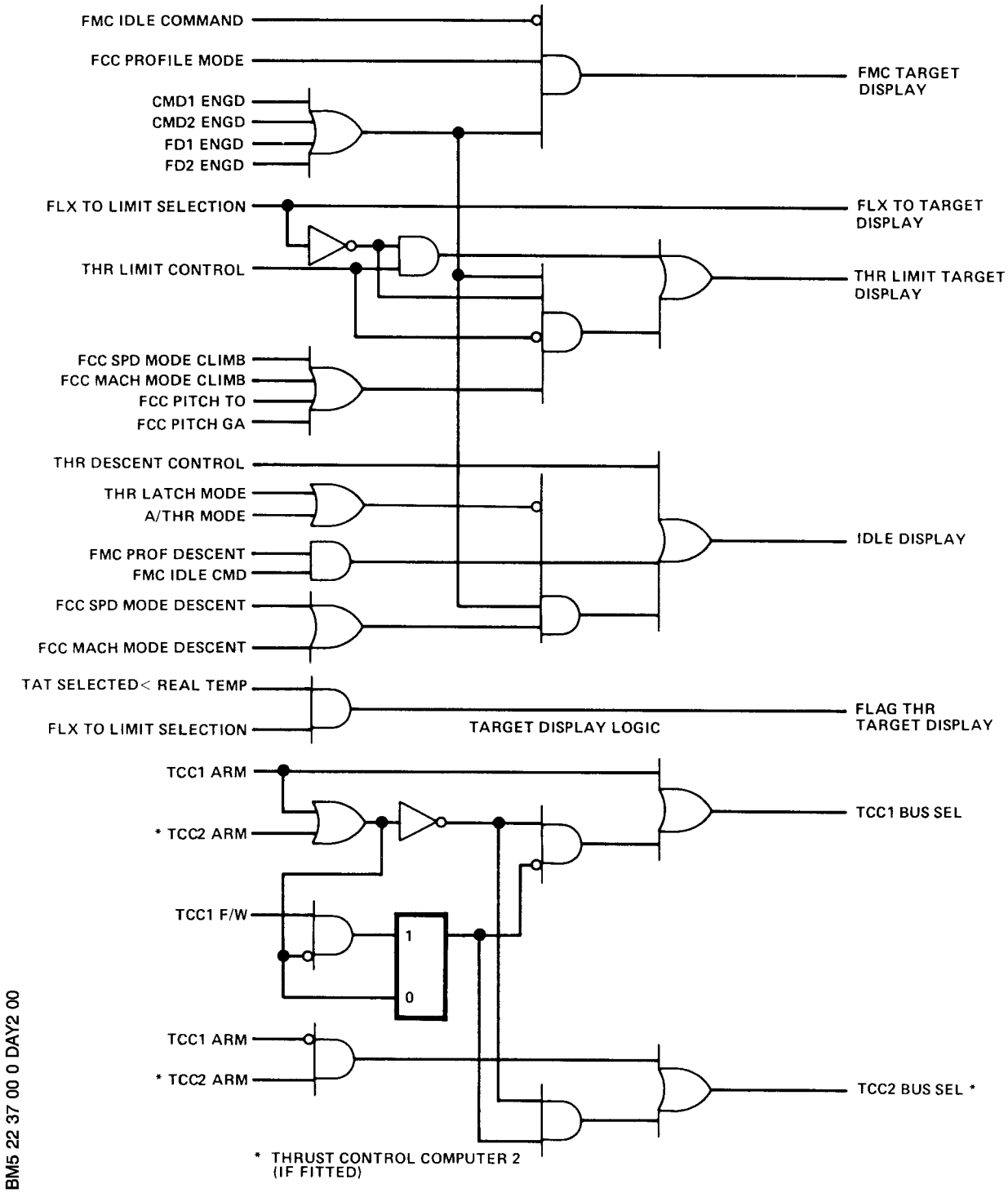
(16)Display of dashes on the TRP
(a)TAT display
Dashes appear in the TAT display if the TAT parameter on ADC1 and 2 is faulty or if the TRP ARINC receiver no longer operates.
(b)THR LIMIT display
Dashes appear in the event of EPR limit computation failure or if the TRP is detected faulty.

(17)TARGET display logic
The TARGET display can show :
- the TARGET value given by the FMC
- the EPR limit value if the THR LIMIT mode is active
- at takeoff when the FLX TO mode is selected, the EPR limit value computed in relation to the selected flexible takeoff temperature
- I-L-when a throttle control lever retard is requested by the FCC (LEVEL CHANGE in DESCENT) or, when in PROFILE mode, the FMC generates an IDLE signal or when the A/THR or THR LATCH modes cannot be engaged and that a throttle control lever retard is requested by the FCC (see Fig. Para 2.B. (5)).

(18)TCC bus selection in the TRP
In single TCC configuration, TCC1 bus is selected.
If the TCC is faulty, the TRP directly reads FMC1 and ADC1 buses.
(Ref. Fig. 021)



TCC Word ARINC Definition (Label 274)
Figure 020



TCC/TRP Bus Validity/Target Display Logic
Figure 021

(19)Logic associated with EECs

(a)Engine trim condition

The **ENG TRIM** failure detection causes **TCC** disarming (**ATS** lever returns to **OFF** position). This failure detection is active only on condition that the engine is running (not failed). The TCC detects EEC failure by monitoring the status matrices of :

- either the **EPR CMD** of the 2 channels
- or the **EPR ACT** of the 2 channels in **PROFILE**
- or the discrete words 270, 271 and 274 of the 2 channels.

(Ref. Fig. 022)

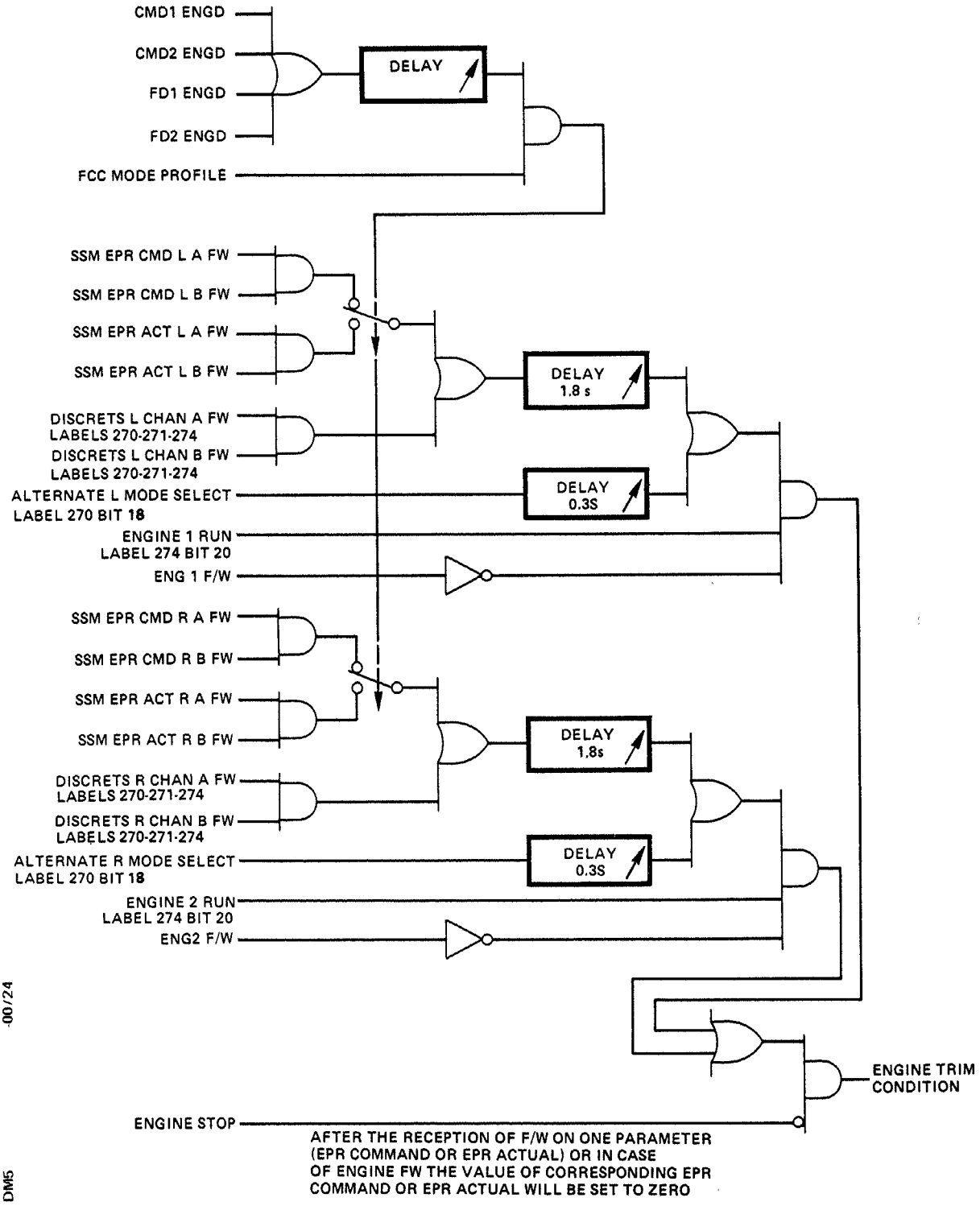
(b)Selection of parameters given by engine trim functions

Channel **A** has priority and upon energization, the TCC takes its information from channels **A** of both engines.

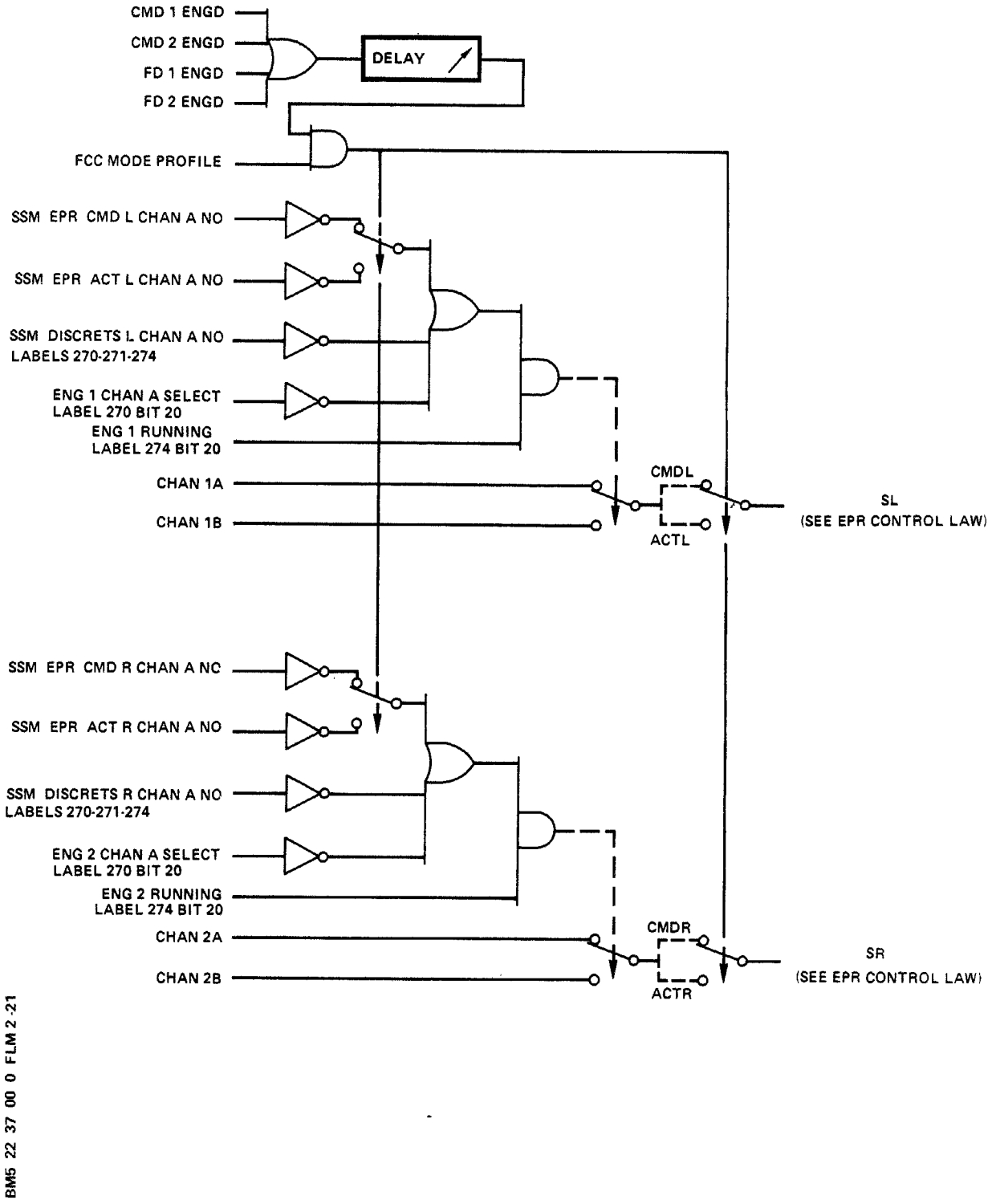
Switching from channel **A** to channel **B** is automatic as soon as a discrepancy is detected :

- either on the labels **EPR CMD** or **EPR ACT** (in **PROFILE**)
- or on the discretes 270, 271 and 274 (bad **SSM**)
- or if the **EEC** has detected an internal failure and requests selection of channel **B** from the TCC (label 270, bit 20 = 0). These switching controls are active as soon as the engine is running (label 274, bit 20 = 1).

(Ref. Fig. 023)


CMD1 ENGD
CMD2 ENGD
FD1 ENGD
FD2 ENGD
FCC MODE PROFILE
DELAY
SSM EPR CMD L A FW
SSM EPR CMD L B FW
SSM EPR ACT L A FW
SSM EPR ACT L B FW
DISCRETS L CHAN A FW LABELS 270-271-274
DISCRETS L CHAN B FW LABELS 270-271-274
ALTERNATE L MODE SELECT LABEL 270 BIT 18
ENGINE 1 RUN LABEL 274 BIT 20
ENG 1 F/W
DELAY 1.8 s
DELAY 0.3S
SSM EPR CMD R A FW
SSM EPR CMD R B FW
SSM EPR ACT R A FW
SSM EPR ACT R B FW
DISCRETS R CHAN A FW LABELS 270-271-274
DISCRETS R CHAN B FW LABELS 270-271-274
ALTERNATE R MODE SELECT LABEL 270 BIT 18
ENGINE 2 RUN LABEL 274 BIT 20
ENG2 F/W
DELAY 1,8s
DELAY 0.3S
ENGINE TRIM CONDITION
ENGINE STOP
AFTER THE RECEPTION OF F/W ON ONE PARAMETER (EPR COMMAND OR EPR ACTUAL) OR IN CASE OF ENGINE FW THE VALUE OF CORRESPONDING EPR COMMAND OR EPR ACTUAL WILL BE SET TO ZERO
BM5 22 37 00 0 FJT2 -00/29 -00/24
DM5

Engine Trim Condition
Figure 022

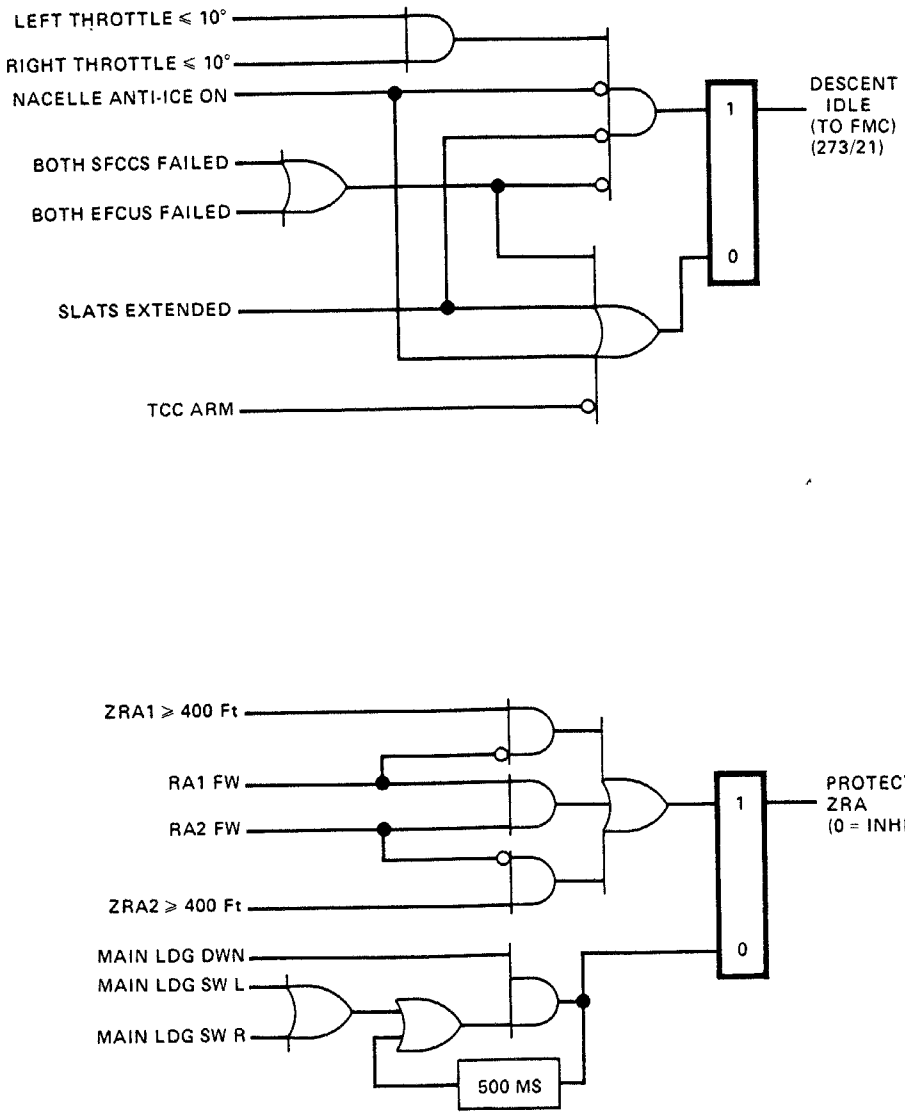


Engine Trim Functions - Selection of Parameters
Figure 023

(20)Descent idle logic to FMC (Ref. Fig. 024)

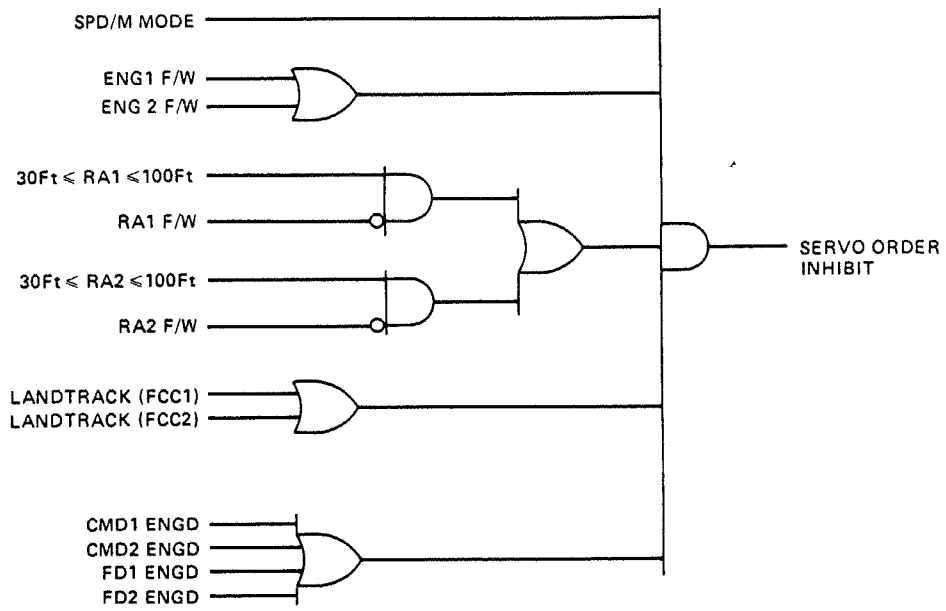
(21)Radio altimeter protection < 400 ft. in PROFILE (***//FNM2//***).
This function is used in control laws and logics to protect against acquisition of the THR TARGET signal from the FMC during takeoff and to 400 ft.
The signal from the FMC is taken into account as soon as the RA altitude is greater than 400 ft. and the landing gear is retracted.

(22)Actuator inhibit in approach
This logic freezes the position of the throttle control levers between 100 ft. and 30 ft. in the event of engine failure in approach, in SPD mode (LAND TRACK).
This prevents the TCC from reacting against thrust reduction due to engine stop and from disturbing automatic approach. Below 30 ft. throttle control lever retraction is active for the flare phase.
(Ref. Fig. 025)



BM5 22 37 00 0 FNM2 21

Descent Idle to FMC ZRA < 400 ft. Protection in FMC
Figure 024



BM5 22 37 00 0 FPM2-21

Actuator Inhibit in Approach
Figure 025

R

1. General

A. Introduction

The Automatic Flight System (AFS) flight line maintenance is based on the use of a Fault Isolation and Detection System (FIDS) integrated into the FACS, FCCS and TCCS.
This system is controlled from the Maintenance Test Panel (MTP) located on the rear panel in the flight compartment.

B. Fault Isolation and Detection System Functions

This system fulfills the following functions :

- Fault isolation
  Data are accessible in flight and on the ground by pressing the DISPLAY pushbutton switch on the Maintenance Test Panel (MTP). It is to be noted that this pushbutton switch is the only one whose function is active in flight.
- Test after replacement of a unit
  This test is performed on the ground by ground maintenance personnel in order to check correct operation of a system after replacement of a unit. This test called AFS TEST is inhibited in flight.
- LAND mode test
  This test is performed on the ground only (inhibited in flight), to check availability of CAT 3 capability.
  If aircraft is operated in actual CAT 3, this test is recommended after removal/installation of the following components :
  FCC, FAC, TCC, FCU (Ref. SIL 22-020).
- Complex fault isolation
  This function is activated on the ground in the event of complex failure detection.
  It is inhibited in flight.
  The MTP can never interfere in the system application program.
  It can only read the information it receives (see "Adopted Safeguards" paragraph).

C. Failure Indication

If a failure was recorded during the last flight, on MTP face, FAILURE
R RECORD magnetic indicator shows clover (black and white). It turns black
R when wheel speed is more than 70KTS.
Alternate solution (MTP with FAILURE RECORD LED)
If a failure was recorded during the last flight, on MTP face, FAILURE
R RECORD LED is on. It goes off when wheel speed is more than 70KTS.

D. AFS Trouble Shooting Procedure

AFS trouble shooting is based on three main steps ; two are mandatory and one conditional.

(1)Principle of the AFS Fault Isolation and Detection System (FIDS)

- Whenever a basic monitoring condition becomes invalid the associated

AFS lever trips,
- The FIDS then takes a snapshot of all parameters and starts its analysis to identify the invalid signal.
  From the invalid signal, and using its own cross-reference table, the FIDS designates and memorizes in the MTP, the system or unit generating this signal.
- In-service experience has shown that the FIDS of the AFS was able to correctly identify nearly all systems when associated levers have tripped (invalid signal generator).
  Upon action on MTP DISPLAY pushbutton switch, the maintenance team is informed of the system or unit involved in the AFS lever disarming.
- In some cases and based on experience or the surrounding context, tripping is considered by the crew as normal and therefore not reported in the log book.
- However the FIDS does not ignore it and a message will be memorized by the MTP when no action is in fact required from the maintenance team.
  Consequently, in order to avoid any confusion :
  The actions on MTP for trouble shooting purpose have to be done only upon pilot entry in the log book.
  Knowing the system involved in an AFS lever tripping is not the end but the starting point for the trouble shooting procedure.
- First, it is necessary to identify a unique LRU when the MTP message points to one system comprising more than one LRU. This is the case when only the central computer of the system is connected to the AFS. It is also the case when the FIDS has no possibility to select between two or three LRUs and therefore displays these LRUs in a probability order.
  Second, and from previous explanations it is well understood that system identification does not necessarily mean component failure. The reason for the invalid signal is still to be found.
- Invalid signal can be the result of :
  . a hardware failure (component failure), the only case where a maintenance action is required (LRU replacement).
  . any other cause such a computation disturbance, transient flag or missing data (e.g. flight plan discontinuity) where no LRU replacement is necessary.
  For these reasons, continuation of the trouble shooting should be done using the self-test of any system displayed by the MTP.

(2)Self-test of the AFS and peripherical systems
- Most of the systems or units have their own built-in self-test,
- For the AFS they consist in power-up and AFS tests,
- The purpose of the self-tests is to check the integrity of the hardware within each computer and the integrity of most of the LRUs without BITE but in direct connection with the computer (sensors, actuators, switches...),
- These tests do not rely on the control laws and logics of the avionics program but are solely turned towards the components integrity.
  For the AFS when power up and AFS test are successful it is considered that all AFS safety devices are operative and that nearly all components within the AFS are healthy.

(3)Conditional testing

- During the trouble shooting procedure, some LRUs linked to the AFS and without internal BITE, can be exhibited by the MTP during the display activation but not entirely tested during the AFS test.
  These LRUs are the actuators and dynamometric rods for which a failure can occur only during movement.
  They also include switches and relays for which a resting contact may remain frozen during activation.
- Consequently, after completion of a successful AFS test and for the trouble shooting of the few remaining components within the AFS computers and above mentioned LRUs it is recommended to arm the levers of any AFS computer reported by the MTP during display activation.
- In addition, whenever an actuator is reported during display but AFS test successfully passed, it is recommended to activate the reported actuator using a command order from its associated computer.
- The same procedure applies to the dynamometric rod with a manual command.
- In all above-mentioned cases the associated AFS levers must stay armed. If not, the ground scan procedure should be used to identify the LRU and confirm the original message exhibited during display activation.

(4)Summary

- MTP display, system self-test and conditional testing are the three tasks to do upon pilot report for AFS disarming (Ref. 22-00-01, P. Block 400, Figure 101).
  Unit removal should be done only if a self-test or a conditional test fails.

E. Additional Recommendations

(1)Complementary information

- The self-test can be affected by disturbances while running. Many activities are conducted during an aircraft stop like switching in electrical networks, supply through electrically noisy ground power units, other trouble shooting actions etc.
  Even though the probability for disturbance is low, it is recommended, when the level of spares is limited, to repeat a failed test for confirmation.
- In some instances, the complete procedure leads anyhow to a list of components for removal without possibility to identify a single unit. In such a case, the units listed by the MTP are classified in a probability order and preceded by a letter ("A" is most probable). Only one unit should be removed but the log book must be checked to see whether the most probable unit was not already replaced for the same reason on previous flights.
- In some remote cases, a real but transient component failure exists (only momentarily) and cannot be identified during ground tests. At that time, the self-test and conditional test are successful. The only available information is the message given by the MTP display activation.
  Therefore when the self-test and conditional test are successful, it is necessary to check the log book for any identical case during the previous flights.
  Obviously this is based on the condition that, beside pilot entry, the

result of MTP display, self-test and conditional test plus any other action are reported in the log book.

2. Component Location

**ON A/C ALL

(Ref. Fig. 001)

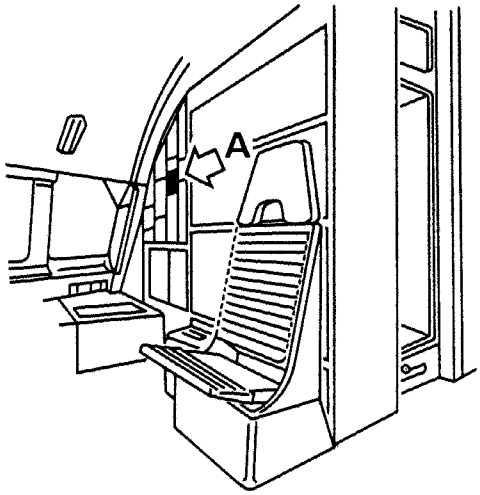
**ON A/C ALL

(Ref. Fig. 002)

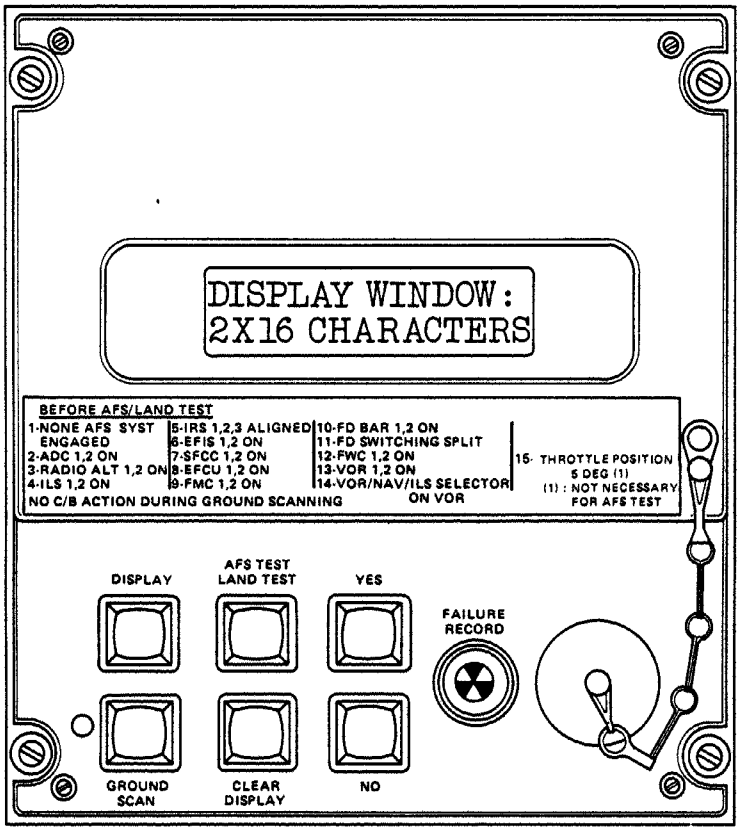
**ON A/C ALL

FIN	FUNCTIONAL DESIGNATION	PANEL	ZONE	ACCESS DOOR	ATA REF.
3CU	MTP	120VU	212		22-42-21

**ON A/C ALL

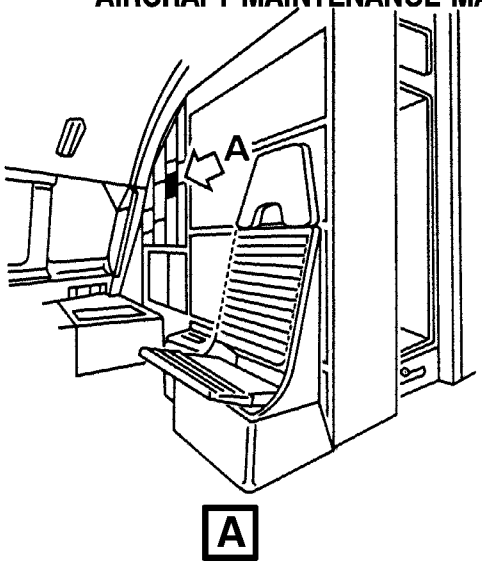


A MAINTENANCE TEST PANEL (3CU)



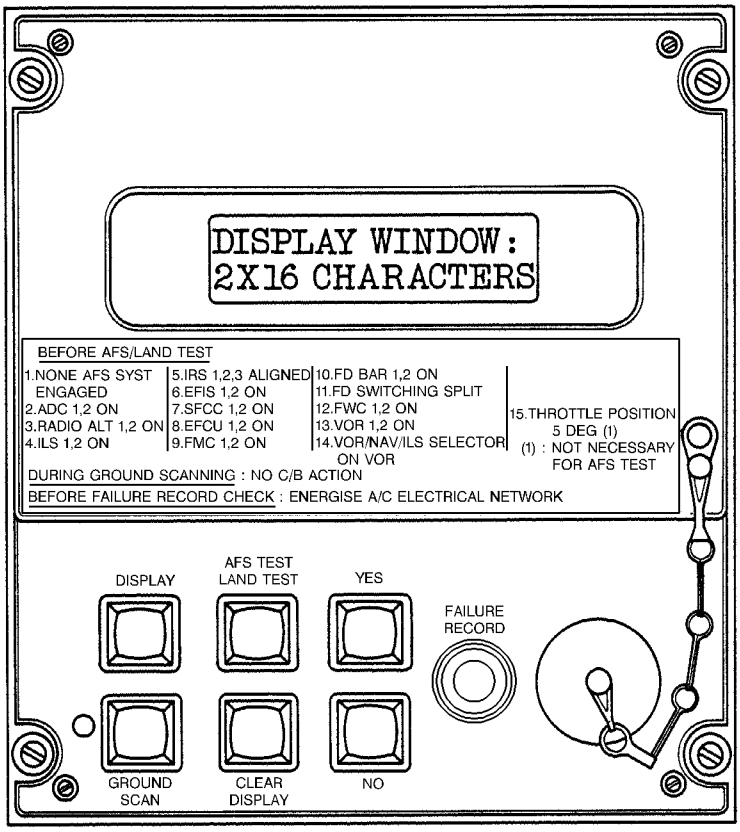
BM5 22 40 00 0 TGP2 23 18
DM5

Maintenance Test Panel - Location
Figure 001



A

MAINTENANCE TEST PANEL (3CU)



BM5 22 40 00 0 ACM0 00

Maintenance Test Panel - Location
(MTP with FAILURE RECORD LED)
Figure 002

R

3. Power Supply
The MTP is supplied with 28VDC from Normal busbar 1PP (sub-busbar 111PP) via circuit breaker 301CU.

A. Power Supply Automatic Restoration and Cut Off
The MTP power supply is automatically cut off when the last system is de-energized.
The MTP power supply is restored :
- automatically by the CP POWER ON signal delivered by FAC1.
- upon initiation of the maintenance panel lamp test,
- by pressing one of the following pushbutton switches : DISPLAY, AFS TEST/LAND TEST or GROUND SCAN.

In the two latter cases the MTP causes energization of the AFS by the TEST POWER ON signals.
NOTE : The MTP is permanently supplied if FAC1 is not in the rack (open circuit on CP POWER ON).

4. Interface with Peripherals
Connections between the MTP and peripherals are mainly accomplished using ARINC 429 data bus lines.
However, they are composed of a certain number of discretes related to the computer and MTP energization, gear information signals and wheelspeed higher than 70 kts information.

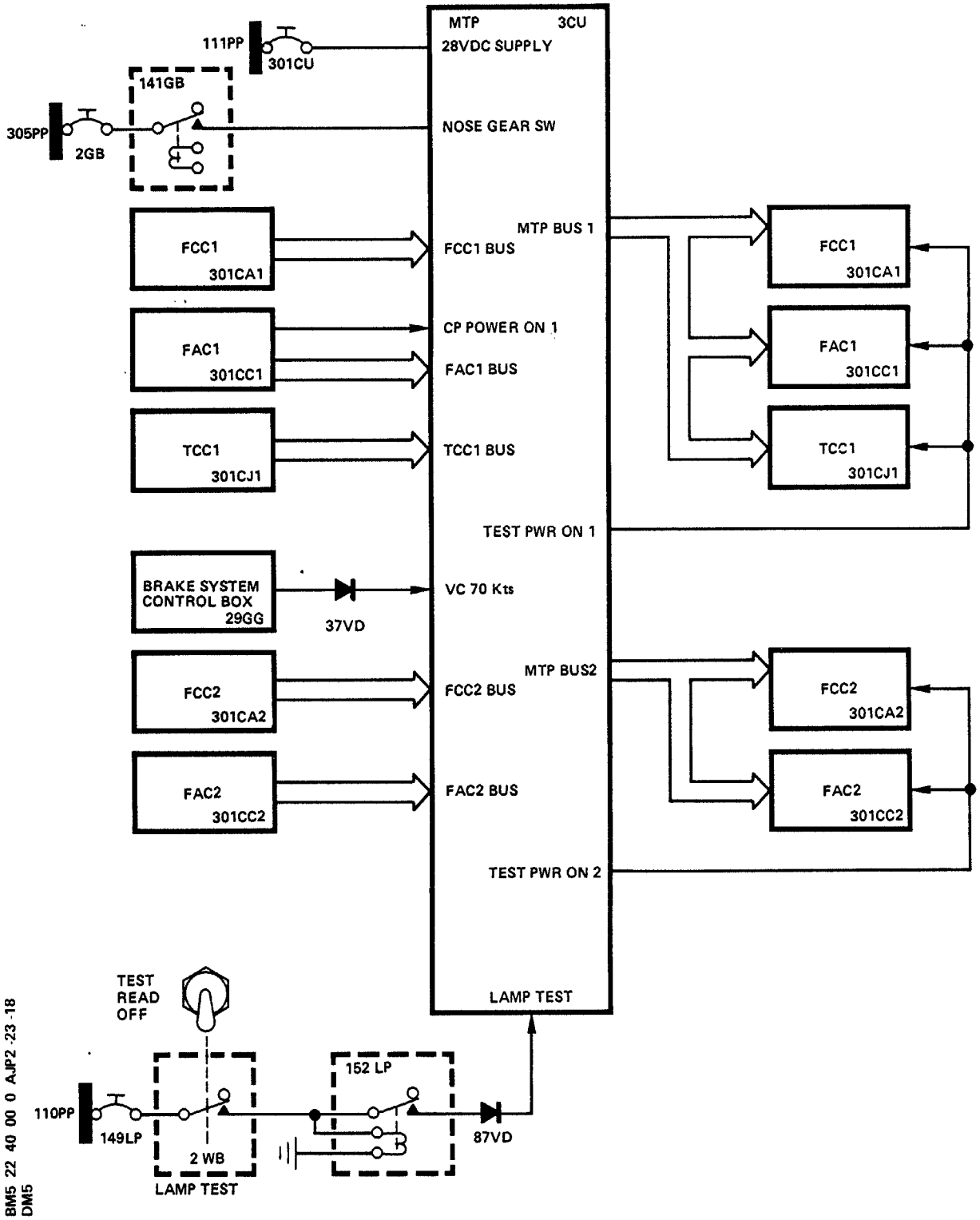
R **ON A/C 101-200,

(Ref. Fig. 003)

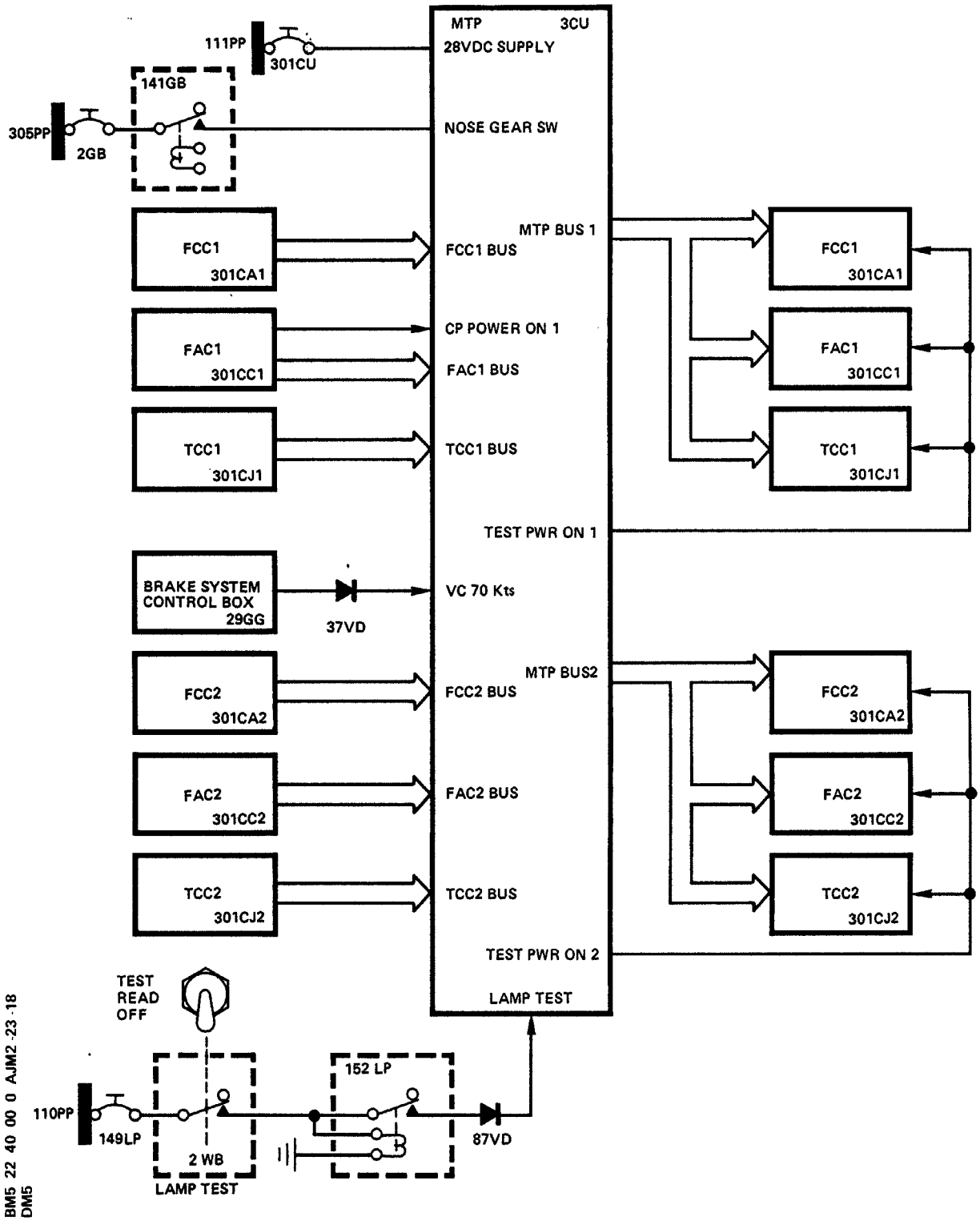
R **ON A/C 001-100,

(Ref. Fig. 004)

R **ON A/C ALL



MTP/AFS Peripheral Interfaces and Interconnections Schematic
Figure 003



MTP/AFS Peripheral Interfaces and Interconnections Schematic
Figure 004

A. ARINC Input Signals

Information Utilized - Characteristics

The MTP utilizes the discrete words and the maintenance words delivered by the computers.

COMPUTER	PARAMETER NAME	LABEL	TRANSMIT INTERVAL (ms)
TCC	TCC Discrete 1	271	125
	TCC Discrete 2	272	125
	TCC Discrete 3	273	125
	TCC Discrete 4	274	125
	TCC Discrete 5	275	125
	TCC Discrete 6	276	125
	TCC Discrete 7	277	125
	Maintenance Data	350	asynchronous
	Maintenance Data	351	asynchronous
FCC	FCC Discrete 1	002	180
	FCC Discrete 2	003	180
	FCC Discrete 3	001	180
	FCC Discrete 4	004	180
	FCC Discrete 5	005	
	FCC Discrete 6	065	
	Maintenance Data	350	asynchronous
	Maintenance Data	351	asynchronous
FAC	AFS Discretes 1	243	30
	AFS Discretes 2	244	30
FAC	AFS Discretes 1	245	30
	AFS Discretes 2	246	30
	Maintenance Data	350	asynchronous
	Maintenance Data	351	asynchronous

DIALOGUE MTP COMPUTER
FCC ---> MTP MAINTENANCE DATA 1 (LABEL 350)

BIT NUMBER	FUNCTION	CODING		
1 to 8	LABEL	350		
9	TYPE OF INFORMATION	0 : COMPLEMENTARY INFORMATION 1 : EVENT REPORT		
10 to 13	TYPE OF EVENT OR CODE FOR COMPLEMENTARY INFORMATION	EVENT REPORT 0 : AP DISCONNECTION 1 : FD LOSS 2 : LAND3 LOSS 3 : LAND2 LOSS 4 : APP2 LOSS 5 : CAPACITY NOT AT MAXIMUM LEVEL 14 : COMPLEMENTARY TEST RESULT 15 : AFS TEST RESULT	COMPLEMENT INFORMATION CODE 1 : END OF THE CONTEXTS (2) TABLE 2 : CLEAR EXECUTED 3 : CLEAR NOT EXECUTED 4 : END OF CONTEXT (2) 0 : OTHERWISE (1)	
14 15	RESULT OF FAILURE LOCALIZATION ANALYSIS	0 : DISCONNECTION BY LEVER 1 : DISCONNNECT BY INST DISC DISC 2 : NORMAL DISCONNECTION 3 : ABNORMAL DISCONNECTION	AFS TESTS COMPL. TESTS BIT 14 0 : ACCEPTED 1 : REFUSED	RETURNED INFORMATION BIT 14 : 0 : NOT CONNECTED 1 : CONNECTED
16 17	NUMBER OF LRU	IN MAINTENANCE DATA 2 (LABEL 351)		BIT 22 : 0 : NOT RECEIVED 1 : RECEIVED
18	COMPLEMENTARY TESTS ON GROUND REQUESTED	0 : NO 1 : YES		
19 to 29	SPARE			
30 31	STATUS MATRIX			
32	PARITY			

(1) : in case of response to a need of information to improve MTP diagnostics these bits are filled with zero
(2) : to print and decode failure information recorded inside the computers :
complementary information code = 1 : end of printing
= 4 : end of a record

FAC ---> MTP MAINTENANCE DATA 1 (LABEL 350)

BIT NUMBER	FUNCTION	CODING		
1 to 8	LABEL	350		
9	TYPE OF INFORMATION	0 : COMPLEMENTARY INFORMATION 1 : EVENT REPORT		
10 to 13	TYPE OF EVENT OR CODE FOR COMPLEMENTARY INFORMATION	EVENT REPORT 0 : YAW DAMPER DISCONNECTION 1 : TRIM DISCONNECTION 2 : ALPHA PROBE FAILURE 3 : 4 : 5 : 14 : COMPLEMENTARY TEST RESULT 15 : AFS TEST RESULT	COMPLEMENT INFORMATION CODE 1 : END OF THE CONTEXTS (2) TABLE 2 : CLEAR EXECUTED 3 : CLEAR NOT EXECUTED 4 : END OF CONTEXT (2) 0 : OTHERWISE (1)	
14 15	RESULT OF FAILURE LOCALIZATION ANALYSIS	0 : DISCONNECTION BY LEVER 1 : DISCONNECT BY INST DISC DISC 2 : NORMAL DISCONNECTION 3 : ABNORMAL DISCONNECTION	AFS TESTS COMPL. TESTS BIT 14 0 : ACCEPTED 1 : REFUSED	RETURNED INFORMATION
16 17	NUMBER OF LRU	IN MAINTENANCE DATA 2 (LABEL 351)		
18	COMPLEMENTARY TESTS ON GROUND REQUESTED	0 : NO 1 : YES		
19 to 29	SPARE			
30 31	STATUS MATRIX			
32	PARITY			

BIT NUMBER	FUNCTION	CODING

(1) : in case of response to a need of information to improve MTP diagnosis
these bits are filled with zero
(2) : to print and decode failure information recorded inside the computers :
complementary information code = 1 : end of printing
= 4 : end of a record

TCC ---> MTP MAINTENANCE DATA 1 (LABEL 350)

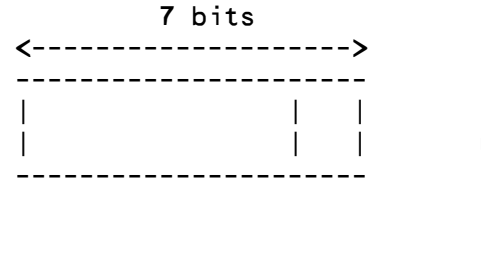
BIT NUMBER	FUNCTION	CODING		
1 to 8	LABEL	350		
9	TYPE OF INFORMATION	0 : COMPLEMENTARY INFORMATION 1 : EVENT REPORT		
10 to 13	TYPE OF EVENT OR CODE FOR COMPLEMENTARY INFORMATION	EVENT REPORT 0 : ATS DISCONNECTION 1 : SPEED MODE LOSS 2 : N1/EPR MODE LOSS 3 : 4 : 5 : 14 : COMPLEMENTARY TEST RESULT 15 : AFS TEST RESULT	COMPLEMENT INFORMATION CODE 1 : END OF THE CONTEXTS (2) TABLE 2 : CLEAR EXECUTED 3 : CLEAR NOT EXECUTED 4 : END OF CONTEXT (2) 0 : OTHERWISE (1)	
14 15	RESULT OF FAILURE LOCALIZATION ANALYSIS	0 : DISCONNECTION BY LEVER 1 : DISCONNECT BY INST DISC 2 : NORMAL DISCONNECTION 3 : ABNORMAL DISCONNECTION	AFS TESTS COMPL. TESTS BIT 14 0 : ACCEPTED 1 : REFUSED	RETURNED INFORMATION
16 17	NUMBER OF LRU	IN MAINTENANCE DATA 2 (LABEL 351)		
18	COMPLEMENTARY TESTS ON GROUND REQUESTED	0 : NO 1 : YES		
19 to 29	SPARE			
30 31	STATUS MATRIX			
32	PARITY			

(1) : in case of response to a need of information to improve MTP diagnosis
these bits are filled with zero
(2) : to print and decode failure information recorded inside the computers :
complementary information code = 1 : end of printing
= 4 : end of a record

COMPUTERS ---> MTP MAINTENANCE DATA 2 (LABEL 351)

BIT NUMBER	FUNCTION	CODING
1 to 8	LABEL	351
9 to 15	LRU1	
16 to 22	LRU2	
23 to 29	LRU3	
30, 31	STATUS MATRIX	
32	PARITY	

LRU CODES



LRU CODE	LRU	LRU CODE	LRU	LRU CODE	LRU
00	NOT USED	36H	EEC SENSORS	60H	LH ATS COUPLING UNIT
02	ADS	38H	PMC SENSORS	62H	ENGINE SENSOR
04	IRS	3AH	NOSE GEAR RELAY	64H	HMC
06	IRS 3	3CH	R MAIN GEAR RELAY	66H	AP DYN ROD NOT CONNECTED
08	ILS	3EH	L MAIN GEAR RELAY	68H	PITCH DYN ROD CAPT/FO
0AH	Radio Altimeter	40H	GROUND SPOILER CONTROL BOX	6AH	AT DYN ROD NOT CONNECTED
0CH	VOR	42H		6CH	AT COUPLING UNIT NOT CON.
0EH	EFIS	44H	AP DISCONNECT PB SW	6EH	115VAC POWER SPLY
10H	AOA SENSOR	46H	AT INST DISCONNECT SW	70H	AP PITCH ACTUATOR
12H	AOA SENSOR 3	48H	AT ACTUATOR NOT CONNECTED	72H	AP ROLL ACTUATOR
14H	PMC TRANSIENT FAULT	4AH	FMC N.C.D.	74H	AP YAW ACTUATOR
16H	FMC	4CH	R THROTTLE POSITION DETECTOR	76H	AT ACTUATOR
18H	EEC	4EH	L THROTTLE POSITION DETECTOR	78H	PITCH TRIM ACTUATOR
1AH	EFCU	50H	SPLIT BARS SWITCH	7AH	YAW DAMPER ACTUATOR
1CH	FWC	52H		7CH	28VDC POWER SPLY
1EH	FCC	54H		7EH	26VAC POWER SPLY
20H	TCC	56H	PITCH TRIM CONTROL SWITCH		
22H	FAC	58H	TO/GA LEVERS		
24H	FCU	5AH	H STABILIZER SWITCH		
26H	TRP	5CH	HYDRAULIC SYSTEM		
28H	PMC	5EH	RH ATS COUPLING UNIT		
2AH	MTP				
2CH	FAC/ATS Engage Unit				
2EH	VARIABLE LEVER				
30H	FD BAR REMOVAL SWITCH				
32H	SFCC				
34H	28 V GEAR				

B. ARINC Output Signals

The MTP transmits information through two 12 KB/s data buses.
Data bus No.1 is associated with FAC1, FCC1 and TCC1.
Data bus No.2 is associated with FAC2, FCC2 and TCC2 (if fitted).

Transmitted Information - Characteristics

BIT No.	FUNCTION	ENCODING			ASSIGNMENT	COMMENTS
1-8	Label 350, 351, 352					
9	Command	0	0	0	Read	
10		0	0	1	Write	
11		0	1	0	Spare	
		0	1	1	LAND Test	
		1	0	0	Ground Scan.	
		1	0	1	Reset	
		1	1	0	Spare	
		1	1	1	Spare	
12	- Encoding of involved data for reading and writing - Complementary test encoding on next page - LAND test encoding - Bit Nos. 12, 13, 14				Clear Coding Bit Nos. 12, 13	11 = C and M 01 = C 10 = M Scanning Condition Encoding Bit Nos. 12, 13 00 = Ground Scan Request 01 = End Flight Scan Request 10 = Flight Scan Request
13						
14						
15						
16						
17						
18						
19						
20						
21	Concerned lane for read and write function	0 : Monitor lane of tee computer 1 : Command lane of tee computer				
22	Useful data for writing					
29						
30	SSM ARINC format					
32						

LAND Mode Test Encoding

0 0 0	AFS TEST
0 0 1	LAND TEST
0 1 0	Provision
0 1 1	Provision
1 0 0	Provision
1 0 1	Provision
1 1 0	Provision
1 1 1	LAND test - End of 2nd phase

C. Discrete Signals

	SIGNAL	STATUS		FUNCTION
		IN EFFECT	NOT IN EFFECT	
INPUT	CP POWER ON 1 NOSE GEAR SWITCH	Open 28 V	Ground Open	MTP energization by FAC1 Inhibition of ground functions in flight
INPUT	CP POWER ON 2	Open	Ground	MTP Energization by FAC2
INPUT	LAMP TEST	28 V	Open	Display test
INPUT	V > 50 kts	Ground	Open	Activation of the fault isolation function
OUTPUT	TEST POWER ON 1	28 V	Open	Energization by the MTP
	TEST POWER ON 2	28 V	Open	Energization by the MTP

D. Circuits Integrated in the Different Computers

The circuits associated with maintenance and test performance are built into each FAC, FCC and TCC.

The following functions are fulfilled :

- generation of hardware stimuli,
- logic gates for reading of the comparator status,
- control of the logic switches for test requirements (synchronization, logic inhibition),
- memories for safeguard of the context (Unit Fault Data) at the time of failure,
- batteries for the supply of the safeguard memories.

The percentage of these circuits, per computer, is less than 10% of the computer hardware.

1. Operational Test

A. AFS Test

R (1)Reason for the job.

R Check validity of Alpha Probes by performing AFS test on MTP.

R (2)Equipment and Materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(2)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT MAKE CERTAIN THAT ELECTRICAL CIRCUIT UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- make certain that electronics racks ventilation is correct.
- close overhead panel circuit breakers particularly those associated with AFS and NAVIGATION.
- close L/G PROX DET & RELAYS circuit breakers on panel 133VU (aircraft in ground configuration).
- close SFCC and EFCU circuit breakers on panel 133VU.
- make certain that aircraft is in accordance with instructions on MTP front placard (rear panel).
- align IRS 1, 2 and 3 (NAV selected on MSU).

(b)Test

ACTION	RESULT
1. On maintenance test panel (MTP) - press AFS TEST/LAND TEST pushbutton switch	On MTP, the following messages appear : - SELF TEST ON 1 - SELF TEST ON 2 - NO REFRESH : 300 (counter is reset to 300) and decremented every second). - SELF TEST ON 3 - SELF TEST ON 4 - SELF TEST ON 5 - AFS TEST : OK LAND TEST?
- press NO pushbutton switch	Messages disappear.

NOTE : One of the following failure messages can appear at the end of the first AFS test after the flight :
- ANGLE OF ATTACK SENSOR 3 (AOA 3 failure)

- ADS1 FLAG (AOA 1 failure)
- ADS2 FLAG (AOA 2 failure)

When AFS test is performed a second time, the message above disappears. This does not mean that the failure has disappeared. An AFS test will be performed after one of the following flights and if the failure message is displayed again, the sensor identified by the message must be replaced.

(c)Close-up

1 On overhead panel, on MSUs 1, 2 and 3 place mode selector switch in OFF position.

2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

B. Land Test and Autoland Lights Test

(1)Reason for the job

To check for correct operation of LAND mode and to restore the aircraft in CAT 3 capability.

(2)Equipment and Materials

ITEM	DESIGNATION
(1)	Cart - Hydraulic Ground Power Supply (Aircraft - Type Pump)

R NOTE : As an alternative it is possible to pressurize the hydraulic systems
R using Green system electric pumps and PTUs (Ref. 29-21-00, 29-22-00,
R 29-23-00, P. Block 301).

Referenced Procedures

- 24-41-00, P. Block 301	AC External Power Control
- 29-10-00, P. Block 301	Main Hydraulic Power - Pressurization/ Depressurization
- 29-21-00, P. Block 301	Green Auxiliary Power
- 29-22-00, P. Block 301	Blue Auxiliary Power
- 29-23-00, P. Block 301	Yellow Auxiliary Power (Power Transfer Unit)

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT, MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.
BEFORE PRESSURIZING HYDRAULIC SYSTEMS, MAKE CERTAIN THAT HYDRAULIC SYSTEMS UNDER MAINTENANCE HAVE BEEN ISOLATED.
MAKE CERTAIN THAT THE TRAVEL RANGES OF THE CONTROL SURFACES ARE CLEAR.

(a)Job set-up

Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- Make certain that hydraulic power is available (GREEN, BLUE, YELLOW).
- Close overhead panel circuit breakers, particularly those associated with AFS, NAVIGATION, ECAM and ADS.
- Align IRS 1, 2, 3 (NAV selected on MSU).
- Adjust brightness on CAPT and F/O PFDs by using knob on CAPT and F/O complementary navigation control panels 412 and 413VU.

(b)Test

ACTION	RESULT
1 Perform AFS test as described in para. A but when LAND TEST? is displayed in MTP window, press YES pushbutton switch.	On lateral panel, on MTP, required actions and questions are displayed.
2 By using YES and NO pushbutton switches answer to questions OR perform required actions according to displayed message.	- END OF TEST : LAND ENGAGE OK

NOTE : During "Land Test", when "AUTOLAND LIGHTS FLASH ? YES OR NO" is displayed on MTP, check that the word AUTO and the word LAND are lighted.

(c)Close-up

1 Shut down and disconnect hydraulic ground power cart.

2 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

C. Display Test

(1)Reason for the job

Read failure messages recorded by flight scanning since last takeoff when FAILURE RECORD magnetic indicator shows clover (black and white).

R Alternate solution (MTP with FAILURE RECORD LED)

R Read failure messages recorded by flight scanning since last takeoff when

R FAILURE RECORD LED is on.

(2)Equipment and materials

ITEM	DESIGNATION
Referenced Procedure - 24-41-00, P. Block 301	AC External Power Control

(3)Procedure

WARNING : BEFORE POWER IS SUPPLIED TO THE AIRCRAFT MAKE CERTAIN THAT ELECTRICAL CIRCUITS UPON WHICH WORK IS IN PROGRESS ARE ISOLATED.

(a)Job set-up

- Energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).
- Make certain that electronics racks ventilation is correct.
- on panel 21VU close circuit breaker 301CU (AFS/MTP line 102/H1).

(b)Test

ACTION	RESULT
1. On maintenance test panel (MTP) - make certain that ground scanning	

ACTION	RESULT
function is not active. If ground scanning function is active - press **GROUND SCAN** pushbutton switch	- green **LED** goes off.
- press and release **DISPLAY** pushbutton switch.	After few seconds : - last **FLIGHT LEG N FAIL RECORD** message appears. - observe messages indicating suspect and recorded **LRUs**.
Observe **END** message.	

NOTE : It is possible to read failures recorded during previous flight legs. To this end press **YES** and **NO** pushbutton switches simultaneously and while keeping them pressed, press **DISPLAY** pushbutton switch and release the three of them.

(c)Close-up

1 De-energize the aircraft electrical network (Ref. 24-41-00, P. Block 301).

R

1. Fault Isolation Procedure Initiation

Whenever a failure is detected, the corresponding system transmits an interruption signal to the command and monitoring channels.
This interruption signal is generated in the three following cases :
- disconnection
- engagement refusal
- landing capability downgrading.

This interruption causes a snapshot to be taken into account (primary data), and location of the failure based on the snapshot is initiated.

(Ref. Fig. 001)

2. Fault Isolation Process

The background task fault isolation procedure is constituted of two phases :
- a first phase which is performed independently by both microprocessors and serves to determine the cause of the disconnection.
  This phase is based on the organization of the engage and disengage logic as well as on the snapshot taken when the fault occurs. The monitoring microprocessor sends the result of the analysis to the command microprocessor.
- a second phase which is performed by the command microprocessor only.
  After comparison of results from both microprocessors and if an agreement exists, an overall result is generated in keeping with a decision logic.
  The command channel can eventually ask the monitoring channel for additional information.
  The overall result is sent to the MTP by the command channel.

All the functions are performed in background task except :
- tests of safety devices which are performed sequentially before activation of the real-time monitor,
- taking of snapshots which initiates the isolation procedure performed as soon as the computer state has changed.

Three cases can arise :

A. Normal Disconnection

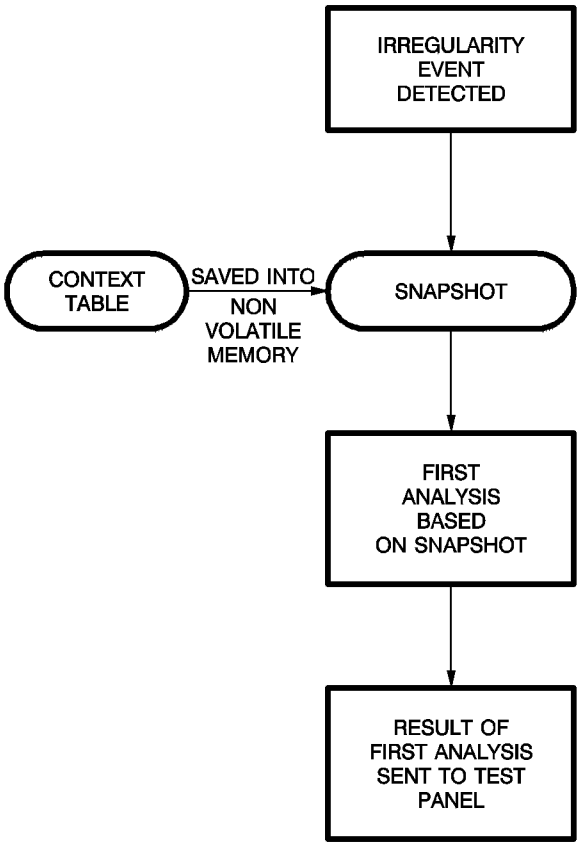
The MTP receives a normal event message and the memory of this snapshot is erased.

B. Abnormal disconnection and unambiguous location

Result of the analysis is sent to the MTP which stores it in memory. The unit fault data are stored in memory in the computer non volatile memories.

C. Abnormal disconnection and ambiguous location (case of power loops)

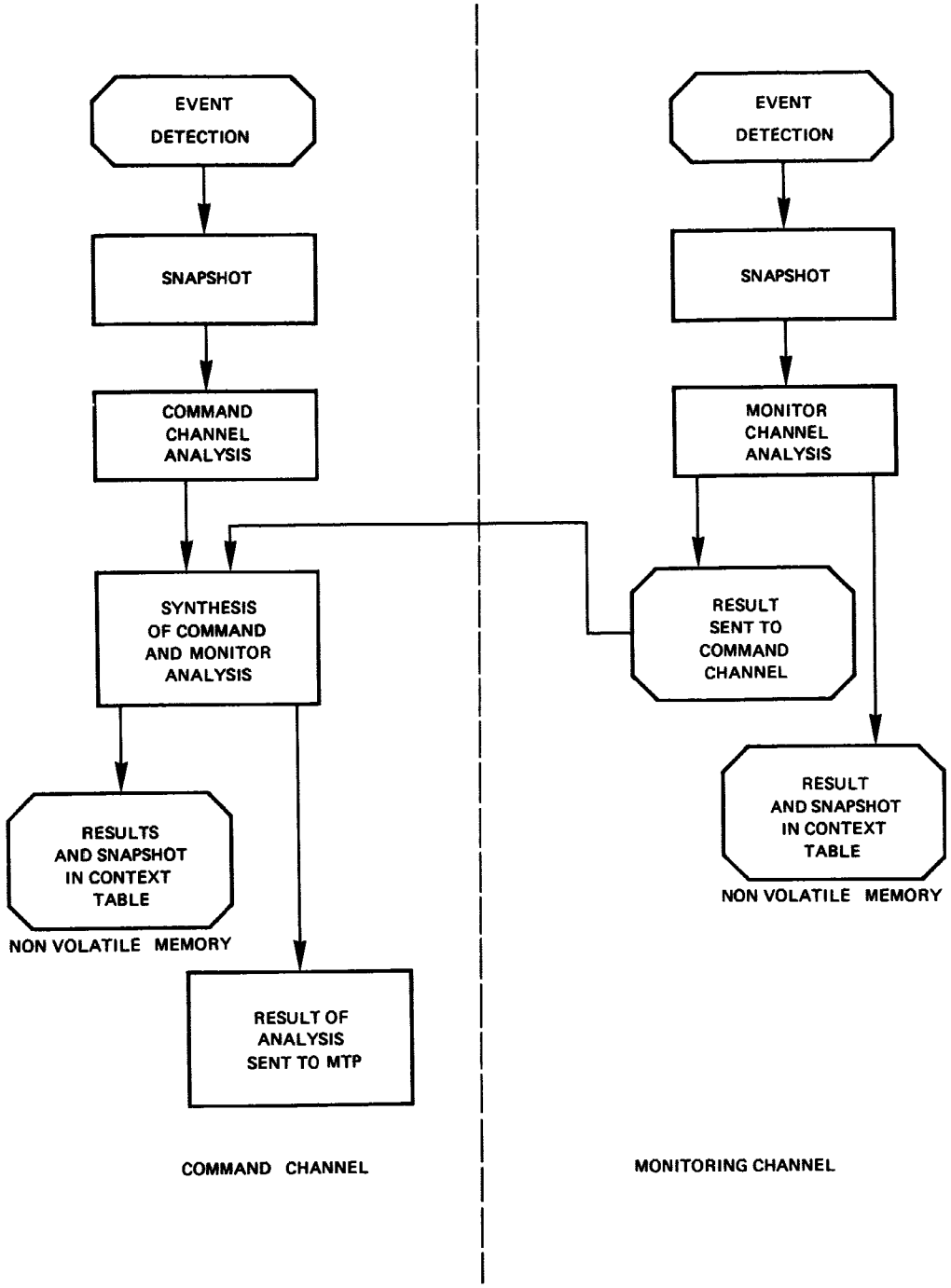
Result of the analysis is sent to the MTP which performs a crosscheck with the other computers by using additional information.
Should this analysis result in an ambiguity the MTP stores it in memory with the label : complementary analysis to be performed.


IRREGULARITY EVENT DETECTED
CONTEXT TABLE
SAVED INTO NON VOLATILE MEMORY
SNAPSHOT
FIRST ANALYSIS BASED ON SNAPSHOT
RESULT OF FIRST ANALYSIS SENT TO TEST PANEL

BM5 22 42 00 0 AAM2 01

MTP - Fault Isolation Procedure
Figure 001

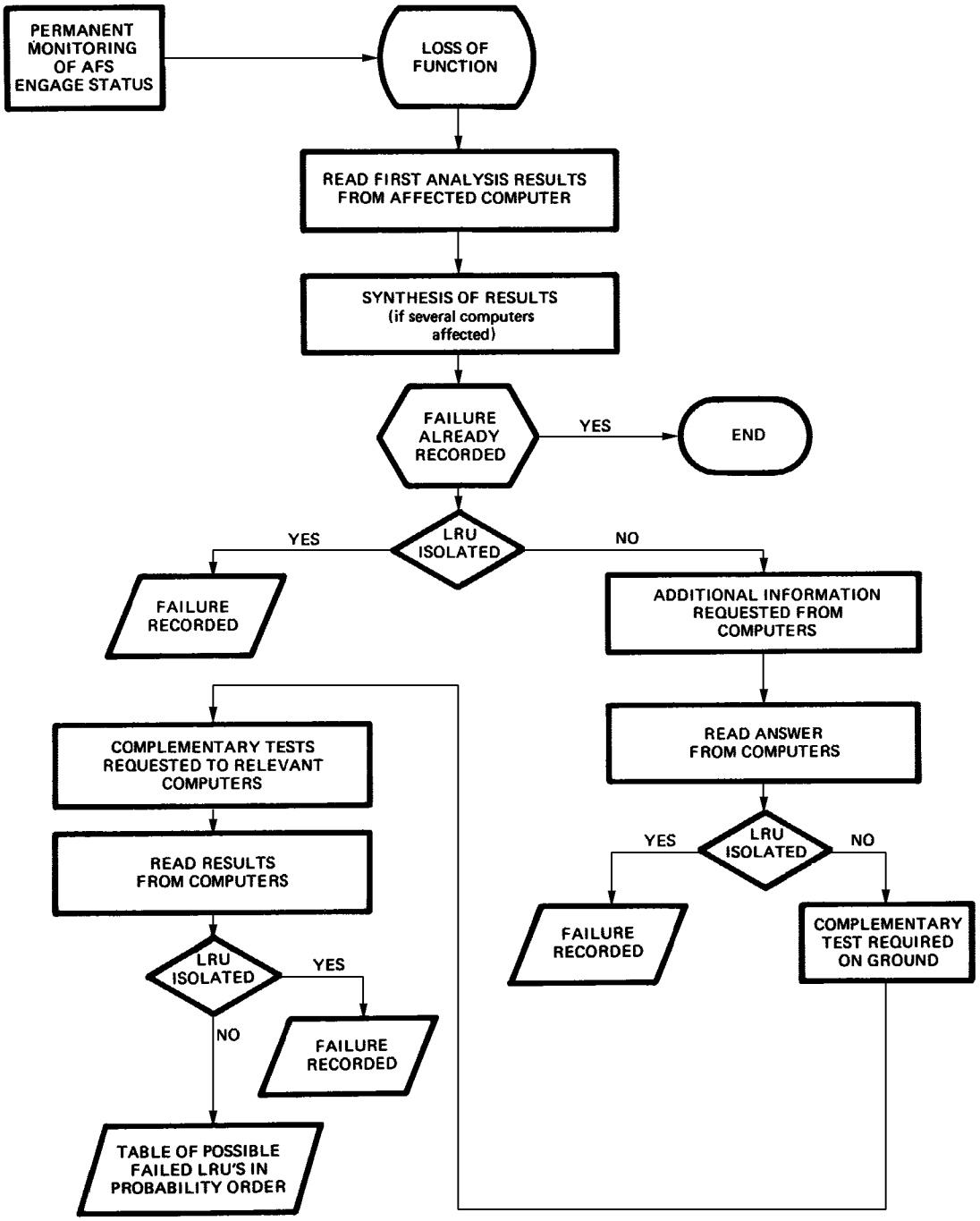
This complementary analysis is automatically performed on the ground after system disconnection (pitch trim, yaw damper and autothrottle systems).
The MTP can memorize up to thirty events on the last six flight legs following the first-in, first-out method (see NOTE).
A failure that occurs several times during the same flight leg is memorized only once.
Only events associated with the last flight leg will be displayed by pressing the DISPLAY pushbutton switch.
NOTE : The last memorized event erases the first memorized one.
(Ref. Fig. 002)
(Ref. Fig. 003)



BM5 22 42 00 0 ACM2 01

FAULT ISOLATION PROCESS INSIDE THE COMPUTER

Fault Isolation Process Inside the Computer
Figure 002



BM5 22 42 00 0 AEM2 01

Fault Isolation Process Outside the Computers (MTP Level)
Figure 003

3. Use of the Maintenance Test Panel

A. Use of the DISPLAY Function (Ref. Fig. 004)

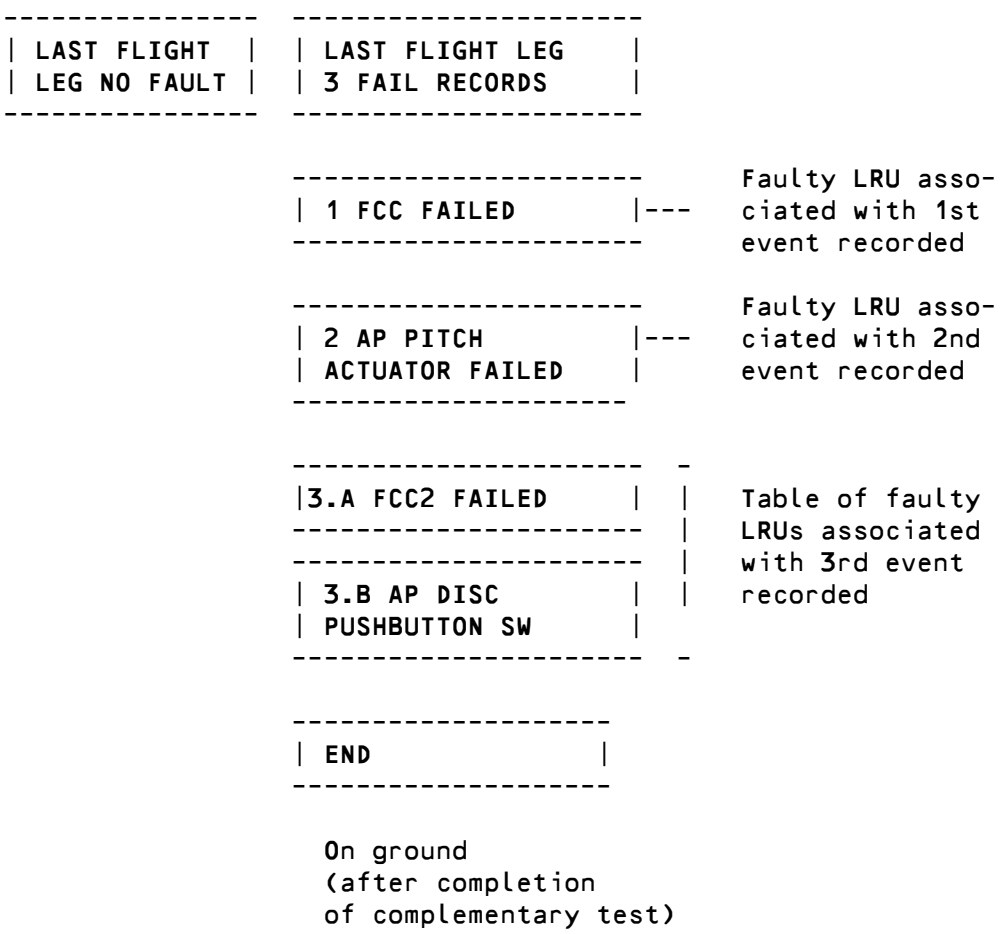
Messages are displayed in plain English on two lines of sixteen alphanumeric characters.
A message is displayed by pressing the DISPLAY pushbutton switch.
The display function can be activated on the ground or in flight.
Display is available at any time during the flight or on the ground.
Several types of messages can be displayed:

REFUSED	If activating GROUND SCAN or AFS TEST in flight (wheel speed > 70 Kts) or if activating AFS TEST when systems are engaged.
WAIT	If activating GROUND SCAN with AFS computers ARINC lines not refreshed (duration 0 to 5 min).
TEST IN PROGRESS	If activating DISPLAY when the failure localization process is running.
SHORT POWER DOWN	Short power down during AFS TEST, LAND TEST, GROUND SCAN or DISPLAY.
END	End of DISPLAY sequence.
TRANSIENT FLAG	A failure was detected once in flight on the associated computer and it was not confirmed on ground after an automatic AFS TEST.
DISCONNECT ALL AFS LEVERS	In case of automatic AFS TEST after a transient flag recorded during the last flight leg.
LAST FLIGHT LEG n FAIL RECORD	AFS/LAND TEST : n FAIL RECORD GROUND SCAN : n FAIL RECORD
xA YAW DAMPER ACTUATOR FAILED	Ambiguity : Most probable failure.
xB FAC 1 FAILED	Ambiguity : Second probable failure.
xC 2 FAILED	Ambiguity : Last probable failure.

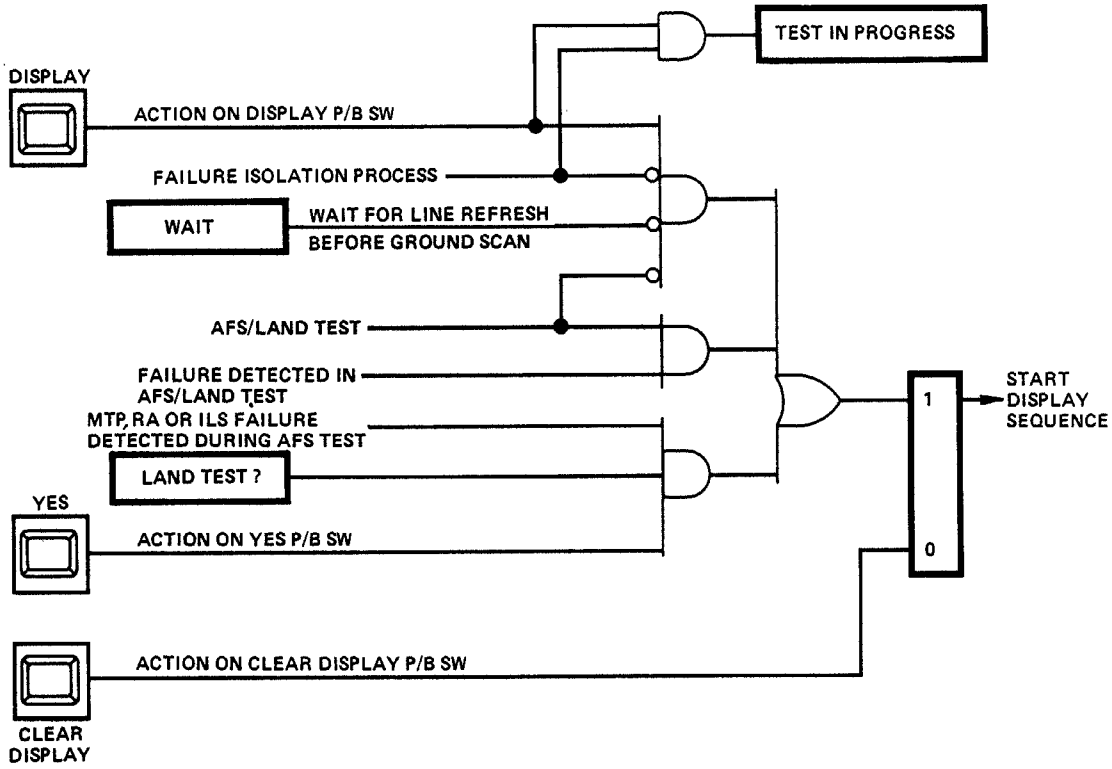
The following table gives an example of the several configurations of messages that can be obtained.
Complementary tests requiring transmission of stimuli can be performed on the ground only. When such tests are required a message is displayed on the MTP together with the relevant LRUs.

R After the **DISPLAY** pushbutton switch has been pressed the messages are
R automatically displayed in cyclic form.
R Each message is displayed for three seconds.
R Keeping the **DISPLAY** pushbutton switch pressed makes the messages shift
R faster.

Examples of Sequences


LAST FLIGHT LEG NO FAULT
LAST FLIGHT LEG 3 FAIL RECORDS
1 FCC FAILED
2 AP PITCH ACTUATOR FAILED
3.A FCC2 FAILED
3.B AP DISC PUSHBUTTON SW
END
Faulty LRU associated with 1st event recorded
Faulty LRU associated with 2nd event recorded
Table of faulty LRUs associated with 3rd event recorded
On ground (after completion of complementary test)

Access to messages recorded in the MTP but not displayed
All the messages memorized by the MTP can be displayed whenever the operator presses the three following pushbutton switches at the same time : DISPLAY , YES and NO.
YES and NO pushbutton switches must be pressed first. Under these conditions all the messages memorized by the MTP and concerning the six previous flights are presented to the operator. The messages presented through the DISPLAY action always concern the last flight leg thus eliminating any ambiguity for the line mechanic.
If no failure has been recorded during the last or current flight leg, the magnetic annunciator is all black. When the magnetic annunciator is black and white, this means that several failures have been recorded by Flight Scanning since the last take off (This magnetic annunciator is not
R activated in GROUND SCAN AFS TEST/LAND TEST.). This magnetic annunciator
R turns black when wheel speed is more than 70KTS.
Alternate solution (MTP with FAILURE RECORD LED)
If no failure has been recorded during the last or current flight leg, the FAILURE RECORD LED is off. When the FAILURE RECORD LED is on, this means that several failures have been recorded by Flight Scanning since the last take off (This FAILURE RECORD LED is not on in GROUND SCAN AFS
R TEST/LAND TEST.). This LED goes off when wheel speed is more than 70KTS.
(Ref. Fig. 004)



BM5 22 42 00 0 BAM2 23 18
DM5

Display Function Activation Logic
Figure 004

R B. AFS TEST

R AFS test has been designed in order to be a single and self-contained test
R to be performed after AFS LRU replacement. As described below, this test
R checks correct installation of an AFS unit on the aircraft by verifying
R good acquisition and reception of peripheral units and integrity of newly
R installed AFS units.
(Ref. Fig. 005)

R (1)Logic for test activation

R For correct AFS TEST accomplishment, aircraft must be configured as
R indicated on MTP front face placard.
R First action on MTP "AFS/LAND TEST" pushbutton switch triggers the AFS
R TEST if following conditions are fulfilled :

R (a)None of AP, AT, TRIM, Y/D levers engaged.

R (b)MTP detects "NOSE GEAR PRESSED" and "WHEEL SPEED" below 70 kts for more
R than 5 seconds.

R (c)"GROUND SCAN" is not active.

R If condition a) or b) is not present, "REFUSED" will be displayed.
R If condition c) is not fulfilled, action on pushbutton switch is
R inhibited until GROUND SCAN is cleared.
R If AFS computer power supply was previously automatically cut off,
R action on AFS/LAND TEST pushbutton switch will first activate the
R safety test of each computer through a discrete signal called "TEST
R POWER ON".
R To stop the AFS TEST, press CLEAR DISPLAY pushbutton switch.

(2)Test description

Test is divided into 5 steps, with a reminder of step number in
progress.

(a)"SELF TEST ON 1"

The MTP checks result of its own sofware test (PROM and RAM) performed
and memorized after the first power up.
If the result is not correct, "MIP FAILED" will be displayed. If the
result is correct, second step is activated.

(b)"SELF TEST ON 2"

The MTP checks reception of AFS computer ARINC bus.
If all bus data are present, this means that all the computers have
successfully performed their safety tests. If not, troubleshooting with
failure location is activated.
A five-minute time out is given for bus reception check. A message
indicating remaining time for this counter is displayed in the window
as follows : "REFRESH TXXX" ; XXX denotes remaining time in seconds
starting at 300.

(c)"SELF TEST ON 3"

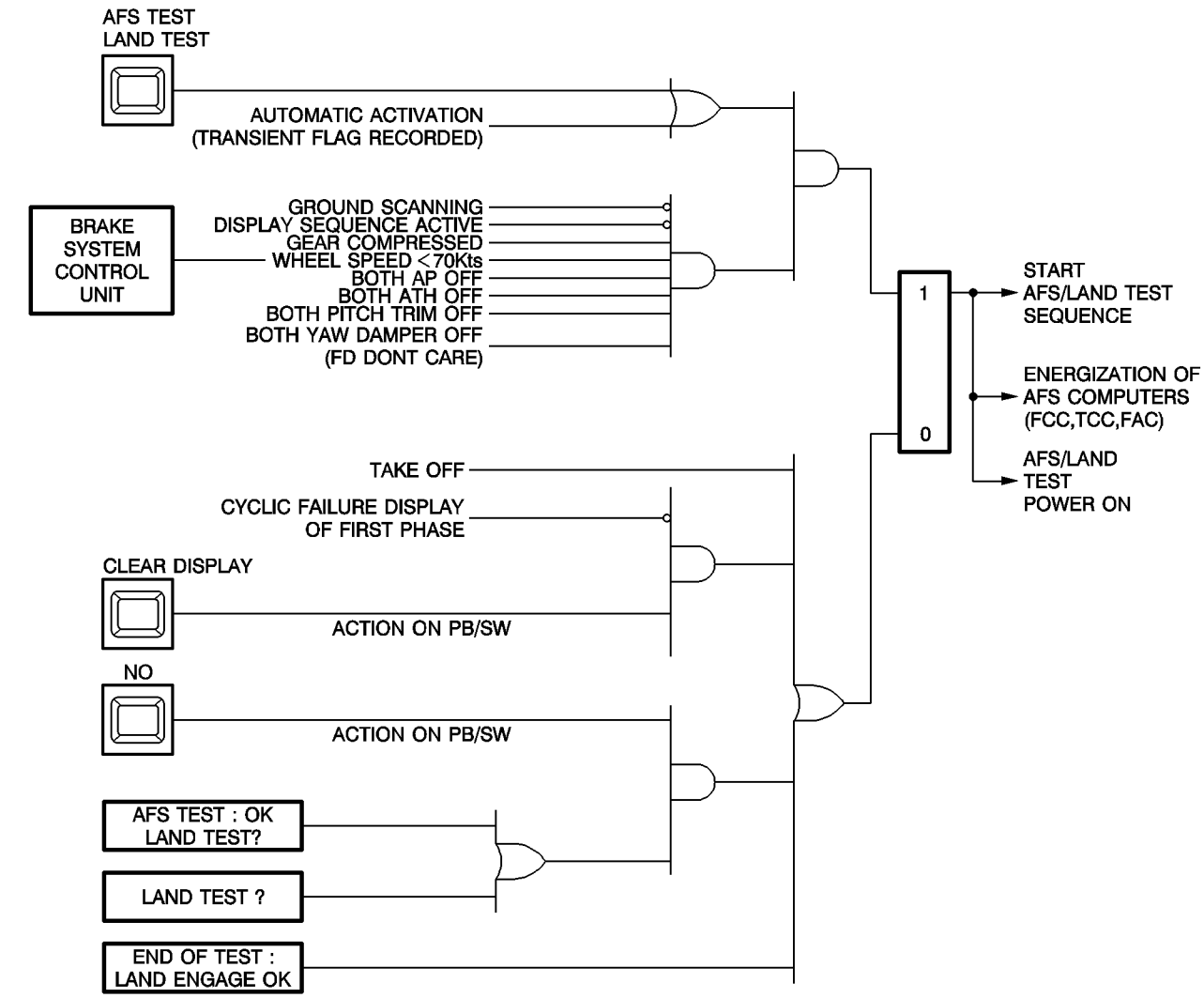
During this step, the MTP interrogates computers for "NOSE GEAR
PRESSED" condition plus "WHEEL SPEED > 70 KTS" for FCC. If there is
no consistency, a failure location procedure is activated as follows :

1 If one computer is in a different configuration, message will be :

- COMPUTER
  FAILED
- NOSE GEAR
  RELAY FAILED

OR

BRAKE SYST CONT
UNIT FAILED
(for FCC only)



NOTE : POWER RE-ESTABLISHMENT FROM MTP
INITIATES SAFETY TEST SEQUENCE
IN AFS COMPUTERS.

BM5 22 42 00 0 BCM2 01

AFS/LAND Test - Activation Logic
Figure 005

R

2 If all computers are not in accordance with MTP, message will be :
- MTP FAILED
- NOSE GEAR RELAY FAILED OR BRAKE SYST CONT UNIT FAILED

3 Discrepancy on more than one computer is considered as a low probability factor.
AFS TEST step 3 is a safety precaution to check the inhibit circuit of AFS TEST in flight.

(d)"SELF TEST ON 4"
When the first three steps have normally run, the MTP sends to each AFS computer an AFS TEST request through ARINC bus and waits for computer answers.
If after a five second delay, one computer has not yet sent its report, test comes back to ARINC bus refreshment check in step 2.
After 3 loops in this sequence, the missing answer is memorized as a failed computer. A detailed description of AFS TEST content in each computer is given below.
- Internal flag monitoring
  Following boolean values are tested for "good" (FF) state in computers :
  FCC : BFAP, BFFD, BFSECTS 3
  FAC : BFTRIM, BFYAW
  TCC : BWDOG opp, BFDW opp, BFSECTS 2, BPWRSUP.
  If any of these flags are faulty, the message will be "COMPUTER FAILED".
- Analog to Digital and Digital to Analog converter check
  Test consists in comparing a stimulus sent through an analog output port with the same signal retransmitted through an analog input port. This test is performed in CMD and MON channels. If comparison fails, message will be : "COMPUTER FAILED".
- Discrete signals test
  Discrete signals are tested according to three different methods according to signal type.

  . Type 1
    These discrete signals are compared between CMD and MON channels.
    In case of discrepency, message will be "COMPUTER FAILED".
    Involved signals in TCC are in the following order :
    ENGINE 1 BLEED (BENG1BLD)
    ENGINE 2 BLEED (BENG2BLD)
    LIFT DUMP 1 (BDUMP1)
    ENGINE 1 ANTI-ICE (BENG1ICE)
    ENGINE 2 ANTI-ICE (BENG2ICE)
    LIFT DUMP 2 (BDUMP2)
    HIGH LIMIT SWITCH (BHLSW)
    ENGINES STOP (BENGSTOP)
    RIGHT GEAR PRESSED (BRGEAR)
    LEFT GEAR PRESSED (BLGEAR)
    SIDE (BSIDE)
    BI TCC CONFIG (BBITCC)
    LAMP TEST (BLMPTST)

IRS3 SWITCHING (BAINSTSW)
Involved Signals in FCC are in the following order :
RIGHT WHEEL SPD (BRWHLSPD)
PFD SWITCHING (BPFDSW)
PFD SWITCHING OPP (BPFDSWO)
TO/GA NO SW (BTOGANO)
TO/GA NC SW (BTOGANC)
INST DISC NO SW (BDISCNO)
INST DISC NC SW (BDISCNC)
F/O FD BAR REMOVAL SW (BOBARFD)
CAPT FD BAR REMOVAL SW (BBARFD)
HYDRAULIC CIRCUIT HEALTHY (BHYDHTY)
A/C 28VDC BARS SPLIT (BSPLIT)
ENGINES STOP (BENSTOP)
Involved signals in FAC are in the following order :
SRS MODE OWN (BSRS)
SRS MODE OPP (BOSRS)
AP OPP ENGD CWS MON (BOAPCWSM)
AP OPP ENGD CWS CMD (BOAPCWSC)
AP OPP ENGD CMD (BOAPENGC)
AP OPP ENGD MON (BOAPENGM)
AP OWN ENGD CWS CMD (BAPCWSC)
AP OWN ENGD CWS MON (BAPCWSM)
AP OWN ENGD CMD (BAPENGC)
AP OWN ENGD MON (BAPENGM)
CAP ELECTRIC TRIM UP (BCAPTUP)
CAP ELECTRIC TRIM DOWN (BCAPTDN)
THS LIMIT SW 1 (BLIMSW1)
THS LIMIT SW 2 (BLIMSW2)
THS LIMIT SW 3 (BLIMSW3)
SIDE (BSIDE)
HYDRAULIC CIRCUIT HEATHY (BHYDHY)
LEFT GEAR PRESSED SW (BLGEARPR)
RIGHT GEAR PRESSED SW (BRGEARPR)
STALL WARNING OPP (BOSTALW)
STALL WARNING OWN (BSTALW)
ANGLE OF ATTACK TEST (BAOATST)
F/O ELECTRIC TRIM UP (BFOUP)
F/O ELECTRIC TRIM DOWN (BFODN)
TRIM CLUTCHED (BTCLUT)

. Type 2
These discrete signals are compared between CMD and MON channels.
In case of dissymmetry, message will be :
A - COMPUTER FAILED
B - PERIPHERAL FAILED
Involved signals in FCC are, in the following order :

SIGNAL	SW LABEL	ASSOCIATED PERIPHERAL
A/T IN SPEED MODE OWN	BTCCSPD	TCC OWN
A/T IN SPEED MODE OPP	BOTCCSPD	TCC OPP *
PITCH CLUTCH SWITCH	BPCLUTSW	PITCH ACTUATOR
ROLL CLUTCH SWITCH	BRCLUTSW	ROLL ACTUATOR
YAW CLUTCH SWITCH	BYCLUTSW	YAW ACTUATOR
OPP AP ENGAGED	BOAP	FCC OPP
OPP FD ENGAGED	BOFD	FCC OPP
AP ENGAGE SWITCH	BAPSW	FCU
YAW DAMPER ENG OPP	BOYDAM	FAC OPP
YAW DAMPER ENG OWN	BYDAM	FAC OWN
TRIM ENGAGED OPP	BOTRIM	FAC OPP
TRIM ENGAGED OWN	BTRIM	FAC OWN

* If installed
Involved signals in FAC are, in the following order :

SIGNAL	SW LABEL	ASSOCIATED PERIPHERAL
CWS LD1 OWN	BCWSLDI	FCC OWN
CWS LD1 OPP	BOCWSLDI	FCC OPP
TRIM TOGGLE SWITCH	BTENGSW	FAC/ATS ENG UNIT
YAW TOGGLE SWITCH	BYENGSW	FAC/ATS ENG UNIT
GEAR DOWN	BGEARDN	LEFT (RIGHT) GEAR RELAY

Involved signals in TCC are, in the following order :

SIGNAL	SW LABEL	ASSOCIATED PERIPHERAL
TRIM OWN ENGAGED	BTRIM	FAC OWN
TRIM OPP ENGAGED	BUTRIM	FAC OPP
YAW DAMPER OWN ENGD	BYDAMP	FAC OWN
YAW DAMPER OPP ENGD	BOYAMP	FAC OPP
AP OWN ENGAGED IN CMD	BCMD	FCC OWN
FD OWN ENGAGED	BFD	FCC OWN
AP OPP ENGAGED IN CMD	BOCMD	FCC OPP
FD OPP ENGAGED	BOFD	FCC OPP
TO/GA SWITCH	BTOGASWC	TO/GA LEVERS
INST DISC SWITCH	BIDISSWC	INST DISC SWITCH ATS

. Type 3

Some of the discrete signals are, in addition, tested to be in an expected state. If state is not the expected one, message will be : PERIPHERAL FAILED
Involved signals in FCC are, in the following order :

SIGNAL	EXPECTED STATE	SW LABEL	PERIPHERAL
RIGHT WHEEL SPEED	CLOSED	BRWHLSPD	BRAKE SYST CONT UNIT
PFD SWITCHING OWN	OPEN	BPFDSW	FD SWITCHING
PFD SWITCHING OPP	OPEN	BOPFDSW	FD SWITCHING
TO/GA NO SWITCH	OPEN	BTOGANO	TO/GA LEVER
TO/GA NC SWITCH	CLOSED	BTOGANC	TO/GA LEVER
INST DISC NO SWITCH	OPEN	BDISCNO	INST DISC SWITCH
INST DISC NC SWITCH	CLOSED	BDISCNC	INST DISC SWITCH
F/O FD BAR REMOVAL	OPEN	BOBARFD	EFIS2/FD BAR REM SW
CAPT FD BAR REMOVAL	OPEN	BBARFD	EFIS1/FD BAR REM SW

Involved signals in TCC are, in the following order :

SIGNAL	EXPECTED STATE	SW LABEL	PERIPHERAL
TO/GA NC SWITCH	CLOSED	BTOGASWC	TO/GA LEVER
INST DISC NC SWITCH	CLOSED	BIDISSWC	INST DISC SWITCH
TO/GA NO SWITCH	OPEN	BTOGASWO	TO/GA LEVER
INST DISC NO SWITCH	OPEN	BIDISSWC	INST DISC SWITCH

- Input Flag tests
  Input flags are tested for validity. They consist in either ARINC inputs or monitored discrete signals.
  Synthesis of CMD and MON diagnosis is made in CMD and sent to the MTP.
  Monitored flags for FCC are :

FLAG	CMD	MON	SW LABEL	REMARKS
FCU F/W	X	X	BFFCU	SSM' = F/W or no refresh
TRS COMPARISON	X	X	BIRSCP	IRS comparison for engagement
TRS COMPARISON	X	X	BIRSCA	IRS comparison for capacity
IRS OWN F/W	X	X	BFIRS	SSM = F/W or no refresh
TRACK ANGLE OWN F/W	X	X	BFIRSTK	" "
TRACK ANGLE COMPARISON	X	X	BIRSTKPA	
IRS IN ATT MODE	X	X	BNIRSAT	
IRS3 IN ATT MODE	X	X	BTRKS3AT	
TRACK ANGLE IRS3	X	X	BTRK3CP	
HEALTHY				
HEADING COMPARISON	X	X	BCHDG	
HEALTHY				
ADS F/W	X	X	BFADC	SSM = F/W or no refresh

FLAG	CMD	MON	SW LABEL	REMARKS
FAC F/W	X	X	BFFAC	" "
LOC F/W	X	X	BFLOC	" "
GLIDE SLOPE F/W	X	X	BFGS	" "
RADIO ALTI F/W	X	X	BFRA	" "
GROUND SPEED F/W	X	X	BFGRDSPD	" "
GROUND SPEED COMPARISON	X	X	BGRSCOMP	NORTH and EAST velocity comp
PITCH DYN ROD COMPARISON	X	X	BFPITROD	
GROUND SPEED OPP F/W	X	X	BFOGDSPO	SSM = F/W or no refresh
PFD F/W	X	X	BFPFD	discrete signal from EFIS
PFD ATT F/W	X	X	BFPFDAT	" "
PFD OPP F/W	X	X	BFOPFD	
PFD OPP ATT F/W	X	X	BFOPFDATT	" "
LOC OPP F/W	X		BFOLOCM	SSM = F/W or no refresh
GLIDE SLOPE OPP F/W	X		BFOGSM	" "
VOR F/W	X		BFVOR	" "
FCU DISCRETE F/W	X		BFFCUD	discrete signal from FCU
FCC OPP F/W	X		BFOFCCAR	SSM = F/W or no refresh
FWC ACQ FAIL	X		BFWCFW	discrete signal from FWC
FWC OPP ACQ FAIL	X		BOFWCFW	" " "
ROLL STEERING F/W	X		BFPHIC	SSM = F/W or no refresh
CWS RODS CONNECTED	X		BCWSON	discrete signal from dyn rods

Monitored flags for FAC are :

FLAG	CMD	MON	SW LABEL	REMARKS
26VAC/400HZ MONITORING	X	X	B26LD	Internal power supply monitor
YAW DAMPER MONITORING	X	X	BFYAW	Internal yaw damper monitor
TRIM MONITORING	X	X	BFTRIM	Internal trim monitor
FAC OPP F/W	X	X	BFODISC2	Discrete word 2
MACH OWN F/W	X	X	BFMACH	SSM = F/W or no refresh
AOA OWN F/W	X	X	BFAOAOWN	" " "
COMP SPEED F/W	X	X	BFCPSPD	" " "
AOA OPP F/W	X	X	BFOAOA	" " "
COMP SPEED OPP F/W	X	X	BFOCPSPD	" " "
YAW RATE IRS3 F/W	X	X	BFY RATE 3	" " "
SPOILER POS F/W	X	X	BFSPOIL	" " "
ROLL ANGLE F/W	X	X	BFROLL	" " "
YAW RATE F/W	X	X	BFYRATE	" " "
LONGI ACCEL F/W	X		BFXACC	" " "
VERTICAL ACCEL F/W	X	X	BFZACC	" " "
PITCH ANGLE F/W	X		BFPITCH	" " "
RADIO HEIGHT F/W	X		BFHRA	" " "
FLAPS POS F/W		X	BFFLA	" " "
SLATS POS F/W		X	BFSLA	" " "

Monitored flags for TCC are :

FLAG	CMD	MON	SW LABEL	REMARKS
WATCHDOG MONITOR	X		BWDOGM	Validity check from opp lane
WATCHDOG COMMAND		X	BWDOGC	" "
DISCRETE WORD MONITOR	X		BFDWN	" "
DISCRETE WORD COMMAND		X	BFDWC	" "
SAFETY TEST OK	X	X	BFSECTS2	Safety test result check
OPPOSITE POWER MONITORING	X	X	BPWRSUP	
SELECTED ADC F/W	X	X	BADCFLD	SSM = F/W or no refresh
26VAC MONITORING	X	X	BF26V	26VAC detector monitoring
ACTUATOR NOT CNCTD	X		BACTNO	discrete signal from actuator
DYNAMOMETRIC RODS NOT CNCTD	X		BFSSNO	" " " dyn rods
RIGHT TLA F/W	X	X	BFTLAR	synchro monitoring
LEFT TLA F/W	X	X	BFTLAL	" " "
TRP F/W	X		BFARTRP	SSM = F/W or no refresh
SELECTED TAT F/W	X		BFSELTAT	" " "
FCU F/W		X	BFARFCU	" " "
FMC F/W	X	X	BFFMC	SSM = F/W or no refresh
FAC DISCRETE WORD F/W	X	X	BFDWFAC	" "
RA 1 F/W	X		BFRAX	" "
RA 2 F/W		X	BFRAZ	" "
FCC DISC WORD F/W		X	BFFCCDW	" "
OPPOSITE TCC F/W *	X	X	BFPTCC	" "

* If installed

(e)"SELF TEST ON 5"

When the MTP has received report from all computers, synthesis is made within the MTP and display is activated.

NOTE : Self test 5 also displays any failure memorized during last flight leg related to alpha probe comparison within the FAC. This message is only displayed after the first AFS TEST then erased from the FAC.

(3)Testing of AFS control panels during AFS TEST

(a)Flight Control Unit

At first power up, an internal self-test is performed in both sides of the FCU. According to test results, a discrete output port (FCU/F/W) is set to a specific state. This logic condition is tested by FCC1 (for SIDE 1 of FCU) and by FCC2 (for SIDE 2) during SELF TEST ON 4 step.
If FCU DISCRETE F/W (BFFCUD) is set to F/W condition, message will be :
A - FCU FAIL
B - FCC1(2) FAIL
This enables FCU side 2 failure to be detected.

(b)TRP

The TRP is tested by checking TRP F/W logic signal inside TCC during SELF TEST ON 4 step. By checking this flag, failure of TRP is detected.

(4)END of AFS test
When "SELF TEST ON 5" step is accomplished, results from all computers are analyzed.
If no failure is detected, AFS TEST is correct and display is :

```
------------------
|  AFS TEST : OK  |
|  LAND TEST ?    |
------------------
```

If some results are not good, the MTP synthesizes received data. All failures detected by different computers are periodically displayed.
After pressing "CLEAR DISPLAY", message "LAND TEST ?" is shown.
It has to be noted that, considering each computer sends one and only one fault per AFS TEST (in fact the first), AFS must be troubleshoot and AFS TEST run as many times as result is not :

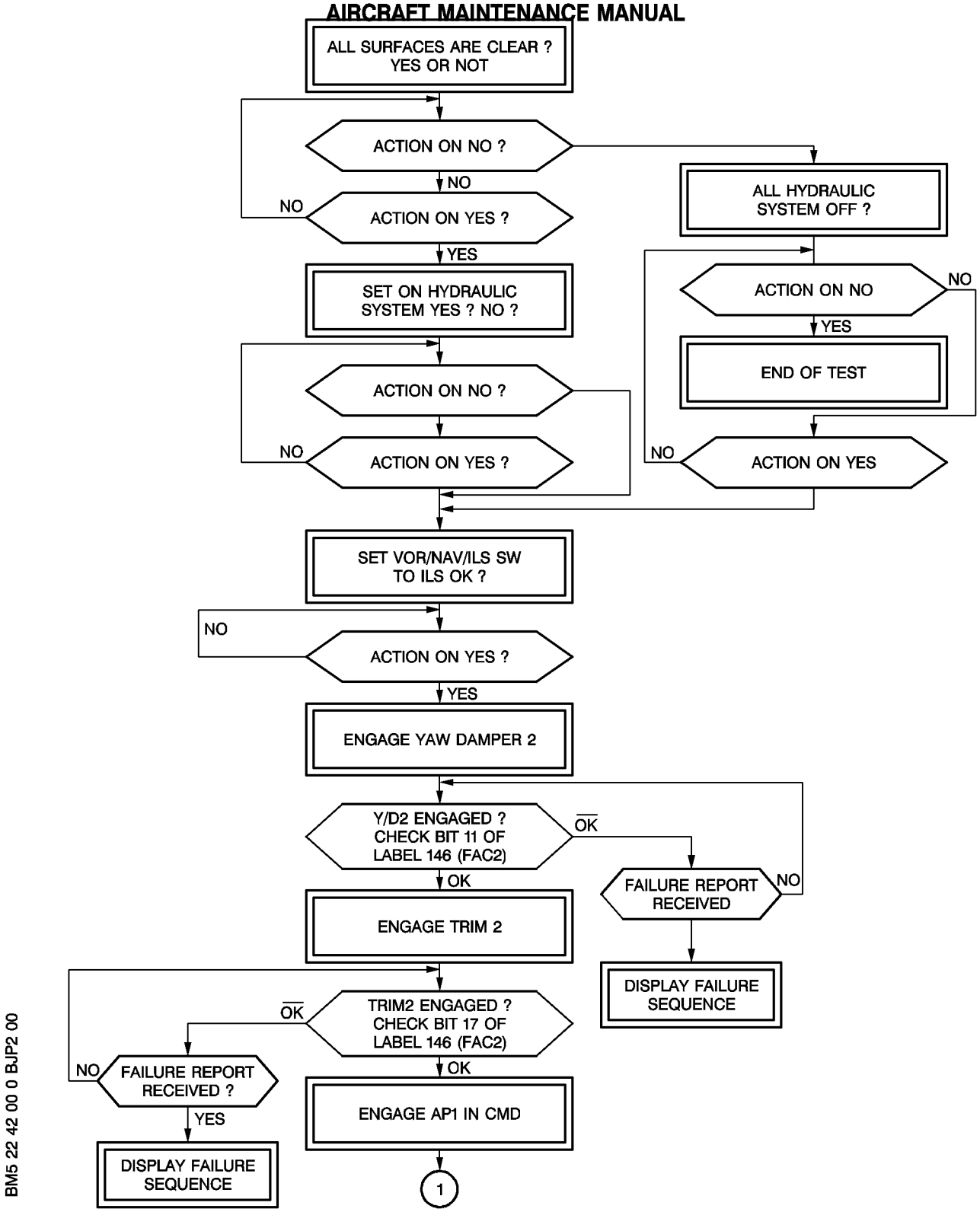
```
------------------
|  AFS TEST : OK  |
------------------
```

C. LAND TEST
(Ref. Fig. 005)
(1)General
The purpose of the automatic LAND TEST is to provide operators with a quick, simple and "self contained" test in order to check availability of CAT 3 autoland function.
This test is not mandatory and will only be performed at the airlines' discretion.
(2)Description
(a)System architecture
The MTP (Maintenance Test Panel) is the interface between the LRUs of the Automatic Flight System and the maintenance team.
By means of ARINC 429 bus interconnection (inputs and outputs), the MTP triggers test activation, monitors running of test, displays actions to be taken by operator or failure messages if existing.
Inside each main AFS computer (FCC, FAC, TCC), a specific subroutine is activated upon request of a LAND TEST in order to initiate fault isolation (this fault isolation program is identical to the one used during flight and ground scanning).
Inside FCC, some additional conditions are forced :
- Radio height = 500 ft. if ATS is not in speed mode or 300 ft. in other case.
- ILS deviation signals are forced to zero.

Since it is possible to perform LAND TEST without hydraulic power on the aircraft, the hydraulic conditions are inhibited within engage logics.
- To stop the LAND TEST, press CLEAR DISPLAY pushbutton switch.

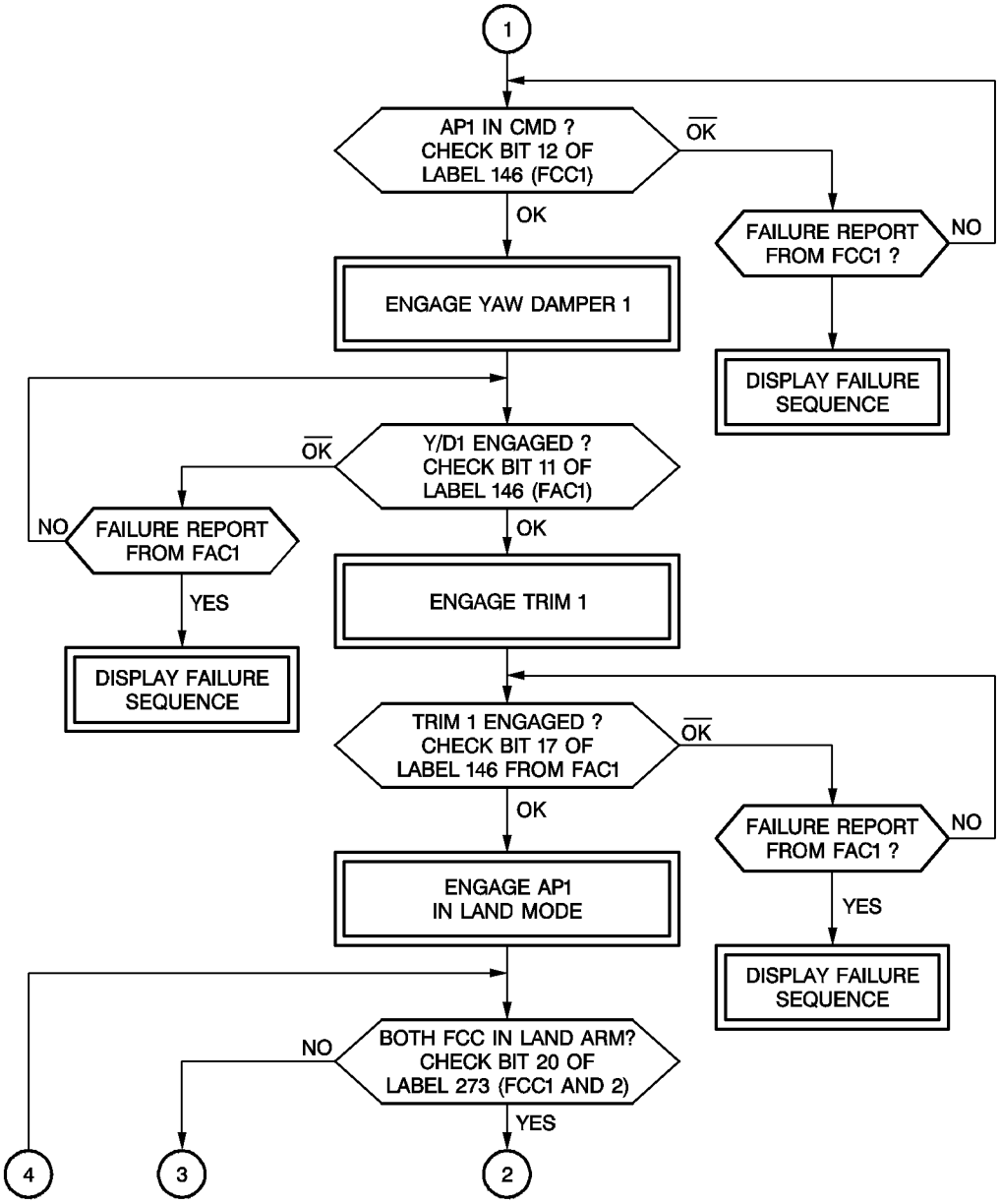
(b)Test description
Access to LAND TEST is possible only after AFS TEST completion even if a failure has been detected during AFS TEST except in case of MTP, ILS or RA detected failure. In this case, the display sequence is re-initiated until CLEAR DISPLAY pushbutton switch is pressed.
(Ref. Fig. 007, 008, 009)

(Ref. Fig. 010, 011, 012)
(Ref. Fig. 013, 014, 015)
(Ref. Fig. 006)



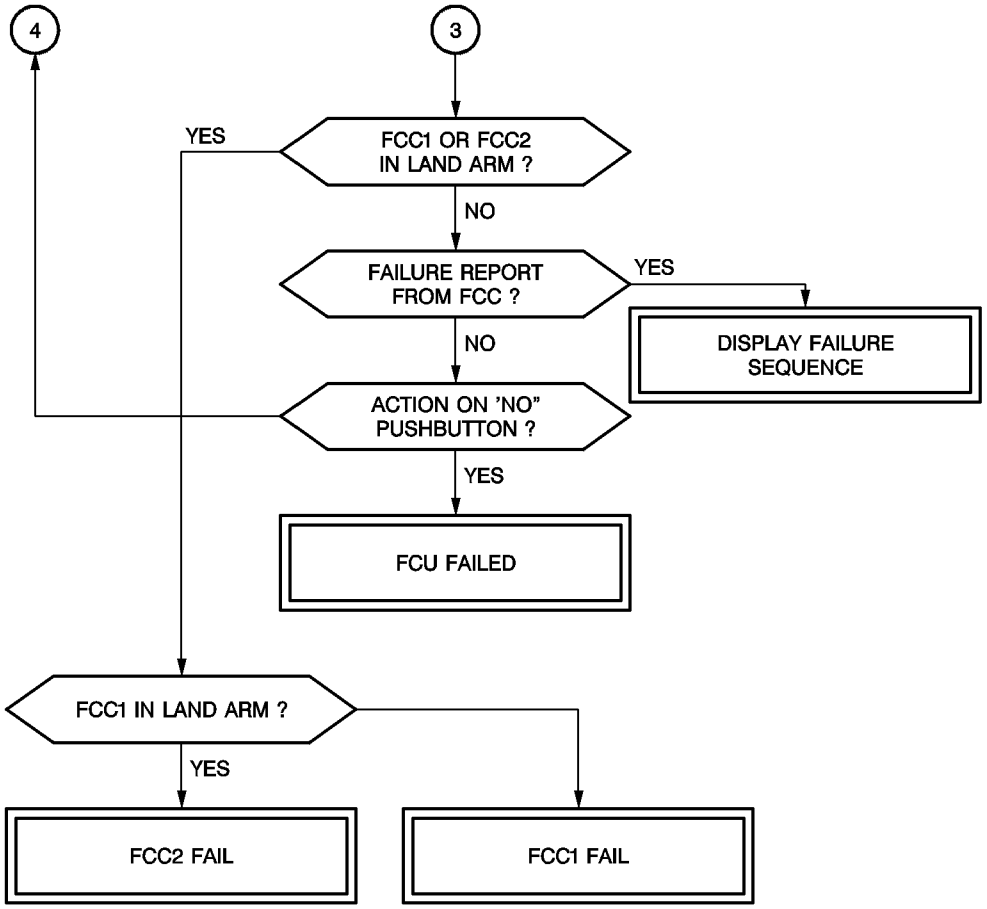
BM5 22 42 00 0 BJP2 00

LAND TEST - Sheet 1
Figure 006



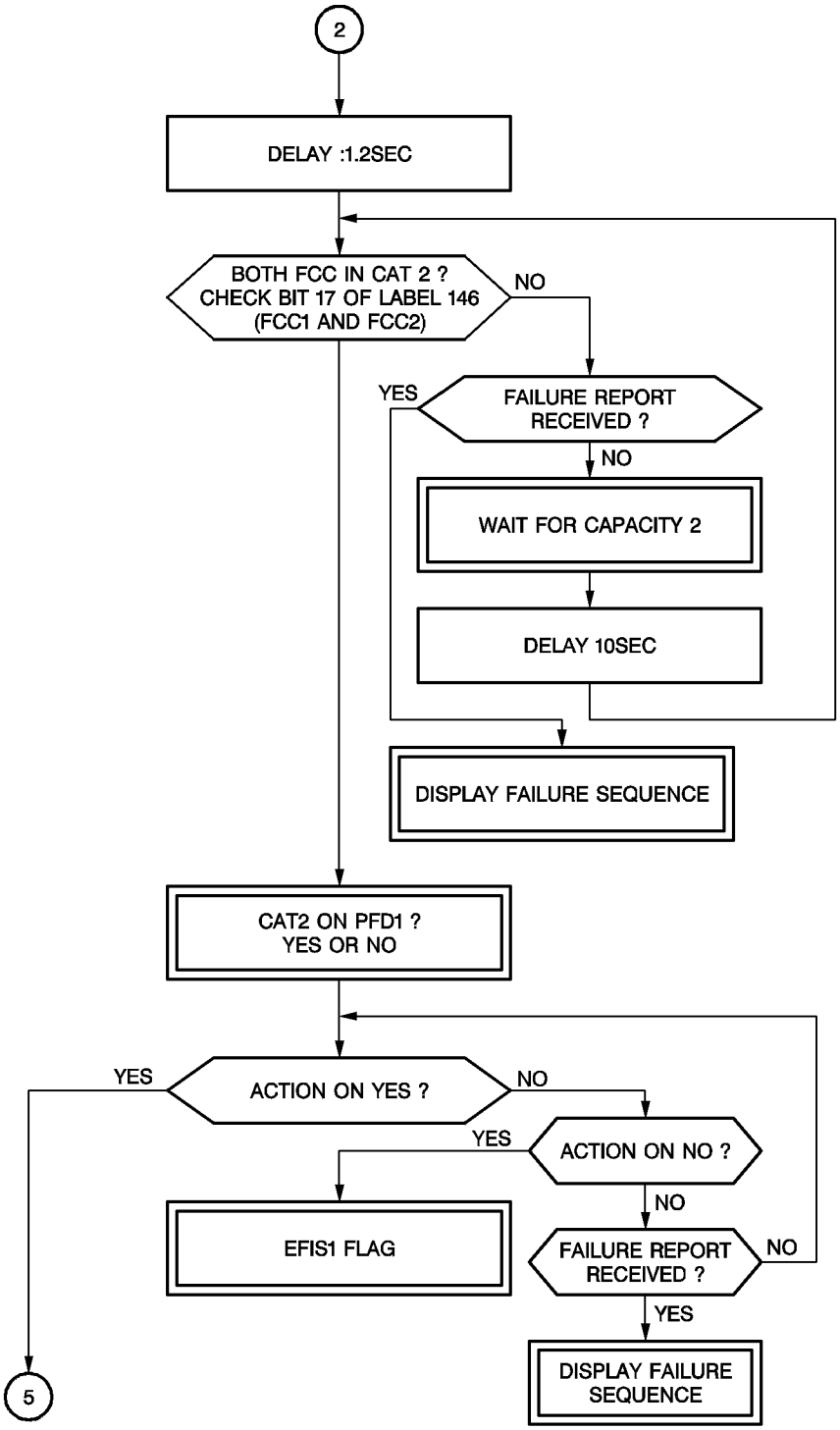
BM5 22 42 00 0 BLM2 00

LAND TEST - Sheet 2
Figure 007



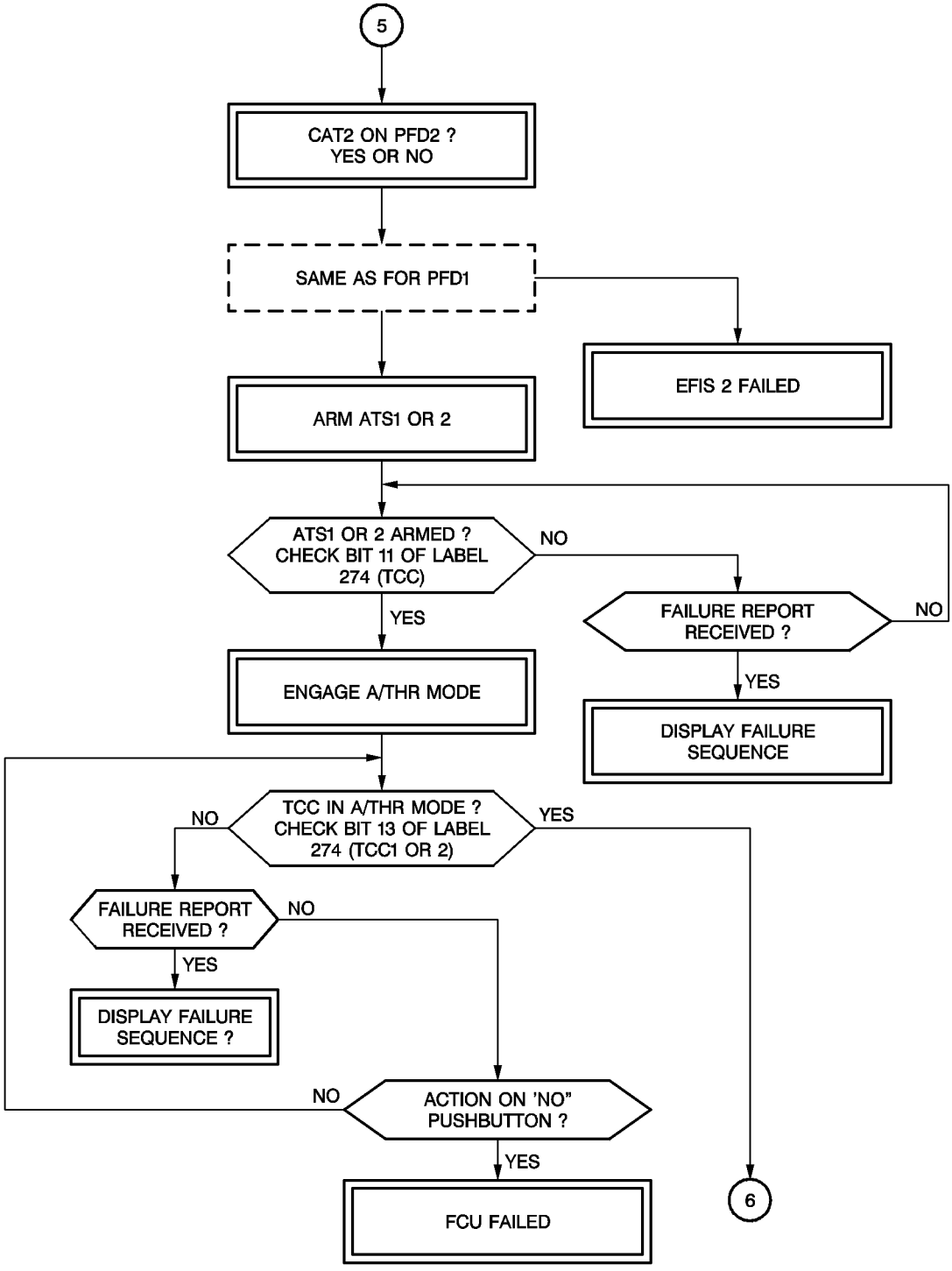
BM5 22 42 00 0 CMM2 00

LAND TEST - Sheet 3
Figure 008



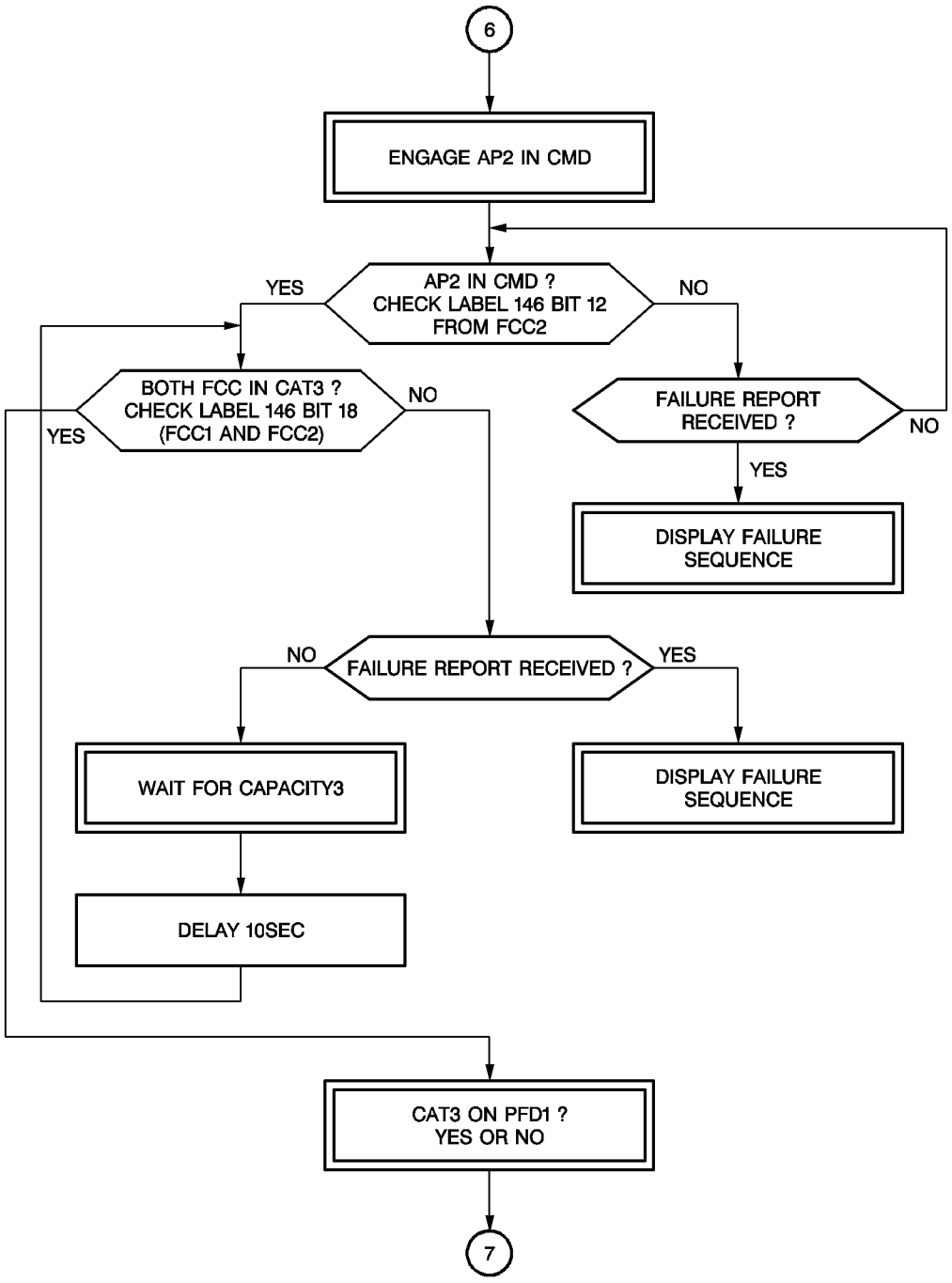
BM5 22 42 00 0 BNM2 00

LAND TEST - Sheet 4
Figure 009



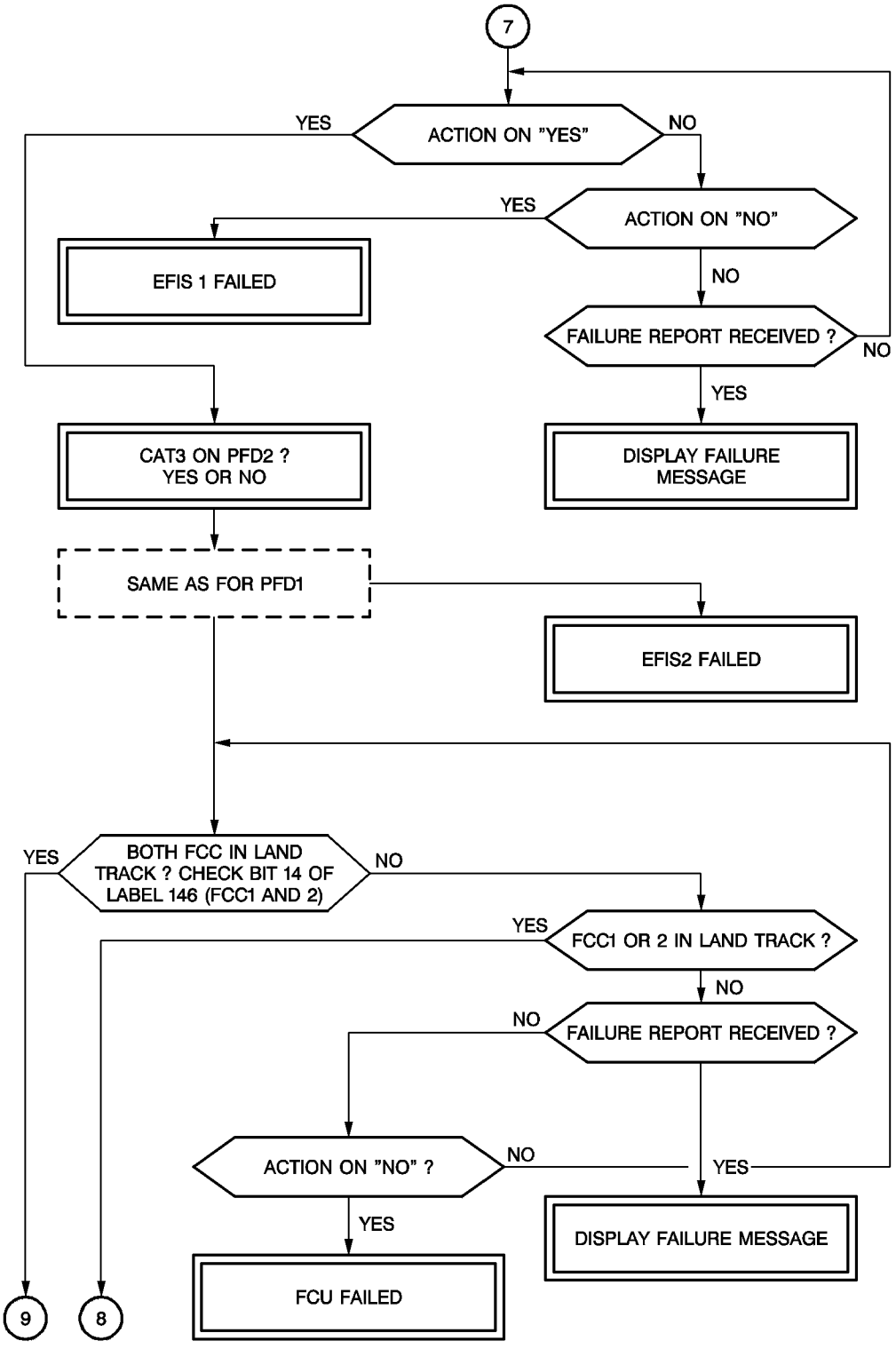
BM5 22 42 00 0 CPM2 00

LAND TEST - Sheet 5
Figure 010



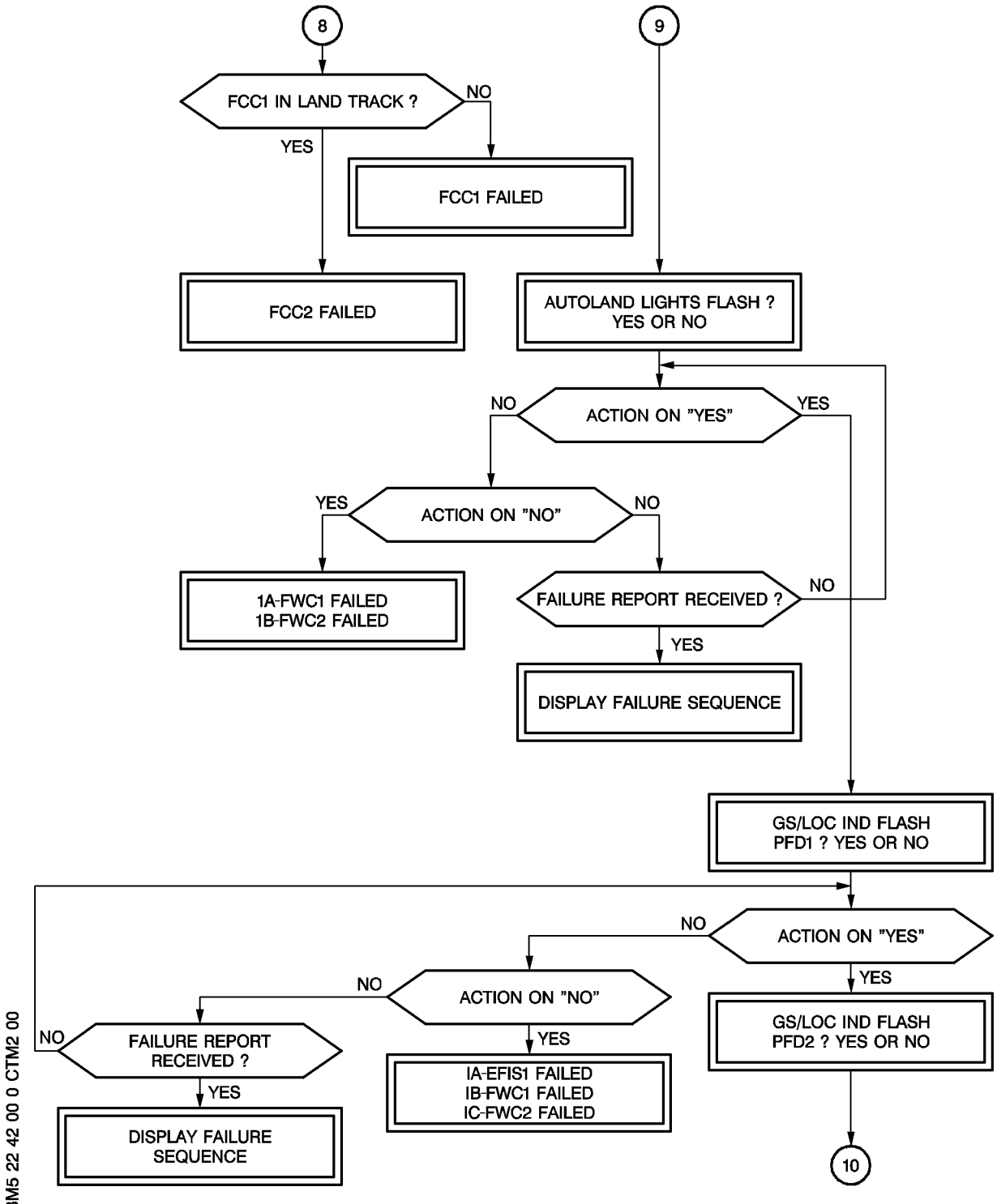
BM5 22 42 00 0 CRM2 00

LAND TEST - Sheet 6
Figure 011

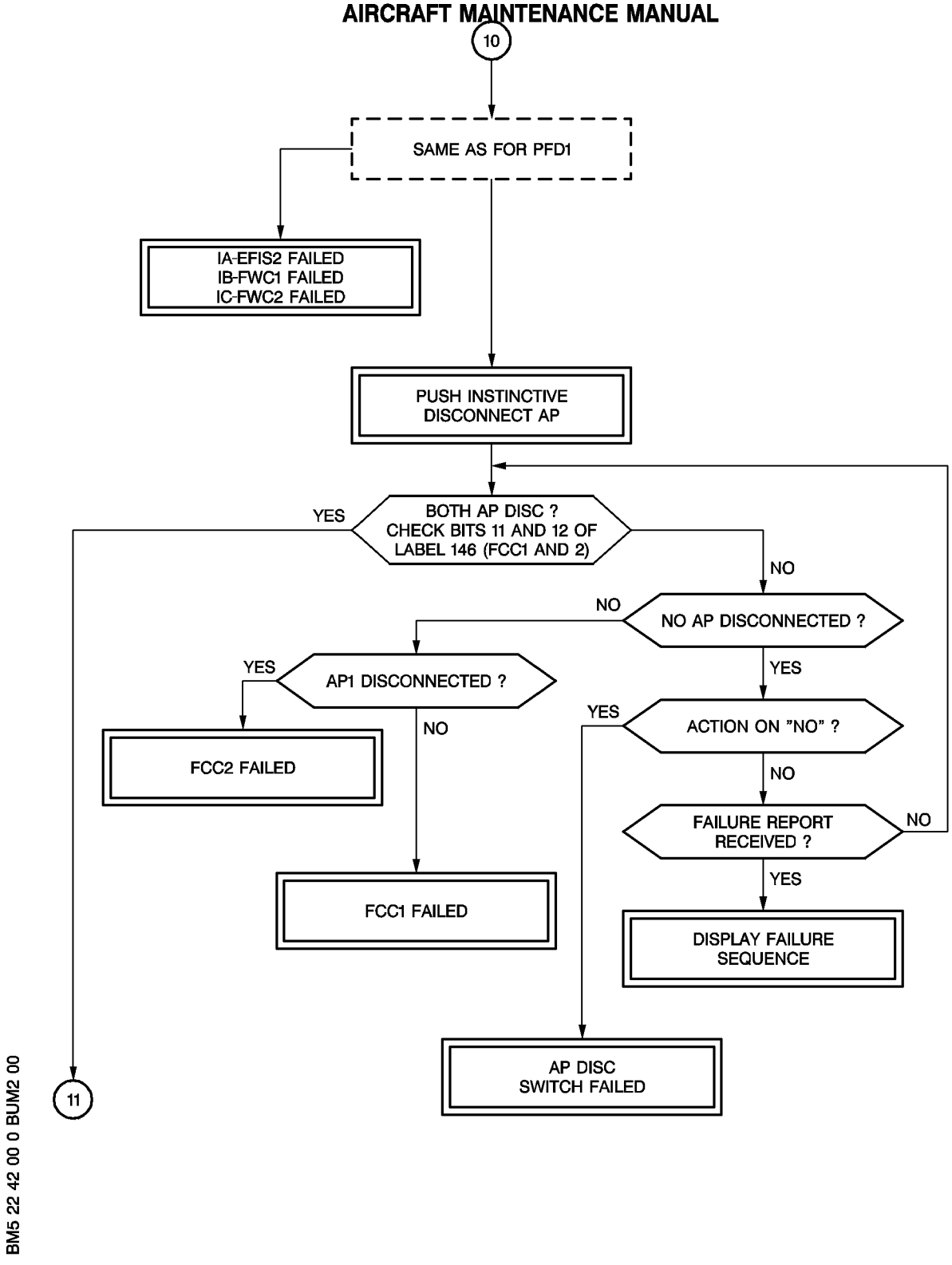


BM5 22 42 00 0 BSM2 00

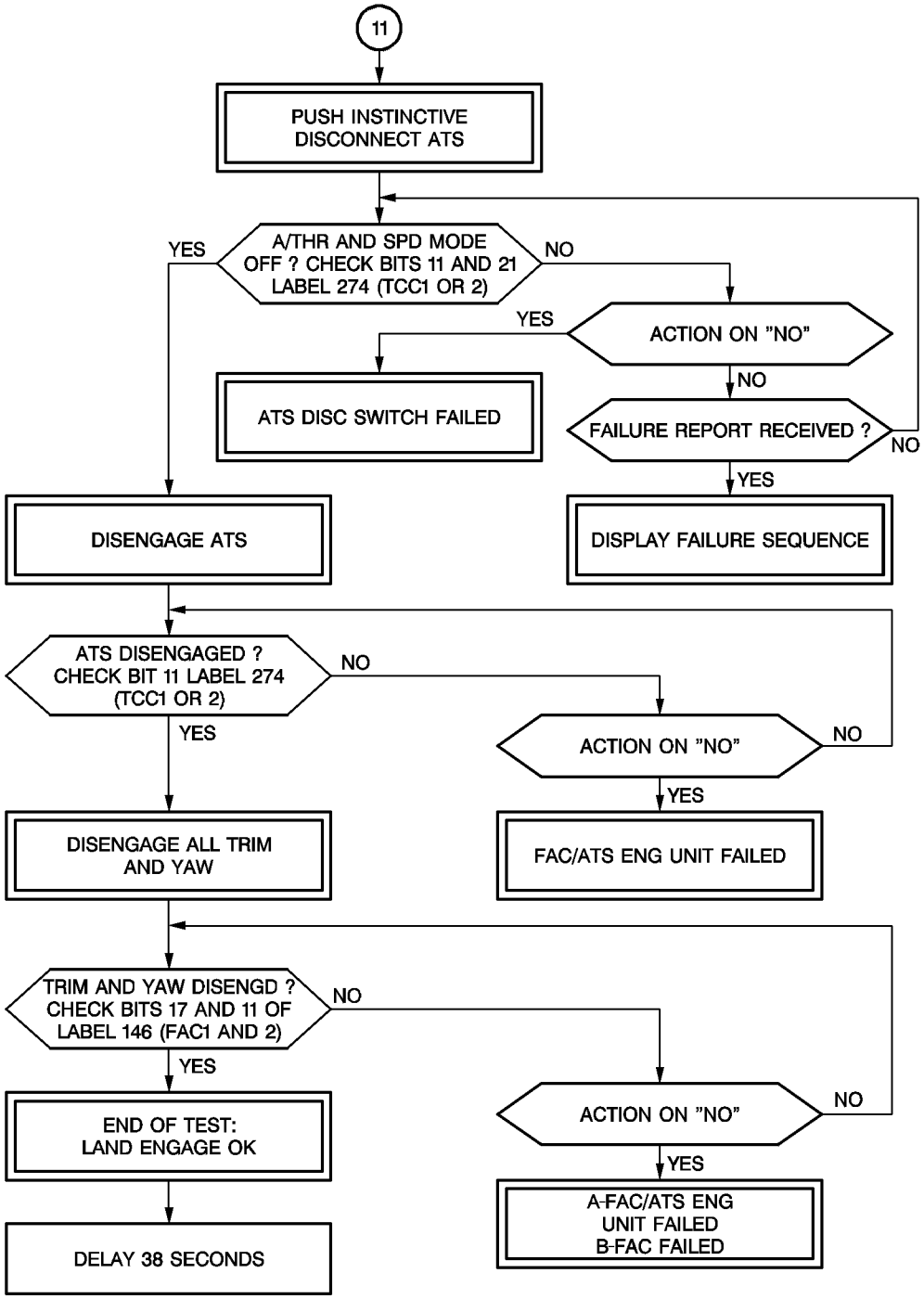
LAND TEST - Sheet 7
Figure 012



LAND TEST - Sheet 8
Figure 013



LAND TEST - Sheet 9
Figure 014



BM5 22 42 00 0 CVM2 00

LAND TEST - Sheet 10
Figure 015

D. TRANSIENT FLAG

The TRANSIENT FLAG message has been introduced in the fault isolation program of a AFS in order to avoid unjustified removals of computers. These unjustified removals may be reduced by confirming a transient flag which occurred during the flight with automatic activation of a test of the computers on ground.

(1)Description

(a)In flight

A failure of one computer will be evaluated as a transient flag if the two following conditions are met :

- an AFS computer (FCC, FAC, TCC) is declared faulty without ambiguity
- this computer has not already been source of a transient flag during the current leg.

In this case, the MTP front face magnetic indicator will remain locked and the displayed message will be : X-(COMPUTER) TRANSIENT FLAG.

R Alternate solution (MTP with FAILURE RECORD LED)

R In this case, the MTP front face FAILURE RECORD LED will remain off

R and the displayed message will be : X-(COMPUTER) TRANSIENT FLAG.

If the computer has already been source of a transient flag, the message will be : X-(COMPUTER) FAIL

and the MTP magnetic indicator will show clover.

R Alternate solution (MTP with FAILURE RECORD LED)

R If the computer has already been source of a transient flag, the

R message will be : X-(COMPUTER) FAIL

R and the MTP FAILURE RECORD LED will be on.

It must be noted that action on DISPLAY pushbutton switch is possible in flight.

(b)After the flight

If as least one transient flag message was stored during last flight leg, the MTP will refuse all other functions till the transient flag situation is not studied.

To solve the transient flag, after occurrence of the automatic power cut off, the MTP will command the powering of AFS computers which will perform their "SAFETY TEST" and their "AFS TEST".

The MTP will only take into account results coming from computers affected by a transient flag message.

If the computer is confirmed as failed by the test, the message will become :

X-(COMPUTER) FAIL.

If the test result is good, the message will remain :

X-(COMPUTER) TRANSIENT FLAG.

If the message list of the last flight leg only comprises transient flag indication, the MTP magnetic indicator will remain black.

R Alternate solution (MTP with FAILURE RECORD LED)

R If the message list of the last flight leg only comprises transient

R flag indication, the MTP FAILURE RECORD LED will remain off.

(c)Remarks

During "Ground Scan" or manual "AFS TEST", the "Transient flag" software is not applicable.

The "Transient flag" software has no impact on the memory context inside computers, the only difference will be in the MTP context table in which a bit is set if a transient flag is detected.

E. GROUND SCANNING Function

The MTP enables troubleshooting to be performed on the ground. This is achieved by activation of the fault isolation process, through a specific ground function which is called GROUND SCANNING.
(Ref. Fig. 016)

(1)Before activation

- the aircraft must be on the ground (gear pressed and wheelspeed lower than 70 kt)
- DISPLAY or AFS TEST/LAND TEST modes must not be active.

(2)Activation

It is activated by pressing the GROUND SCAN pushbutton switch.
The green ground scan LED is then on but the magnetic indicator is inactive.

R Alternate solution (MTP with FAILURE RECORD LED)

R The green ground scan LED is then on but the FAILURE RECORD LED is

R off.

Under these conditions, the MTP can isolate faulty LRU upon disconnections, engagement refusal, or landing capability change (downgrading).

CAUTION : Do not reset AFS computers using circuit breakers in ground scanning because GROUND SCAN configuration is then lost in the relevant computer (in that case the function must be reset and set again).

(3)GROUND SCAN messages

DISPLAY mode is available in a standard way and the origin of the messages clearly indicated.

e.g. :

```
| GROUND SCAN |
| 2 FAIL RECORD |
```

During GROUND SCANNING only one context can be stored on a specific table named TCNTXTS. Any new context erases the previous one.

(4)To stop the GROUND SCANNING function :

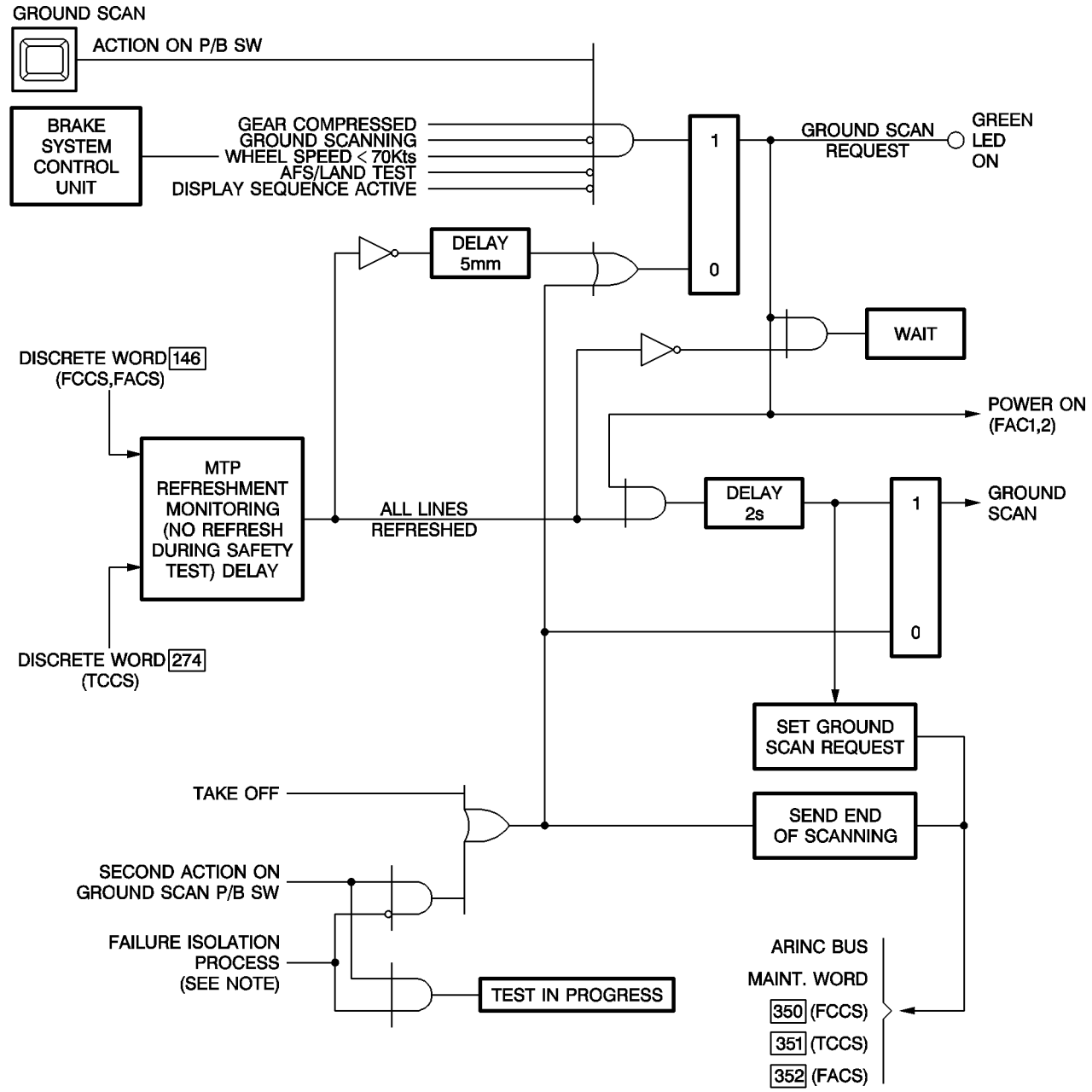
- press GROUND SCAN pushbutton switch
- reset will be automatic at takeoff.

(5)If ground scanning function is not in true ground condition, the MTP displays :

```
| REFUSED |
```

The following figure shows MTP messages associated with the different functions
(Ref. Fig. 017).

R (Ref. Fig. 018).



BM5 22 42 00 0 BEM2 01

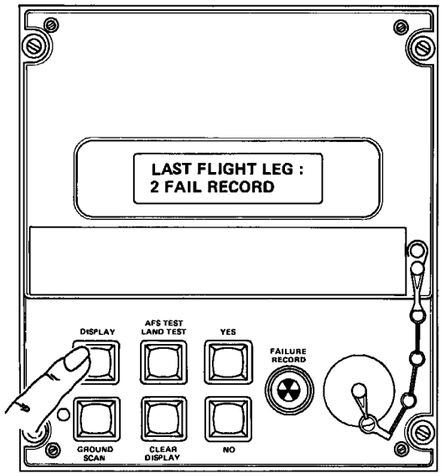
NOTE : IF GROUND SCAN P/B IS DEPRESSED DURING A FAILURE LOCALIZATION SEQUENCE (WHICH CANNOT OCCUR AT GROUND SCAN P/B ACTIVATION BUT ONLY AT DEACTIVATION) MTP DISPLAY.

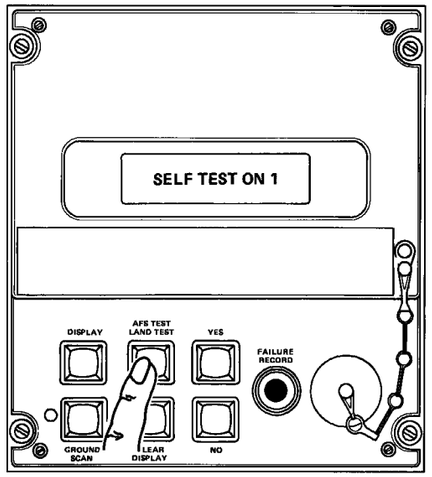
R

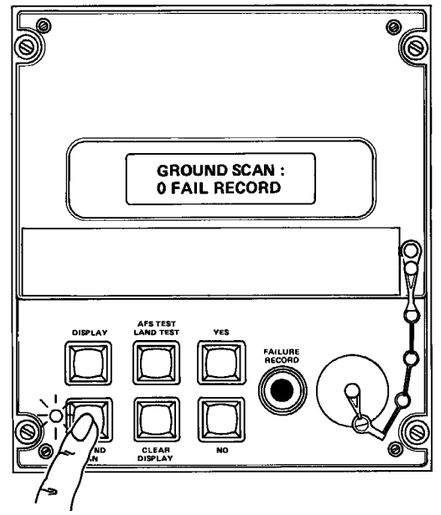
GROUND SCAN Activation Logic
Figure 016

INTENTIONALLY BLANK

Printed in France







FAULTY LRU ASSOCIATED WITH 1ST EVENT RECORDED

1 FCC1 FAILED

TABLE OF FAULTY LRU'S ASSOCIATED WITH 2ND EVENT RECORDED (DISPLAYED IN PROBABILITY ORDER)

2.A FCC2 FAILED

2.B AP DISC SW FAILED

END

NOTE:

TEST IN PROGRESS

IF PRESSING DISPLAY PB/SW WHEN FAILURE ISOLATION PROCESS IS RUNNING

EXAMPLE OF FAILURE DISPLAY

IF FAILURE :

CYCLIC DISPLAY (WHEN ALL SELF. TEST STEPS ARE COMPLETED)

MTP FAILED

NOSE GEAR RELAY FAILED

CLEAR DISPLAY?

NO

YES

RESULTS OF AFS TEST (1ST PHASE)

LAND TEST ?

IF NO FAILURE :

(WHEN ALL SELF. TEST STEPS ARE COMPLETED)

AFS TEST OK
LAND TEST ?

AFS/TEST DISPLAYS

NOTE:

REFUSED

- IF ACTIVATING GROUND SCAN OR AFS TEST IN FLIGHT
- IF ACTIVATING AFS TEST WHEN SYSTEMS ARE ENGAGED

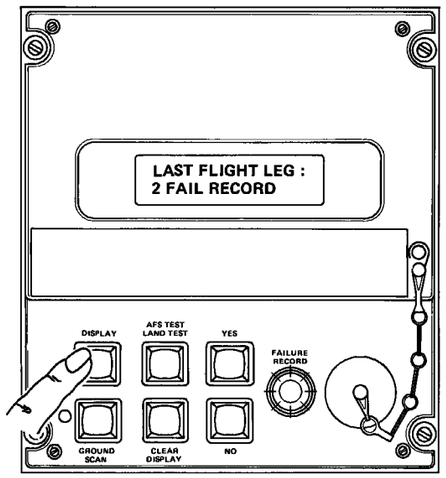
WAIT

IF ACTIVATING GROUND SCAN WITH AFS COMPUTER LINES NOT REFRESHED (0 TO 5MIN)

GROUND SCAN DISPLAY

BM5 22 42 00 0 BGM2 01

MTP Messages - Examples
Figure 017



NOTE : TO CHECK IF A FAILURE IS RECORDED IN THE MTP, IT IS REQUIRED TO POWER THE MTP FIRST.
- AIRCRAFT TO BE ENERGISED.
- MTP OVERHEAD PANEL CIRCUIT BREAKER CLOSED.

FAULTY LRU ASSOCIATED WITH 1ST EVENT RECORDED

1 FCC1 FAILED

TABLE OF FAULTY LRU'S ASSOCIATED WITH 2ND EVENT RECORDED (DISPLAYED IN PROBABILITY ORDER)

2.A FCC2 FAILED

2.B AP DISC SW FAILED

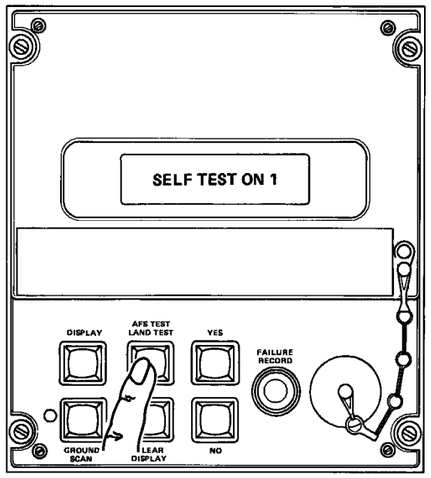
END

NOTE :

TEST IN PROGRESS

IF PRESSING DISPLAY PB/SW WHEN FAILURE ISOLATION PROCESS IS RUNNING

EXAMPLE OF FAILURE DISPLAY



IF FAILURE :

CYCLIC DISPLAY (WHEN ALL SELF. TEST STEPS ARE COMPLETED)

MTP FAILED

NOSE GEAR RELAY FAILED

NO

CLEAR DISPLAY?

YES

RESULTS OF AFS TEST (1ST PHASE)

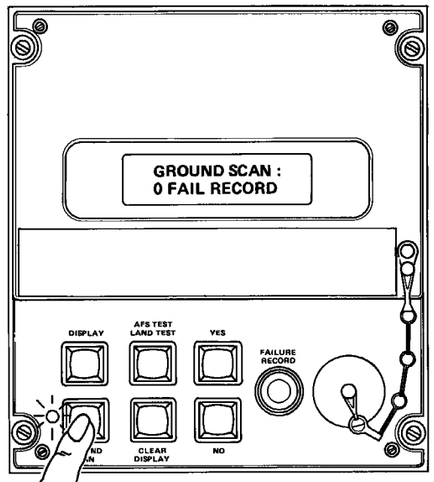
LAND TEST ?

IF NO FAILURE :

(WHEN ALL SELF. TEST STEPS ARE COMPLETED)

AFS TEST OK LAND TEST ?

AFS/TEST DISPLAYS



NOTE :

REFUSED

- IF ACTIVATING GROUND SCAN OR AFS TEST IN FLIGHT
- IF ACTIVATING AFS TEST WHEN SYSTEMS ARE ENGAGED

WAIT

IF ACTIVATING GROUND SCAN WITH AFS COMPUTER LINES NOT REFRESHED (0 TO 5MIN)

GROUND SCAN DISPLAY

BM5 22 42 00 0 BGM0 01

MTP Messages - Examples
(MTP with FAILURE RECORD LED)
Figure 018

R

EFFECTIVITY: ALL

PIA

MAINTENANCE TEST PANEL (3CU) - REMOVAL/INSTALLATION

1. Equipment and Materials

ITEM	DESIGNATION
A.	Circuit Breaker Safety Clips
B.	Blanking Caps
Referenced Procedure - 22-40-00, P. Block 501 (Ref. Fig. 401)	System Monitor

2. Procedure

A. Job Set-Up

(1) Open, safety and tag the following circuit breakers :

PANEL	SERVICE	IDENT.	LOCATION
21VU	AFS/MTP	301CU	102/H 1
21VU	AFS/FAC1/28VDC	306CC1	103/G 3
21VU	AFS/FAC2/28VDC	306CC2	103/G12
132VU	INT LT/ANN LT/TEST MAN	149LP	324/L82
133VU	L/G PROX DET & RELAYS/SYS1/RELAYS & RETRACT CTL	2GB	332/U51

B. Removal

(1) Unlock the four quick-release fasteners (2) securing maintenance test panel (MTP) (1).
(2) Withdraw MTP (1) from its housing (5).
(3) Disconnect and cap electrical connectors (4).
(4) Cap receptacles (3) on MTP (1).
(5) Remove MTP (1).

C. Installation

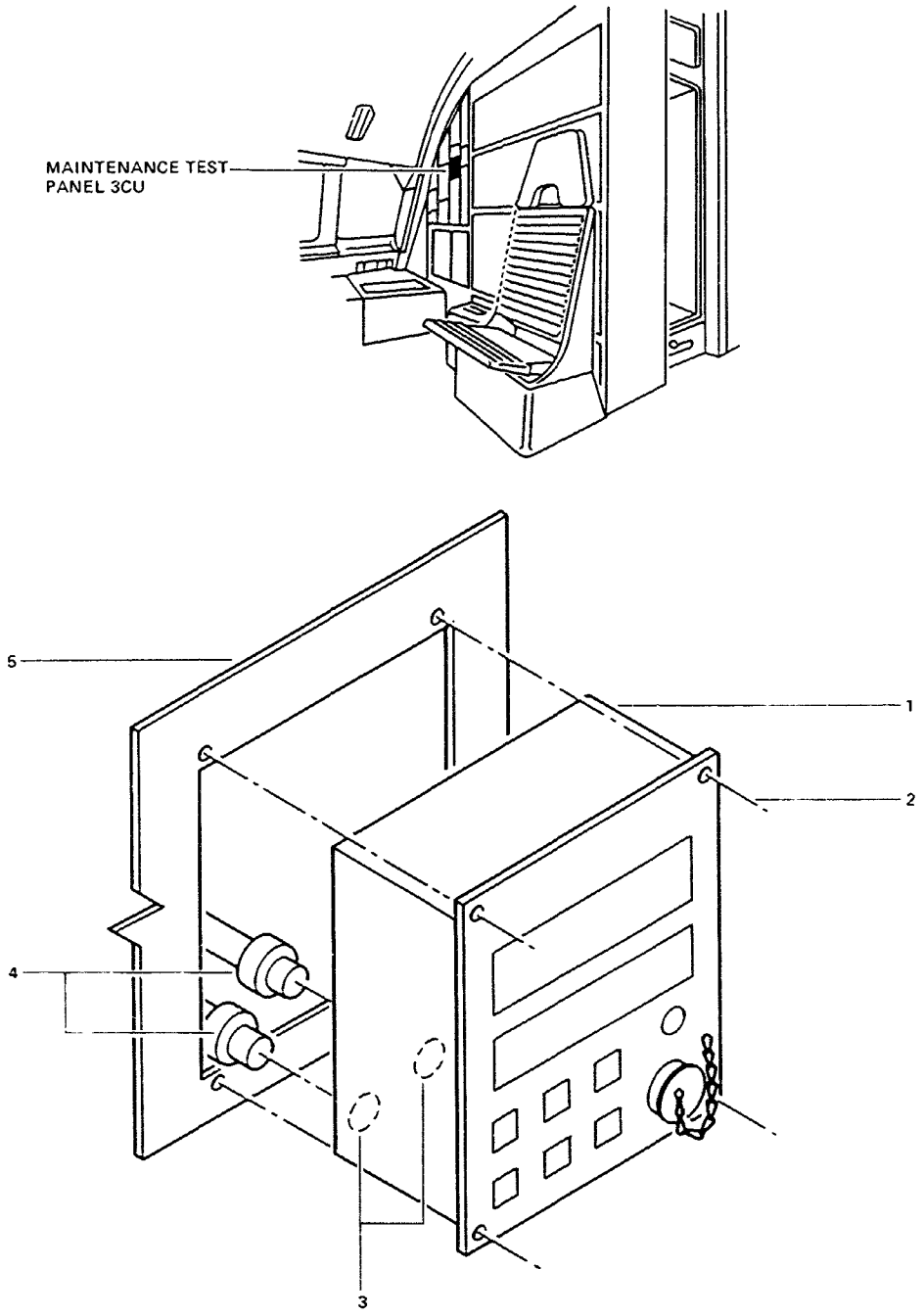
(1) Clean and inspect MTP interface and/or adjacent area.
(2) Remove blanking caps from electrical connectors (4) and receptacles (3).
(3) Connect the two electrical connectors (4).
(4) Install MTP (1) in its housing.
(5) Lock quick-release fasteners (2).
(6) Remove safety clips and tags and close circuit breakers 301CU, 306CC1, 306CC2, 149LP and 2GB.

D. Test

Carry out AFS test (Ref. 22-40-00, P. Block 501, para. 1.A.).

E. Close-Up

Make certain that working area is clean and clear of tools and miscellaneous items of equipment.



BM5 22 42 21 4 TGM2 12 07
DM5

Maintenance Test Panel
Figure 401

COMPLEX FAILURE ISOLATION SUPPORT

The AFS system provides information for complex failure isolation whenever repairs cannot be performed by first echelon maintenance personnel.
Two types of information are provided :
- History of events concerning the previous flights by the MTP :
  . by storage of the messages concerning the six previous flights (during which the anomalies occurred) (Ref. 22-42 paragraph 3.A.)
  . by storage of primary data in non volatile memory inside the MTP
- Storage of primary data in non volatile memory inside each AFS computer.

1. Context Table Inside the MTP

PRIMARY DATA are stored in a "context table" by non volatile memory inside the MTP. This context table is 240 bytes long, each context being 8 bytes long. thirty contexts can be stored over the last six flight legs.

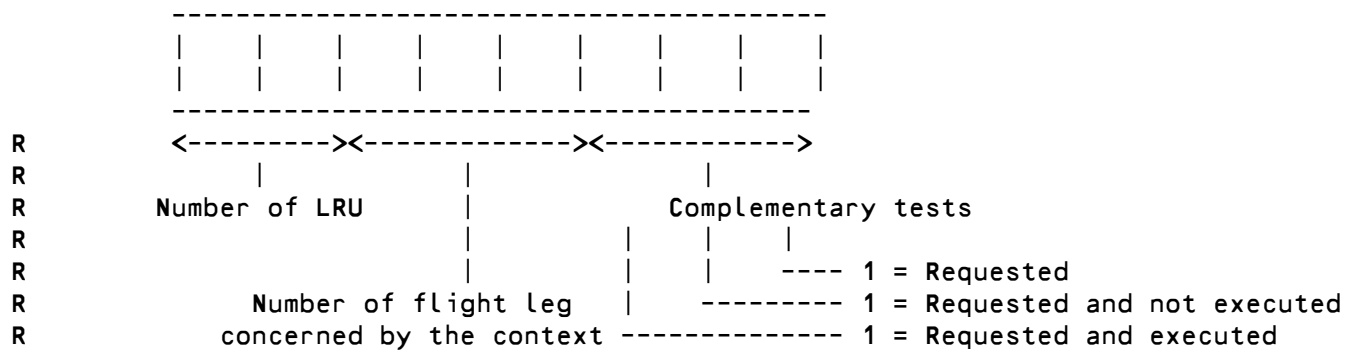
A. MTP Context Description

Structure

BYTE 1
BYTE 2
BYTE 3
BYTE 4
SPARE
LRU 1
LRU 2
LRU 3

Description

- BITE 1 ENCODING


Number of LRU
Complementary tests
Number of flight leg concerned by the context
---- 1 = Requested
--------- 1 = Requested and not executed
---------- 1 = Requested and executed

(1)BITE 2 encoding
The content of BITE 2 depends on the localization type given on BITE 3.

(a)ARINC line not refreshed (detected by the MTP)
In that case :
- localization type is 0 on BITE 3
- BITE 2 indicates which computer is faulty

FCC1 FCC2 FAC1 FAC2 TCC1 TCC2

Corresponding bit = 1 means ARINC LINE not refreshed.

(b)Maintenance word sent by a computer
In that case :
- localization type is 1 on BITE 3
- BITE 2 indicates by which computer the maintenance word has been sent.

FCC1 FCC2 FAC1 FAC2 TCC1 TCC2

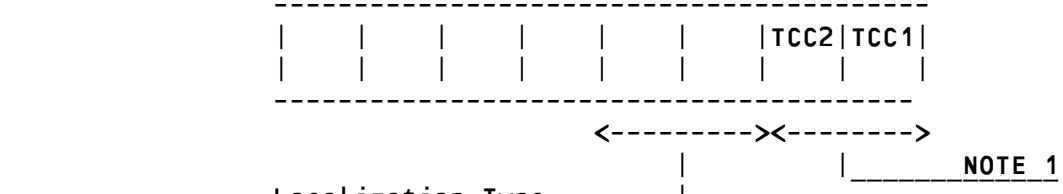
Corresponding bit = 1 means maintenance word received from relevant computer.

(c)Disconnection detected by the MTP
In that case :
- localization type is 2 on BITE 3
- byte 2 coding is as follows :

YAW DAMPER1 TRIM1 YAW DAMPER2 TRIM2 FD2 AP2 FD1 AP1

NOTE : Two bits of BITE 3 are utilized in that case for ATS1 and ATS2 information.

(2)BITE 3 encoding



0 : ARINC line not refreshed
1 : Maintenance word sent by a computer
2 : Disconnection detected by the MTP
3 : Confirmation requested/no answer

NOTE 1 : These two bits are utilized in addition to bite 2 for TCC1 and TCC2 disconnection information.
NOTE 2 : Confirmation is requested by the MTP when a computer detects an ARINC line is not refreshed and that this failure is not detected by the MTP.
Example : TCC1 detects that the FAC1 ARINC line is not refreshed. In that case the MTP confirms the information from the FCC.

(3)BITE 4 encoding

Localization condition
- 0 to 61 : Flight leg number
  62 : Ground scanning
  63 : AFS/LAND TEST

BITES 6, 7, 8 are used for the display of the faulty LRU's.

B. MTP Context Reset

The MTP context can be reset :
- at the end of the acceptance tests
- by the MIP when connected to the front face of the MTP.

C. Context Table Inside the Computer (Ref. Fig. 001)

(1)Contexts stored during **FLIGHT SCANNING** on a table named **TCNTXT**

Size of the contexts			
TCC :	B217	B298/B351/B472	B415
	16 bytes	18 bytes	20 bytes

FAC : 12 bytes
FCC : 24 bytes

Size of TCNTXT			
TCC :	B217	B298/B351/B472	B415
	208 bytes (13 contexts)	216 bytes (12 contexts)	200 bytes (10 contexts)

FAC : 204 bytes (17 contexts)
FCC : 216 bytes (9 contexts)
The number of contexts can be different between **COMMAND** and **MONITORING** channels.

- The first byte of the last stored context in **TCNTXT** is pointed at by **CNTXT POI** (byte) **(CONTEXT POINTER)**
- The available place in **TCNTXT** is indicated by the value of **PLADISP** (byte)
- When **TCNTXT** is full, **PLADISP** is reset to zero, in that case if an event occurs, the new context replaces the oldest which is lost.

(2)During **GROUND SCANNING** or **AFS TEST/LAND TEST** only one context can be stored on a separate table named **TCNTXTS**. Any new context replaces the previous one.

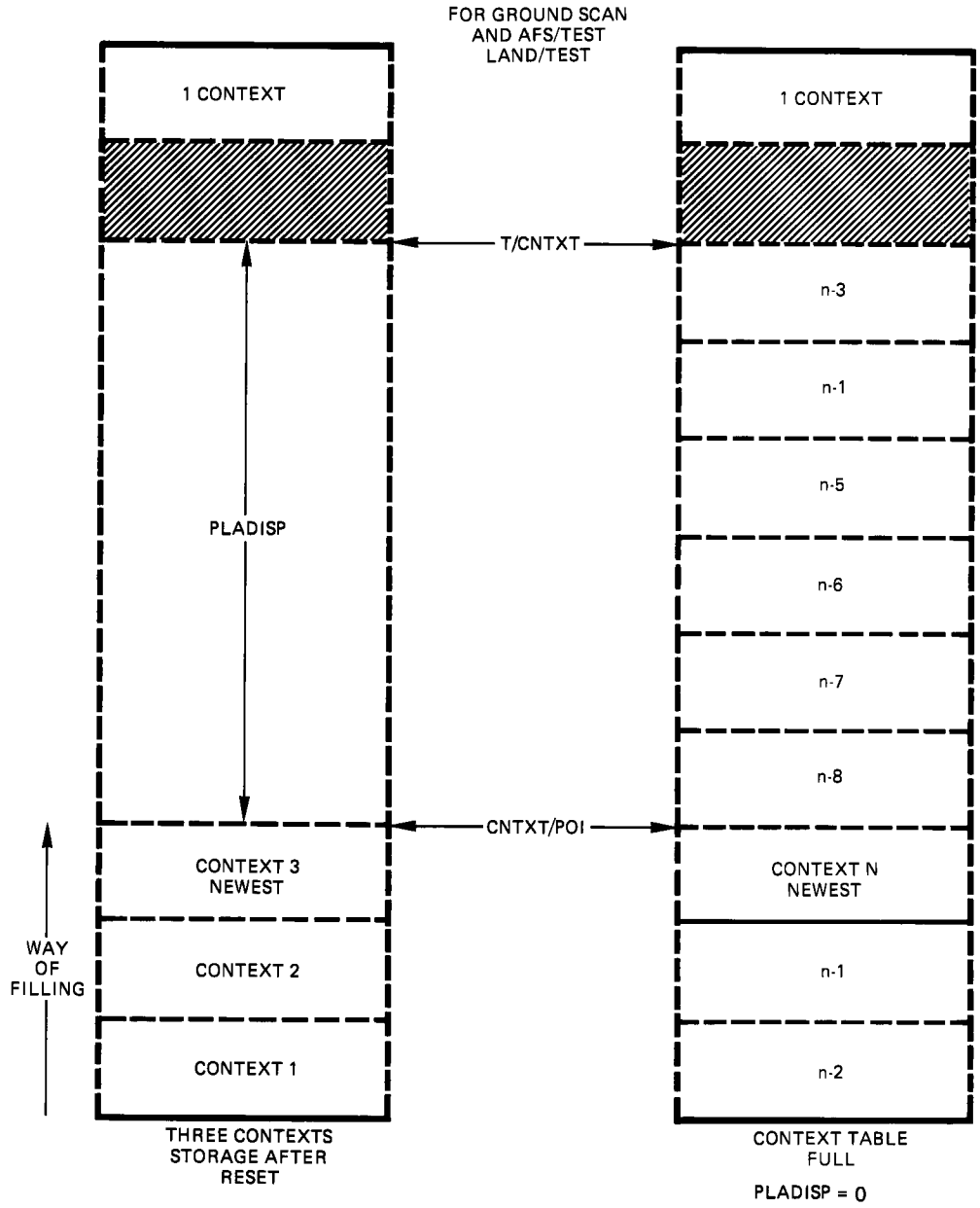
REMARKS :

- If the same fault occurs several times consecutively during the same flight leg, only one context is memorized but a counter is incremented (number of events).
- There is no context recording upon voluntary disconnection (lever or instinctive disconnect).
- The number of contexts recorded can be different between **COMMAND** and **MONITORING** channels.

ADDRESSES :

TCNTXTS :
PLADISP :
CNTXPOI :
TCNTXT :

To reach the newest context :
ADDRESS = ADDRESS (TCNTXT) + CNTXPOI
(Ref. Fig. 001)

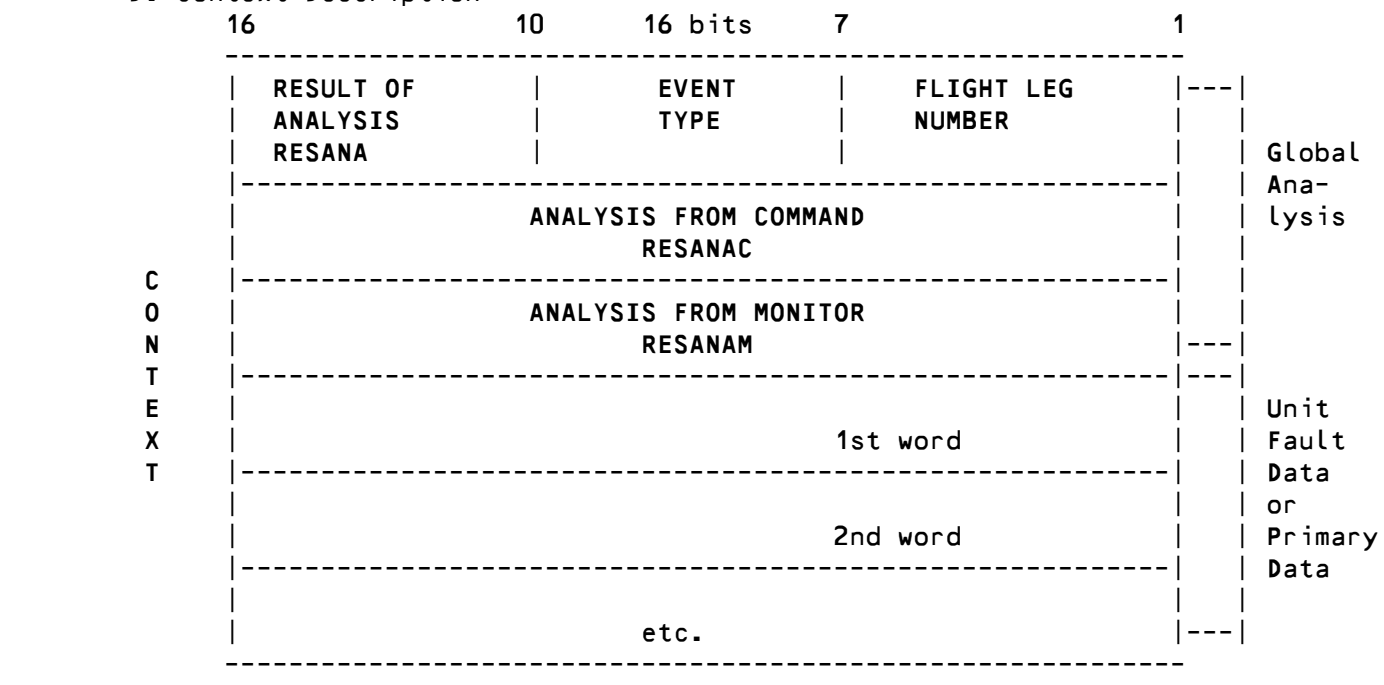


BM5 22 43 00 0 AAM2 02

Context Table
Figure 001

R

D. Context Description



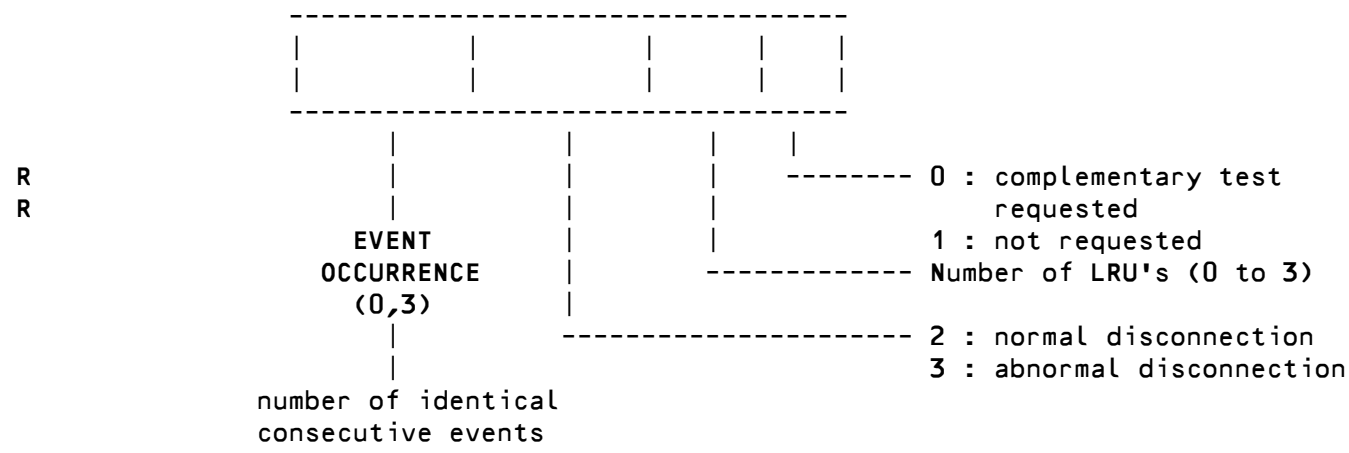
- in monitoring channel **RESANA** = 0
  RESANAC = 0
- in command channel if no result from monitoring channel, **RESANAM** = 1FH

A context can be divided into two parts :
- the first part includes the global analysis **(RESANA)** information on **EVENT TYPE** and **FLIGHT** leg number, analysis from command **(RESANAC)** and monitoring channels **(RESANAM)**
- the second part includes the **UNIT FAULT DATA** or **PRIMARY DATA** which concern the primary data utilized by the command or monitoring channel to establish **RESANAC** or **RESANAM.**

E. Details on the First Part of the Context

First Word	RESANA	EVENT TYPE	FLIGHT LEG NUMBER

(1)RESANA



(2)Event Type

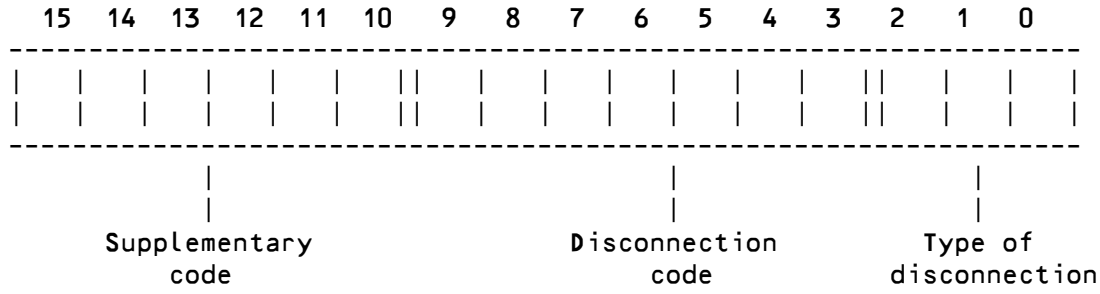
	FCC	FAC	TCC
0	AP DISCONNECTION	YAW DAMPER DISCONNECTION	ATS DISCONNECTION
1	FD LOSS	PITCH TRIM DISCONNECTION	
2	CAT3 LOSS	WINDSHEAR ALERT	
3	CAT2 LOSS		
4	SPARE		
5	CAT2/CAT3 NOT AVAILABLE		

(3)Flight leg number

0 to 61 : flight leg number
0 : newest flight leg
62 : ground scanning
63 : LAND TEST

F. Second and Third Words

RESANAC
RESANAM



(1)Type of disconnection (same on the 3 computers)
0 = voluntary disconnection by lever
1 = voluntary disconnection by instinctive disconnect
2 = normal disconnection
3 = abnormal disconnection
4 = unidentified disconnection
5 to 7 spares
example of normal disconnection :
- for the **FCC** if both trims are disengaged
- for the **TRIM** if it is disengaged by action on the pitch trim control wheel
- for the **TCC** if bad setting of **FLEX TEMP** at **FLEX TO** mode engagement.

(2)Supplementary code
This coding is different for **FCC** and **TCC** and is not used for the **FAC**.

(3)**FCC** supplementary coding
The supplementary coding is different between **RESANAC** and **RESANAM** for the same failure. This comes from the fact that a part of the logic is processed by the command line and another part by the monitor line. This applies for the following **FCC** failures.
1st case : disconnection code = **SERVO HYDRAULIC PRESSURE OFF** (associated failure processed by the command channel ; code 29H).
SUPPLEMENTARY CODES ON COMMAND CHANNEL

R bit 10 : FU oscillation monitoring
R bit 11 : pitch clutch switch (1 = closed)
R bit 12 : roll clutch switch (1 = closed)
R bit 13 : yaw clutch switch (1 = closed)

SUPPLEMENTARY CODES ON MONITORING CHANNEL
less signification ; depending on monitoring channel analysis result.

2nd case : disconnection code : GLOBAL IRS F/W (code 14H)
SUPPLEMENTARY CODES ON COMMAND CHANNEL
less signification (as monitor for the previous case)
SUPPLEMENTARY CODES ON MONITORING CHANNEL
R bit 15 : IRS OWN FW (also command channel)
R bit 14 : IRS OWN COMPARISON F/W (0 = fail)
R bit 13 : IRS OPPOSITE F/W (0 = fail)
R bit 12 : IRS OPP COMPARISON F/W (0 = fail)
R bit 11 : IRS3 F/W (0 = fail)
R bit 10 : IRS3 COMPARISON F/W (0 = fail)

When the failure is determined as DISCONNECTION BY LEVER either by the command or by the monitoring channel, the disconnection code is not utilized and the associated bits are grouped to the coding as follows :
R Bit 15 : Stick Force AP in CMD
R Bit 14 : Pitch CWS sensor FW
R Bit 13 : 0 YES |---> Several axes (PITCH-ROLL-YAW)
R 1 NO | affected
R Bit 12 : ROLL VOTER COMP
R Bit 11 : PITCH VOTER COMP
R Bit 10 : YAW VOTER COMP
R Bit 9 : TRIM OWN -
R Bit 8 : TRIM OPP |---> 1 = engaged
R Bit 7 : STAB OWN |
R Bit 6 : STAB OPP -
R When the failure is determined as FAC F/W (code 11H, refer to table)
R the supplementary coding is as follows (this applies for command and
R monitor)
R Bit 13 : TRIM OWN
R Bit 12 : TRIM OPP
R Bit 11 : STAB 1
R Bit 10 : STAB OPP
R When the failure is detected by a comparator (codes 1 to 9), the coding
R of bits 10 to 13 is the same as for disconnection by LEVER.
R (4)TCC supplementary coding
R Bit 15 : FMC ARINC SWITCHING (monitoring channel only) ; 1 = opp
R Bit 13 : RA Selection Monitor only 0 = own
R 1 = opp
R Bit 12 : A/THR mode engaged
R Bit 11 : FAC ARINC SW 0 = own
R 1 = opp
R Bit 10 : TO/GA switch 0 = active

AFS COMPUTERS CONTEXT RESET

The AFS computer context can be reset :

- at the end of the acceptance tests
- by the MIP when connected to the front face of the MTP.

R FCC DISCONNECTION CAUSE CODES - TABLE 2
R (7 Bits) - Abnormal disconnection

	CAUSES FOR ABNORMAL DISCONNECTION	CMD	MON		CMD	MON
R	PITCH SERVO LOOP SYNC	01H	01H	GLIDE N.C.D	19H	19H
	DELAYED			LOC F/W	1AH	1AH
R	PITCH SERVO LOOP DELAYED	02H	02H	LOC N.C.D	1BH	1BH
R	ROLL SERVO LOOP SYNC	03H	03H	FCU F/W (no refresh)	1CH	1CH
R	DELAYED			FLAPS/SLATS REDUNDANCE	1DH	1DH
R	ROLL SERVO LOOP DELAYED	04H	04H	LOST		
R	YAW SERVO LOOP SYNC	05H	05H	PITCH CWS SENSOR F/W	1EH	1EH
	DELAYED			COMP GROUND SPEED F/W	1FH	1FH
R	YAW SERV LOOP DELAYED	06H	06H	FWC ACQ FAIL	20H	20H
R	PITCH VOTER COMP DELAYED	07H	07H	FWC OPPOSITE ACQ FAIL	21H	21H
R	ROLL VOTER DELAYED	08H	08H	ILS OPP F/W	22H	22H
R	YAW VOTER DELAYED	09H	09H	FCC OPP F/W	23H	23H
R	26V/400HZ SURVEILLANCE	0AH	0AH	PFD ATTITUDE F/W	24H	24H
	PITCH CLUTCH SWITCH	0BH	0BH	PFD OPP ATTITUDE F/W	25H	25H
	ROLL CLUTCH SWITCH	0CH	0CH	PFD F/W	26H	26H
R	YAW CLUTCH SWITCH	0DH	0DH	PFD OPP F/W	27H	27H
	WATCH DOG CPU OPPOSITE	0EH	0EH	IRS OWN F/W	28H	
R	HYDRAULIC CIRCUIT NOT	0FH	0FH	SERVO HYDRAULIC PRESSURE	29H	
	HEALTHY			OFF		
R	FCC F/W	10H	10H	CWS SENSOR NOT CONNECTED	2AH	
R	FAC F/W	11H	11H	NOSE GEAR VALIDITY	2BH	
	GROUND SPEED F/W	12H	12H	FCU F/W (DISCRETE)	2CH	
R	SPARE OPPOSITE GROUND	13H	13H	VOR F/W	2DH	
R	SPEED F/W					
	GLOBAL IRS F/W	14H	14H	VOR N.C.D	2EH	
	ADS F/W	15H	15H	FMC F/W	2FH	
	R/A F/W	16H	16H	FMC N.C.D	30H	
R	R/A N.C.D	17H	17H	TRK OWN, HDG OWN	31H	31H
R				COMPARISON		
R	GLIDE F/W	18H	18H	RUNWAY HEADING F/W	32H	32H
R				IRS3 FOR CAPACITY	33H	33H

R FCC DISCONNECTION CAUSE CODES - TABLE 3
R (7 Bits) - Normal disconnection

CAUSES FOR NORMAL DISCONNECTION	CMD	MON
NEITHER TRIM ENGAGED	01H	01H
NEITHER STAB ENGAGED	02H	02H
ROLL > 45 DEG	03H	03H
AP OPPOSITE ENGAGED	04H	04H
FD BARS REMOVED	05H	05H
GEARS PRESSED AND ENGINES ON	06H	06H
ALPHA F/W	07H	07H
MINIMUM & PLACARD SPEED F/W	08H	08H
FLAPS/SLATS NOT HEALTHY	09H	09H
GEAR VALIDITY	0AH	0AH
AIRCRAFT POWER SUPPLY NOT SPLIT	0BH	0BH
NEITHER TCC IN SPEED MODE	0CH	0CH
TRIM OWN/STAB OWN NOT ENGAGED	0DH	0DH
TRIM OWN/STAB OPP NOT ENGAGED	0EH	0EH
A/P OPPOSITE/FD OPPOSITE NOT ENGAGED	0FH	0FH
NOT (LAND ARM/LAND TRACK) MODE WNW	10H	10H
NOT (LAND ARM/LAND TRACK) MODE OPPOSITE	11H	11H
SPLIT PFD	12H	12H
NAV SELECTOR NOT ON ILS	13H	
NAV SELECTOR ON ILS NOT ON VOR	14H	
AP OWN/FD OWN NOT ENGD	15H	15H
NEITHER AP IN CMD	16H	16H
STICK FORCE (AP IN CMD)	*17H	*17H

R * For B350AMM and B470AAM only

FAC DISCONNECTION CAUSE CODES - TABLE 4
- Abnormal disconnection

	CAUSES FOR ABNORMAL DISCONNECTION	CMD	MON
R	26V/400Hz SURVEILLANCE	01H	01H
R	TRIM COMPARATOR 3 (COMMAND AND MONITORING CHANNEL TRIM ORDERS)	02H	02H
R	TRIM COMPARATOR 4 (FU1-FU2)	03H	03H
R	TRIM COMPARATOR 5 (FU1- TRIM ORDER)	04H	04H
R	TRIM COMPARATOR 12 (INTEGRATOR MONITORING)	05H	05H
R	TTP RATE MONITORING (SPEED MONITORING	06H	06H
R	YAW FU2 COMPARATOR (FU2- YAW IMAGE)	07H	07H
R	YAW VOTER COMPARATOR DELAYED	08H	08H
R	HYDRAULIC CIRCUIT NOT HEALTHY	09H	09H
	FAC OWN F/W	0AH	0AH
R	FAC OPP F/W	0BH	0BH
	ADS OWN F/W	0CH	0CH
	IRS F/W	0DH	0DH
	IRS 3 F/W	0EH	0EH
	EFCU F/W	10H	10H
R	BOTH SFCC F/W	11H	11H
	ANGLE-OF-ATTACK FAILURE	12H	12H
	ELECTRIC TRIM FAILURE	13H	13H
	R/A OWN F/W	14H	14H
R	IRS MONITORING	15H	15H
R			(4)
	WATCH DOG CPU OPPOSITE	16H	16H
	28V GEAR F/W	17H	17H
	GEAR F/W	18H	18H
	NOSE GEAR F/W	19H	
	ADS OPPOSITE F/W	1AH	1AH
	SFCC OWN F/W		1BH
R	TRIM COMPARATOR 6 (Autotrim	1CH	1CH(1)
R	Monitoring)		
R	DADS/IRS OWN F/W (Windshear	1DH	- (2)
R	Alert Inhibit)		
R	TRIM COMPARATOR 7 (ALPHATRIM	1EH	1EH(3)
R	MONITORING		

R (1) : For B352AAM + B471AAM + B471ABM
R (2) : For B471AAM + B471ABM
R (3) : For B471AAM4 + B471ABM2

R (4) : For B471AAM5 ONLY

R FAC DISCONNECTION CAUSE CODES
R - Normal disconnection

CAUSE FOR NORMAL DISCONNECTION	CMD	MON
ENGINES STOPPED	01H	

R TCC DISCONNECTION CAUSE CODES - TABLE 5
R - Abnormal disconnection

CAUSES FOR ABNORMAL DISCONNECTION	CMD	MON	CAUSES FOR ABNORMAL DISCONNECTION	CMD	MON
MOTOR COMPARATOR COMP 2	00H	00H	IRS OWN F/W		10H
VOTER COMPARATOR COMP 1	01H	01H	IRS OPPOSITE F/W		11H
WATCH DOG CPU OPPOSITE	02H	02H	IRS 3 F/W		12H
TCC F/W	03H	03H	FMC F/W	13H	13H
ADS COMMUNICATION	04H	04H	FAC F/W	14H	14H
26V/400Hz SURVEILLANCE	05H	05H	RA OWN F/W	15H	
ACTUATOR NOT CONNECTED	06H		RA OPPOSITE F/W		16H
FORCE SENSOR NOT CONNECTED	07H		EEC OWN F/W	17H	
RIGHT THROTTLE LEVER	08H	08H	EEC OPPOSITE F/W	18H	
ANGLE F/W			FCC F/W		19H
LEFT THROTTLE LEVER	09H	09H	TCC F/W OPPOSITE	1AH	1AH
ANGLE F/W			NOSE/MAIN GEAR	1BH	
			COMPRESSION		
ADS F/W	0AH	0AH	BOTH SFCC FAILED	1CH	1CH
ARINC ENGINE F/W	0BH	0BH	ALTITUDE TARGET F/W		1DH
SPEED ERROR F/W	0CH		DFA SPEED PROG F/W		1EH
N1/EPR LIMIT F/W		0CH	DFA F/W	1FH	
TRP F/W	0DH		HRA F/W		20H
EEC F/W	0EH		IRS OWN F/W FROM FCC		21H
FCU F/W		0FH	INSTINCTIVE DISCONNECT		22H
			FAILURE		
			ALTITUDE TARGET F/W		21H
			DFA SPEED MONITORING		22H
			EEC OWN RESOLVERS		23H
			EEC OPP RESOLVERS		24H

R TCC DISCONNECTION CAUSE CODES - TABLE 6
R - Normal disconnection

CAUSES FOR NORMAL DISCONNECTION	CMD	MON
LEFT & RIGHT ENGINES NOT COMPATIBLE	01H	
GEAR UNVALIDITY FROM AC	02H	02H
N1/EPR > HIGH LIMIT	03H	03H
ENGINE TYPE F/W	04H	
SEL TAT MONITORING F/W	05H	
FAC ANALOG DATA F/W		06H
NO TRP MODE SELECTED	07H	

R TCC
R TCC COMPLEMENTARY CODING
R

Bit 10 : TO/GA SWITCH	=	0 Active
Bit 11 : FAC ARINC Switching	=	0 Own
	=	1 OPPOSITE
Bit 12 : ATHR Mode	=	1 Active
Bit 13 : R/A Selection	=	0 Own
(Monitor Lane Only)	=	1 OPPOSITE
Bit 15 : FMC ARINC Switching	=	1 OPPOSITE

B. Failure Messages

IDENTIFICATION OF SENSORS

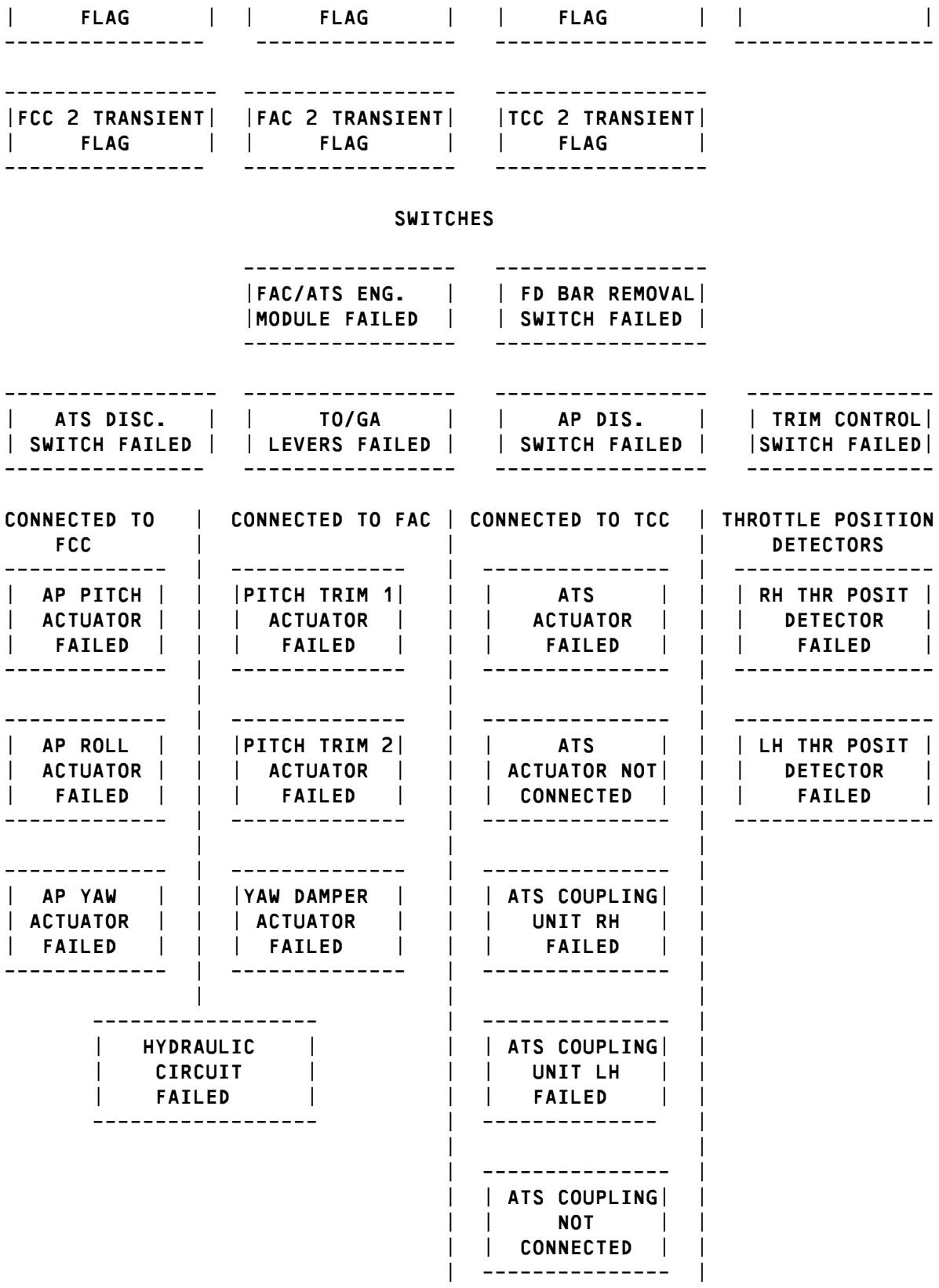
ADS 1 FLAG	R/A 1 FLAG	VOR 1 FLAG	ENGINE SENSORS 1 FAILED
ADS 2 FLAG	R/A 2 FLAG	VOR 2 FLAG	ENGINE SENSORS 2 FAILED
IRS 1 FLAG	EFIS 1 FLAG	BRAKE SYST CONT UNIT FAILED	FWC FAILED
IRS 2 FLAG	EFIS 2 FLAG	SFFC 1 FAILED	EEC 1/2 FAILED
IRS 3 FLAG	ANGLE OF ATTACK SENSOR 3	SFFC 2 FAILED	EEC 1/2 FAILED SENSORS FAILED
ILS 1 FLAG	FMC 1 FLAG	EFCU 1 FLAG	PMC 1

			2 SENSORS FAILED
ILS 2 FLAG	FMC 2 FLAG	EFCU 2 FLAG	EFCU 1/2 FLAG
GND SPOILER CNTL BOX FAILED	FMC 1/2 NCD	PMC 1/2 TRANSIENT FAULT	HMC 1/2 FAILED
	ONLY WITH K159ADM AND K159AEM		

GEAR RELAYS	DYNAMOMETRIC RODS	POWER SUPPLIES	
		28V GEAR PWR SUPPLY 1 LOSS	26 VAC POWER SUPPLY 1 LOSS
NOSE GEAR RELAY FAILED	AP DYNROD NOT CONNECTED	28V GEAR PWR SUPPLY 2 LOSS	26 VAC POWER SUPPLY 2 LOSS
RH MAIN GEAR RELAY FAILED	ATS DYNROD NOT CONNECTED	28 VDC POWER SUPPLY 1 LOSS	115 VAC POWER SUPPLY 1 LOSS
LH MAIN GEAR RELAY FAILED	PITCH DYNROD CAPT/FO FAILED	28 VDC POWER SUPPLY 2 LOSS	115 VAC POWER SUPPLY 2 LOSS

AFS COMPUTERS AND CONTROL UNITS

FCC 1 FAILED	FAC 1 FAILED	TCC 1 FAILED	FCU FAILED
FCC 2 FAILED	FAC 2 FAILED	TCC 2 FAILED	TRP FAILED
FCC 1 TRANSIENT	FAC 1 TRANSIENT	TCC 1 TRANSIENT	MTP FAILED


FLAG
FLAG
FLAG
FCC 2 TRANSIENT FLAG
FAC 2 TRANSIENT FLAG
TCC 2 TRANSIENT FLAG
SWITCHES
FAC/ATS ENG. MODULE FAILED
FD BAR REMOVAL SWITCH FAILED
ATS DISC. SWITCH FAILED
TO/GA LEVERS FAILED
AP DIS. SWITCH FAILED
TRIM CONTROL SWITCH FAILED
CONNECTED TO FCC
CONNECTED TO FAC
CONNECTED TO TCC
THROTTLE POSITION DETECTORS
AP PITCH ACTUATOR FAILED
PITCH TRIM 1 ACTUATOR FAILED
ATS ACTUATOR FAILED
RH THR POSIT DETECTOR FAILED
AP ROLL ACTUATOR FAILED
PITCH TRIM 2 ACTUATOR FAILED
ATS ACTUATOR NOT CONNECTED
LH THR POSIT DETECTOR FAILED
AP YAW ACTUATOR FAILED
YAW DAMPER ACTUATOR FAILED
ATS COUPLING UNIT RH FAILED
HYDRAULIC CIRCUIT FAILED
ATS COUPLING UNIT LH FAILED
ATS COUPLING NOT CONNECTED

EEC FADEC MAINTENANCE STATUS GROUPING
PW Type 4000 B415 AAM 1/2/3/4

MESSAGES ON THE MTP	ASSOCIATED MAINTENANCE ACTION	BIT CONFIGURATION IN EEC MAINTENANCE WORDS				
		350	351	352	353	354
EEC1 FAILED (2)	EEC1 (2) TO BE CHECKED BY FADEC GROUND/BITE TEST ON THE ECAM					
XA LH THR POSIT (RH) DETECTOR XB EEC1 FAILED (2)	a) check TLA resolver b) check wiring between EEC 1(2) and resolver c) check resolver shaft indexing d) check EEC last					
XA EEC1 FAILED (2) XB TCC1 FAILED (2)	a) check wiring between EEC 1 and TCC1 (2) (2) b) check TCC					

PMC MAINTENANCE STATUS GROUPING
GE Type CF6 80C2 A1/A2 - TCC B351 ACM1

MESSAGES ON THE MTP	ASSOCIATED MAINTENANCE ACTION	BIT CONFIGURATION IN PMC MAINTENANCE WORDS				
		350	351	352	353	354
PMC1 FAILED (2)	Check PMC 1 internal BITE (2)			17 21 22 24 25 27	17 18 19 20	19 20 21 22 23 24
PMC1 SENSORS (2) FAILED	a) check engine T12 signal b) check engine PC signal c) check N1 signal out of range high d) N1 ACT RATE signal out of range e) check for loose connector f) replace PMC	17	17	23		18

XA LH THR POSIT (RH) DETECTOR XB PMC1 FAILED (2)	a) check TLA resolver b) check wiring between PMC 1 and resolver (2) c) check resolver shaft indexing d) check PMC last			18 19 20			
XA HMC1 FAILED (2) XB PMC1 FAILED (2)	Tm current out of range a) check for loose connectors b) check for cable/MEC faults c) replace PMC Tm = Torque Motor					17	
XA PMC1 FAILED (2) XB TCC1 FAILED (2)	a) remove PMC b) check TCC c) check wiring between PMC 1 and TCC1 (2) (2)	18	18 20 23				
XA PMC1 FAILED (2) XB ADC1 FAILED (2)	a) remove PMC1(2) b) check ADC1(2) operation c) check wiring between PMC 1(2) and ADC1(2) d) check TCC operation	19	19 21 24				